NASA/SP–2009–566-ADD

# Human Exploration of Mars

# Design Reference Architecture 5.0

# Addendum

*Mars Architecture Steering Group*
*NASA Headquarters*

*Bret G. Drake, editor*
*NASA Johnson Space Center, Houston, Texas*

July 2009

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# Foreword

This report serves as an Addendum to NASA/SP–2009–566, "Human Exploration of Mars Design Reference Architecture 5.0." The data and descriptions contained within this Addendum provide additional detail and analyses conducted in the development of DRA 5.0. NASA/SP-2009-566 serves as the primary document describing DRA 5.0, including potential areas where discrepancies exist with this Addendum. The individuals listed in the appendix assisted in the generation of the concepts as well as the descriptions, images, and data described in this report. Specific contributions to this document were provided by Dave Beaty, Stan Borowski, Bob Cataldo, John Charles, Cassie Conley, Doug Craig, John Elliot, Chad Edwards, Walt Engelund, Dean Eppler, Stewart Feldman, Jim Garvin, Steve Hoffman, Jeff Jones, Frank Jordan, Sheri Klug, Joel Levine, Jack Mulqueen, Gary Noreen, Hoppy Price, Shawn Quinn, Jerry Sanders, Jim Schier, Lisa Simonsen, George Tahu, and Abhi Tripathi.

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# 1 INTRODUCTION

## 1.1 Background and Purpose

The NASA authorization act of 2005 articulated a new Vision for Space Exploration, specifically stating that "The Administrator shall establish a program to develop a sustained human presence on the Moon, including a robust precursor program, to promote exploration, science, commerce, and United States preeminence in space, and as a stepping-stone to future exploration of Mars and other destinations." This Vision calls for a progressive expansion of human capabilities beyond low-Earth orbit (LEO), seeking to answer profound scientific and philosophical questions while responding to discoveries along the way. This vision sets forth goals of: returning the space shuttle safely to flight; completing the International Space Station (ISS); retiring the space shuttle when the ISS is complete; sending precursor robotic orbiters and landers to the Moon; sending human expeditions to the Moon, conducting robotic missions to Mars in preparation for a future human expedition; and conducting robotic exploration across the solar system. In addition, the Vision articulates the strategy for developing the revolutionary new technologies and capabilities that are required for the future exploration of the solar system. This vision specifically calls for: (1) implementation of a sustained and affordable human and robotic program to explore the solar system and beyond; (2) extending human presence across the solar system, starting with a human return to the Moon no later than the year 2020, in preparation for human exploration of Mars and other destinations; (3) developing the innovative technologies, knowledge, and infrastructures to support human and robotic exploration; and (4) promoting international and commercial participation in exploration to further U.S. scientific, security, and economic interests. The Vision represents a bold new step for the nation and NASA.

In January 2004, NASA established the Exploration Systems Mission Directorate (ESMD) to lead the development of new exploration systems to accomplish the task of implementing the Vision. To determine the best exploration architecture and strategy to implement these many changes, the Exploration Systems Architecture Study (ESAS) was conducted from May to July 2005. The ESAS provided the top-level architectural foundation and driving requirements for the lunar transportation systems. From June 2006 through July 2007, NASA conducted the Lunar Architecture Team (LAT) series of studies that was aimed at further definition of the goals and objectives, activities, and systems that are necessary to conduct the lunar surface portion of the Vision. The ESAS focused on the transportation system (getting to and from the lunar surface), whereas the LAT studies concentrated on the activities that would be conducted on the surface.

## 1.2 Mars Architecture Working Group

During execution of the second half of the LAT studies, NASA Headquarters recognized that lunar architecture work must be conducted in an environment of what comes next, most predominately of which is human exploration of Mars. Significant progress was being made in the definition of the lunar transportation system (Ares crew and cargo launch vehicles, the Orion crew vehicle, lunar lander, the supporting ground and mission operations infrastructure) as well as the lunar surface architecture and systems; however, further refinement and confirmation of how these systems would either be used or modified for future exploration capabilities was required. In addition, the Science Mission and Aeronautics Research Mission Directorates were in the process of defining future Mars robotic missions as well as fundamental research activities related to future human exploration missions. NASA Headquarters, in recognition of the need for an updated and unified vision for human exploration of Mars, commissioned The Mars Architecture Working Group (MAWG) in January 2007 specifically to:

- Update NASA's human Mars mission reference architecture, which defines:
    - Long-term goals and objectives for human exploration missions
    - Flight and surface systems for human missions and supporting infrastructure
    - An operational concept
    - Key trade studies for future analysis
    - Key challenges, including risk and cost drivers
    - Development schedule options
- Develop an approach for reducing the cost/risk of human Mars missions through investment in research, technology development, and synergy with other exploration plans, including:

    - Robotic Mars missions
    - Cis-lunar activities
    - ISS activities
    - Earth-based activity, including analog sites, laboratory studies, and computer simulations
- Perform additional research and technology development investment
- Assess strategic linkages between lunar and Mars strategies

The MAWG was established as an agency-wide team including representatives and working groups from the ESMD), Science Mission Directorate (SMD, Aeronautics Research Mission Directorate (ARMD), and Space Operations Mission Directorate (SOMD). During the 2007 study effort, employees from NASA Headquarters and the field centers were involved in design, analysis, planning, and costing activities. The MAWG relied heavily on the deep body of work that is related to human exploration of Mars as the starting point for further deliberations and analyses.

This report does not constitute a formal plan for human exploration of Mars. Instead, it serves as a vision for future human exploration of Mars that is one potential approach based on current best estimates of what we know today. This approach is used to provide a common framework for future planning of systems concepts, technology development, and operational testing. In addition, it provides a common reference for integration between multiple agency efforts including Mars robotic missions, research conducted on the ISS, and future lunar exploration missions and systems.

A Joint Steering Group was established at the beginning of the study to provide representation of the major NASA Headquarters mission directorates. The Steering Group reviewed the primary products that were produced by the MAWG,, providing insight, guidance, and, ultimately, concurrence of recommendations that were made by the team. The MAWG itself was organized into a Strategy Team (providing resources and strategic study guidance), an Integration Team (focusing on the daily study performance, risk, and cost trades as well as product development), and Study Elements (providing the expertise that is associated with the technical study) (figure 1-1).

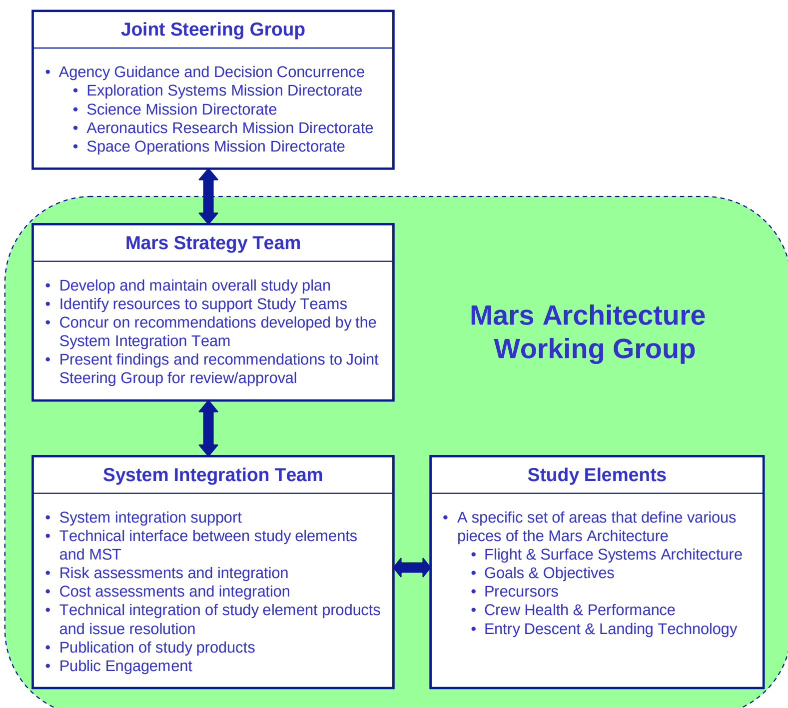


**Figure 1-1.** Mars Architecture Study Working Group organization.

The MAWG began its deliberations by gathering and reviewing all pertinent data and technical analyses of human exploration of Mars that were performed in the recent past. This included the retrieval of dozens of studies, technical papers, and policy documents. In addition, a set of study ground rules and assumptions (GR&As) as well as figures of merit (FOMs) to be used in the architectural decision process were developed and reviewed with the Joint Steering Group for concurrence. Specifics of the GR&As and FOMs are discussed later in this document.

## *1.3 History of the Design Reference Architecture*

During the past several years NASA has either conducted or sponsored numerous studies of human exploration beyond LEO (figure 1-2). These studies have been used to understand requirements for human exploration of the Moon and Mars in the context of other space missions and research and development programs. Each exploration architecture provides an end-to-end mission reference against which other mission and technology concepts can be compared. The results from the architecture studies are used to:

- Derive technology research and development plans
- Define and prioritize requirements for precursor robotic missions
- Define and prioritize flight experiments and human exploration mission elements, such as those involving the space shuttle, ISS, and space transportation
- Open a discussion with international partners in a manner that allows identification of potential interests of the participants in specialized aspects of the missions
- Provide educational materials at all levels that can be used to explain various aspects of human interplanetary exploration
- Describe to the public, media, and political system the feasible, long-term visions for space exploration

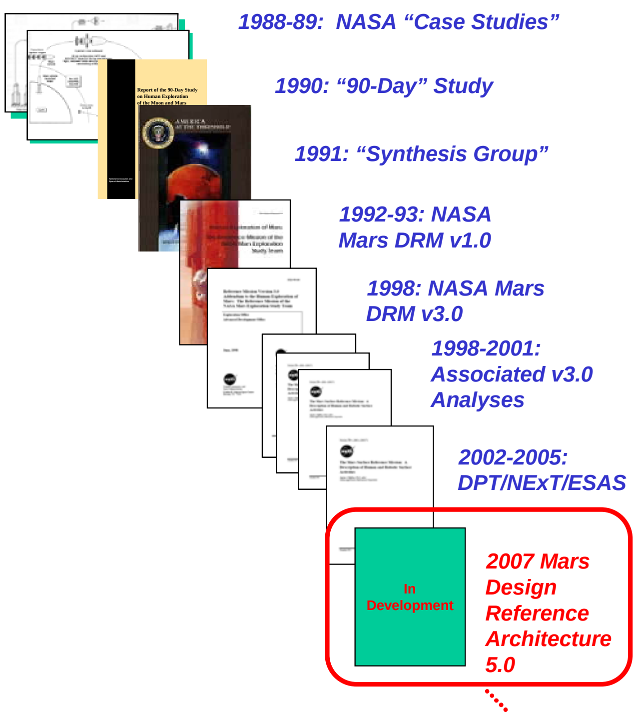


**Figure 1-2.** History of the Mars design reference architecture (DRA).

Each architecture study emphasized one or many aspects that are critical for human exploration to determine basic feasibility and technology needs. Example architectural areas of emphasis include:

| | |
|---|---|
| Destination: | Moon ⇔ Mars ⇔ Libration Points ⇔ Asteroids ⇔ Phobos/Deimos |
| System Reusability: | Expendable ⇔ Reusable |
| Architecture Focus: | Sorties ⇔ Colonization |
| Surface Mobility: | Local ⇔ global |
| Launch Vehicles: | Existing ⇔ New Heavy Lift |
| Transportation: | Numerous technologies traded |
| LEO Assembly: | None ⇔ Extensive |
| Transit Modes: | Zero gravity ⇔ Artificial gravity |
| Surface Power: | Solar ⇔ nuclear |
| Crew Size: | 4 ⇔ 24 |
| ISRU[1]: | None ⇔ Extensive |

### 1.3.1 Office of Exploration case studies (1988)

In June 1987, the NASA Administrator established the Office of Exploration in response to an urgent national need for a long-term goal to energize the U.S. civilian space program. The Office of Exploration originated as a result of two significant assessments that were conducted just prior to its creation. In 1986, the National Commission on Space, as appointed by the President and charged by Congress, formulated a bold agenda to carry the U.S. civilian space enterprise into the 21st century. Later that year, the NASA Administrator asked scientist and astronaut Sally Ride to lead a task force to look at potential long-range goals of the U.S. civilian space program. The task force report, “Leadership and America’s Future in Space,” outlined four initiatives, which included both human and robotic exploration of the solar system.

In response to the task force report, the Office of Exploration conducted a series of studies of human and robotic exploration beyond LEO during the 1987–1988 timeframe. These studies ranged in scope and scale as well as utilization of various technology implementations with the direct purpose of providing an understanding of the driving mission, technology, and operational concepts for various exploration missions. In all, four focused case studies were examined, including: Human Expeditions to Phobos, Human Expeditions to Mars, Lunar Observatory, and Lunar Outpost to Early Mars Evolution.

The case studies were deliberately set at the boundaries of various conditions in order to elicit first principles and trends toward the refinement of future options, as well as to define and refine prerequisites. The objective of this approach is to avoid making simple distinctions between exploring the Moon or Mars, but rather, to determine a viable pathway into the solar system.

Recommendations resulting from the 1988 case studies include the following key points:

- Space station is the key to developing the capability to live and work in space.
- Continued emphasis on research and technology will enable a broad spectrum of space missions and strengthen the technology base of the U.S. civilian space program.
- A vigorous life science research base program must be sustained.
- A heavy-lift transportation system must be pursued with a capability that is targeted to transport large quantities of mass to LEO.
- Obtaining data via robotic precursor missions is an essential element of future human exploration efforts.
- An artificial-gravity research program must be initiated in parallel with the zero-gravity countermeasure program if we are to maintain our ability to begin exploration in the first decade of the next century.
- An advanced development/focused test program must be initiated to understand the performance and capability of selected new technologies and systems.

[1]ISRU = in-situ resource utilization

### 1.3.2 Office of Exploration case studies (1989)

Continuing from the 1988 studies, the Office of Exploration continued to lead the NASA-wide effort to provide recommendations and alternatives for a national decision on a focused program of human exploration of the solar system. During 1989, three case studies were formulated for detailed development and analysis: Lunar Evolution, Mars Evolution, and Mars Expedition. In addition, a series of special assessments was conducted. These special assessments focused on areas of “high leverage” that were independent of the case studies and cover a generally broad subject area with potential for significant benefit to all mission approaches. Special assessments included: Power System, Propulsion System, Life Support Systems, Automation and Robotics, Earth-Moon Node Location, Lunar Liquid Oxygen Production, and Launch/On-Orbit Processing.

Results from the 1989 Office of Exploration studies were published in the fiscal year (FY) 1989 Office of Exploration Annual Report. Key conclusions from the 1989 studies include:

> *Mars Trajectories*: Human missions to Mars are characterized by surface stay. Short stay refers to Opposition Class missions, and long-stay pertains to Conjunction Class missions.
>
> *In-space Propulsion*: All-propulsive, all-chemical transportation results in prohibitive total mission mass for Mars missions (1,500–2,000 mt per mission). On the other hand, using aerobraking at Mars can provide significant mass savings (50%) as compared to all-propulsive chemical transportation. Incorporation of advanced propulsion, such as nuclear thermal rockets (NTRs) or nuclear electric propulsion, can result in mission masses that are comparable to chemical/aerobraking missions.
>
> *Reusable Spacecraft*: Employing reusable spacecraft is predominantly driven by economic considerations; however, reusing spacecraft requires facilities that are located in space to store, maintain, and refurbish the vehicles or the vehicles must be designed to be space-based with little/no maintenance.
>
> *In-situ Resources*: The use of in-situ resources reduces the logistical demands on Earth of maintaining a lunar outpost and helps to develop outpost operational autonomy from Earth.
>
> *Space Power*: As the power demands at the lunar outpost increase above the 100 kWe level, nuclear power offers improved specific power.

### 1.3.3 NASA 90-Day Study (1989)

On July 20, 1989, the President announced a major new vision for exploration. In that speech he asked the Vice President to lead the National Space Council in determining what was needed to chart a new and continuing course to the Moon and Mars. To support this endeavor, NASA Administrator Richard Truly created a task force to conduct a 90-day study of the main elements of a human exploration program. Data from this study were to be used by the National Space Council in its deliberations. Five reference approaches were developed, each of which along the lines of the President’s strategy of space station, Moon, then Mars. Regardless of the reference architecture, the study team concluded that heavy-lift launch vehicles (HLLVs), space-based transportation systems, surface vehicles, habitats, and support systems for living and working in deep space are required. Thus, the five reference architectures make extensive use of the space station (*Freedom*) for assembly and checkout operations of reusable transportation vehicles, ISRU (oxygen ($O_2$) from the lunar regolith), and chemical/aerobrake propulsion.

### 1.3.4 The U.S. at the threshold – “The Synthesis Group” (1991)

In addition to the internal NASA assessment of the Space Exploration Initiative (SEI) that was conducted during the NASA 90-Day Study, the Vice President and NASA Administrator chartered an independent group, named the Synthesis Group, to examine potential paths for implementation of the exploration initiative. This group examined a wide range of mission architectures and technology options. In addition, it performed a far-reaching search for innovative ideas and concepts that could be applied to implementing the initiative.

The four candidate architectures that were chosen by the Synthesis Group include: Mars Exploration, Science Emphasis for the Moon and Mars, The Moon to Stay and Mars Exploration, and Space Resource Utilization. Several supporting technologies were identified as key for future exploration, including:

- HLLV (150–250 mt)
- Nuclear thermal propulsion (NTP)
- Nuclear electric surface power
- Extravehicular activity (EVA) suit
- Cryogenic transfer and long-term storage
- Automated rendezvous and docking
- Zero-g countermeasures
- Telerobotics
- Radiation effects and shielding
- Closed-loop life support systems
- Human factors research
- Lightweight structural materials
- Nuclear electric propulsion.
- ISRU

The Synthesis Group also conducted an extensive outreach program with nationwide solicitation for innovative ideas. The directive from the Vice President was to "cast the net widely." Ideas were solicited from universities, professional societies and associations, the American Institute of Aeronautics and Astronautics, the Department of Defense Federal Research Review, Department of Energy, Department of Interior, Aerospace Industries Association, as well as announcements in the *Commerce Business Daily*. Nearly 45,000 information packets were mailed to individuals and organizations that were interested in SEI, which resulted in over 1,500 submissions. "The ideas submitted show innovative but not necessarily revolutionary ideas. The submissions supported a wide range of Space Exploration [Initiative] mission concepts and architectures." (Synthesis Group, 1991) [2]

### 1.3.5 Mars Exploration Design Reference Missions (1994–1999)

During the period from 1994 to 1999, the NASA exploration community conducted a series of studies that was focused on the human and robotic exploration of Mars. Key studies include Mars Design Reference Mission (DRM) 1.0, Mars DRM 3.0 , Mars Combo Lander, and Dual Landers . Each subsequent revision of the design approach provided greater fidelity and insight into the many competing needs and technology options for exploration of Mars. Key mission aspects of each of these studies include the following:

> *Mission Mode*: Each of the Mars mission studies during this period employed Conjunction-class missions, which are often referred to as long-stay missions, to minimize the exposure of the crew to the deep space radiation and zero-gravity environment while also maximizing the scientific return from the mission. This is accomplished by taking advantage of optimum alignment of the Earth and Mars for both the outbound and the return trajectories by varying the stay time on Mars, rather than forcing the mission through nonoptimal trajectories as in the case of the short-stay missions. This approach allows the crew to transfer to and from Mars on relatively fast trajectories, on the order to 6 months, while allowing the crew members to stay on the surface of Mars for a majority of the mission, on the order of 18 months.
>
> *Split Mission*: The surface exploration capability is implemented through a split mission concept in which cargo is transported in manageable units to the surface or Mars orbit, and is checked out in advance of committing the crews to their mission. Emphasis is placed on ensuring that the design of the space transportation systems could be flown in any Mars injection opportunity. This is vital to minimize the programmatic risks associated with funding profiles, technology development, and system design and verification programs.
>
> *Heavy-lift Launch*: HLLVs were used in each of these studies due to the large mission mass for each human mission to Mars (on the order of the ISS at Assembly Complete) as well as due to the large volume payloads that were required.
>
> *Long Surface Stay*: Emphasis was placed on the surface strategy that was associated with each mission approach. Use of Conjunction-class missions provides on the order of 500 days on the surface of Mars for each human mission.

### 1.3.6 Decadal Planning Team/NASA Exploration Team (2000–2001)

In June 1999, the NASA Administrator chartered an internal NASA task force, which was termed the Decadal Planning Team (DPT), to create a new integrated vision and strategy for space exploration. The efforts of the DPT evolved into an agency-wide team that was known as the NASA Exploration Team (NEXT). The DPT was also instructed to identify technology roadmaps that would enable a science-driven exploration vision by establishing a

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[2] Synthesis Group (1991), "America at the Threshold," page A-45.

cross-enterprise, cross-center systems engineering team with a focus on revolutionary, not evolutionary, approaches. The strategy of the DPT and NEXT teams was to “Go Anywhere, Anytime” by conquering key exploration hurdles of space transportation, crew health and safety, human/robotic partnerships, affordable abundant power, and advanced space systems performance. Early emphasis was placed on revolutionary exploration concepts such as rail gun and electromagnetic launchers, propellant depots, retrograde trajectories, nano-structures, and gas core nuclear rockets, to name a few. Many of these revolutionary concepts turned out to be either not feasible for human exploration missions or well beyond expected technology readiness for near-term implementation. During the DPT and NEXT study cycles, several architectures were analyzed, including missions to the Earth-Sun Libration Point (L2), the Earth-Moon Gateway and the Earth-Moon Libration Point (L1), the lunar surface, Mars (both short and long stays), 1-year round trip Mars, and near-Earth asteroids. Common emphasis of these studies included utilization of the Earth-Moon Libration Point (L1) as a staging point for exploration activities, current (shuttle) and near-term launch capabilities (evolved expendable launch vehicle (EELV)), advanced propulsion, and robust space power. Although much emphasis was placed on utilization of existing launch capabilities, the team concluded that missions in near-Earth space are only marginally feasible and human missions to Mars were not feasible without a heavy-lift launch capability. In addition, the team concluded that missions in Earth’s neighborhood, such as to the Moon, can serve as stepping-stones toward further deep-space missions in terms of proving systems, technologies, and operational concepts.

### 1.3.7 Integrated Space Plan (2002–2003)

During the summer of 2002, the NASA Deputy Administrator charted an internal NASA planning group to develop the rationale for exploration beyond LEO. This team, which was termed the Exploration Blueprint, performed architecture analyses to develop roadmaps for how to accomplish the first steps beyond LEO through the human exploration of Mars. The previous NEXT activities laid the foundation and framework for development of NASA’s Integrated Space Plan. The reference missions resulting from the analysis performed by the Exploration Blueprint team formed the basis for requirement definition, systems development, technology roadmapping, and risk assessments for future human exploration beyond LEO. Emphasis was placed on developing recommendations on what could be done now to effect future exploration activities. The Exploration Blueprint team embraced the “stepping-stone” approach to exploration where human and robotic activities are conducted through progressive expansion outward beyond LEO. Results from this study produced a long-term strategy for exploration with near-term implementation plans, program recommendations, and technology investments. Specific results included the development of a common exploration crew vehicle concept (which later would be termed the crew exploration vehicle (CEV)), a unified space nuclear strategy, focused bioastronautics research objectives, and an integrated human and robotic exploration strategy. Recommendations from the Exploration Blueprint included endorsement of the Nuclear Systems Initiative, augmentation of the bioastronautics research, a focused space transportation program including heavy-lift launch, and a common exploration vehicle design for ISS and exploration missions as well as an integrated human and robotic exploration strategy for Mars.

Following the results of the Exploration Blueprint study, the NASA Administrator asked for a recommendation by June 2003 on the next steps in human and robotic exploration to put into context an updated Integrated Space Transportation Plan (post-*Columbia*) and guide agency planning. NASA was on the verge of committing significant funding in programs that would be better served if longer-term goals were better known, including the Orbital Space Plane, research on the ISS, National Aerospace Initiative, Shuttle Life Extension Program, and Project Prometheus as well as a wide range of technology development throughout the agency. Much of the focus during this period was on integrating the results from previous studies into more concrete implementation strategies to understand the relationship among NASA programs, timing, and resulting budgetary implications. This resulted in an integrated approach that included lunar surface operations as a test bed to retire risk of human Mars missions, maximum use of common and modular systems including what was termed the exploration transfer vehicle, Earth orbit and lunar surface demonstrations of long-life systems, collaboration of human and robotic missions to vastly increase mission return, and high-efficiency transportation systems (nuclear) for deep-space transportation and power.

### 1.3.8 Exploration Systems Mission Directorate (2004)

On January 14, 2004, the President announced a Vision for Space Exploration. In his address, the President presented a vision that was bold and forward-thinking, yet practical and responsible – one that explored answers to longstanding questions of importance to science and society and would develop revolutionary technologies and capabilities for the future, while maintaining good stewardship of taxpayer dollars.

NASA's ESMD was created in January of that year to begin implementing the Vision. During 2004, the ESMD Requirements Division conducted a formal requirements formulation process to understand the governing Requirements and systems that would be necessary to implement the Vision. Included were analyses of requirements definition, exploration architectures, system development, technology roadmaps, and risk assessments for advancing the Vision for Space Exploration. This analysis provided an understanding as to what is required for human space exploration beyond LEO. In addition, these analyses helped to identify system "drivers," or significant sources of cost, performance, risk, and schedule variation along with the areas needing technology development. During the early ESMD years, emphasis was placed on definition of initial lunar missions that support long-term exploration endeavors.

### 1.3.9 Exploration Systems Architecture Study (2005)

The NASA ESAS was conducted during between May 2005 and July 2005. The purpose of the study was to:

- Assess the top-level CEV requirements and plans that would enable the CEV to provide crew transport to the ISS, and would accelerate the development of the CEV and crew launch system to reduce the gap between shuttle retirement and CEV initial operational capability (IOC)
- Define the top-level requirements and configurations for crew and cargo launch systems to support the lunar and Mars exploration programs
- Develop a reference exploration architecture concept to support sustained human and robotic lunar exploration operations
- Identify the key technologies that are required to enable and significantly enhance these reference exploration systems, and to perform a reprioritization of near- and far-term technology investments.

To quote the ESAS final report (page 1): "Dr. Michael Griffin was named the new NASA Administrator in April 2005. With concurrence from Congress, he immediately set out to restructure NASA's Exploration Program by making it priority to accelerate the development of the CEV to reduce or eliminate the planned gap in U.S. human access to space. He established a goal for the CEV to begin operation in 2011 and to be capable of ferrying crew and cargo to and from the ISS. To make room for these priorities in the budget, Dr. Griffin decided to down-select to a single CEV contractor as quickly as possible and cancel the planned 2008 subscale test demonstration. He also decided to significantly reduce the planned technology expenditures and focus on existing technology and proven approaches for exploration systems development. In order to reduce the number of required launches and ease the transition after [space shuttle] retirement in 2010, Dr. Griffin also directed the [agency] to carefully examine the cost and benefits of developing a [shuttle] derived [heavy-lift launch vehicle] (HLLV) to be used in lunar and Mars exploration. To determine the best exploration architecture and strategy to implement these many changes, the Exploration Systems Architecture Study (ESAS) team was established at NASA Headquarters."

The ESAS used the Mars studies that were noted previously as the reference approach for assessment of alternative lunar architectures for their intrinsic value as a step towards Mars. Most notable was the emphasis on establishing an HLLV and crew-to-LEO transportation system, which were clear elements of all previous Mars architectures.

# 2 GOALS AND OBJECTIVES

## 2.1 Goals

### 2.1.1 Taxonomy

After extensive discussion, the MAWG concluded that the goals for the initial human exploration of Mars were best organized under the following taxonomy:

- *Goals I-III*: The traditional planetary science goals (from MEPAG, 2006[3]) for understanding Mars Life (Goal I), Climate (Goal II), and Geology/Geophysics (Goal III).
- *Goal IV+*: Preparation for sustained human presence. MEPAG (2006) uses the term "Goal IV" to describe preparation for the first human explorers. By definition, this cannot be a goal for the first human missions; by then the preparation would have to have been complete. However, a goal of the first human missions is to prepare for the subsequent future after that.
- *Goal V (Ancillary Science)*: This includes all scientific objectives that are unrelated to Mars, including those that are related to astrophysics, observations of the Sun, Earth, Moon, and the interplanetary environment. Note that these objectives may be important during the transit phase for missions to and from Mars.

Analysis of Goals I through III was prepared by an analysis team that was sponsored by the MEPAG, which went by the name of HEM-SAG. HEM-SAG produced a substantial white paper (MEPAG HEM-SAG, 2008[4]), and section 2.2 of this report contains a summary of that more detailed analysis.

The scientific objectives for the initial first three human missions to Mars are summarized in table 2-1.

**Table 2-1.** Summary of Objectives for the Initial Program of Human Missions to Mars

| Goals I–III | Goals IV+ | Goal V |
|---|---|---|
| Quantitatively characterize the different components of the *martian geologic system* (at different times in martian geologic history), and understand how these components relate to each other (in three dimensions). | Learn to make effective use of *martian resources*, including providing for crew needs and, if possible, power and propulsion consumables. | Ancillary science (heliophysics, astrophysics) |
| Search for *ancient life* on Mars. | Develop reliable and robust *exploration systems*; increase the level of self-sufficiency of Mars operations. | |
| Make significant progress towards the goal of understanding whether or not *martian life* forms have persisted to the present (extant biological processes). | Address *planetary protection* concerns regarding sustained presence. | |
| Quantitatively understand early Mars *habitability* and early Mars possible *pre-biotic biogeochemical cycles and chemistry*. | Promote the development of *partnerships* (international, commerical, etc.) and sustain *public engagement*. | |
| Characterize the structure, composition, dynamics, and evolution of the *martian interior* (core to crust). | | |
| Quantitatively understand *martian climate history* with attention to the modern climate/ weather system. | | |

**Notes:** 1. Not listed in priority order. 2. For Goal V, it was not possible to be specific.

[3]MEPAG (2006), Mars Scientific Goals, Objectives, Investigations, and Priorities: 2006, J. Grant, ed., 31 pp. white paper posted February 2006 by the Mars Exploration Program Analysis Group (MEPAG) at http://mepag.jpl.nasa.gov/reports/index.html.

[4]MEPAG HEM-SAG (2008). Planning for the Scientific Exploration of Mars by Humans. Unpublished white paper (J. B. Garvin and J. S. Levine, Editors) posted March 2008 by the MEPAG Human Exploration of Mars-Science Analysis Group (HEM-SAG) at http://mepag.jpl.nasa.gpv/reports/index.html.

## *2.2 Mars Planetary Science Objectives (Goals I–III)*

### 2.2.1 Introduction

Mars is a diverse and complex world. Many of the same processes/mechanisms operate, or have operated, on both Earth and Mars; e.g., early heavy bombardment, impact craters, planetary dipole magnetic field (at least in the early history of Mars), widespread and extensive volcanism, the presence of liquid water on the surface, geochemical cycles, the condensation of atmospheric gases forming polar caps, etc. Mars, like Earth, is a terrestrial planet with very diverse and complex geological features and processes. Again like the Earth, Mars is also a possible abode for past and/or present life. The geological record suggests that the atmosphere/climate of Mars has changed significantly over its history. Early Mars may have possessed a significantly denser atmosphere that was lost (Jakosky and Phillips, 2001[5]). A denser atmosphere on Mars would have permitted liquid water on its surface. Present-day Mars has a thin (6 millibars) cold atmosphere that is devoid of any surface liquid water. Why has Mars changed so drastically over its history? How and why has the habitability of Mars changed over its history? Is there a message in the history of Mars to better understand the future of the Earth? Did life form on early Mars? Is there evidence of early life in the geological record? Is there life on Mars today?

### 2.2.2 The unique attributes of humans in scientific exploration

It is important to consider the unique capabilities that humans bring to the exploration of Mars. In so doing, a common set of human traits emerges that applies to exploration relating to the MEPAG science disciplines of Geology, Geophysics, Life, and Climate. These characteristics include: speed and efficiency to optimize field work; agility and dexterity to go places that are difficult for robotic access and to exceed currently limited degrees-of-freedom robotic manipulation capabilities; and, most importantly, the innate intelligence, ingenuity, and adaptability to evaluate in real time and improvise to overcome surprises while ensuring that the correct sampling strategy is in place to acquire the appropriate sample set. Real-time evaluation and adaptability especially would be a significant new tool that humans on Mars would bring to surface exploration. There are limitations to the autonomous operations that are possible with current robotic systems; fundamental limitations to direct commanding from Earth are the time difference that is imposed by the 6- to 20-minute communications transit time and the small number of daily uplink and downlink communications passes.

Humans are unique scientific explorers. We can obtain previously unobtainable scientific measurements on the surface of Mars. Further, we possess the abilities to adapt to new and unexpected situations in new and strange environments; human explorers can make real-time decisions, and have strong recognition abilities and are intelligent. Humans can perform detailed and precise measurements of the surface, subsurface, and atmosphere while on the surface of Mars with state-of-the-art scientific equipment and instrumentation brought from Earth. The scientific exploration of Mars by humans would presumably be performed as a synergistic partnership between humans and robotic probes that are controlled by the human explorers on the surface of Mars (MEPAG HEM-SAG, 2008).

Robotic probes can explore terrains and features that are not suitable or too risky for human exploration. Under human real-time control, robotic probes can traverse great distances from the human habitat, covering distances/terrains that are too risky for human exploration; undertake sensitive, delicate sample handling operations; and return rock and dust samples to the habitat for triage and laboratory analyses.

### 2.2.3 Scientific objectives for Mars in the future

Our current scientific objectives for the exploration of Mars have been described in detail by MEPAG (2006), and a high-level summary is shown in figure 2-1.

[5]Jakosky, B. M. and R. J. Phillips, 2001: Mars' Volatile and Climate History, *Nature* **142**, 237.

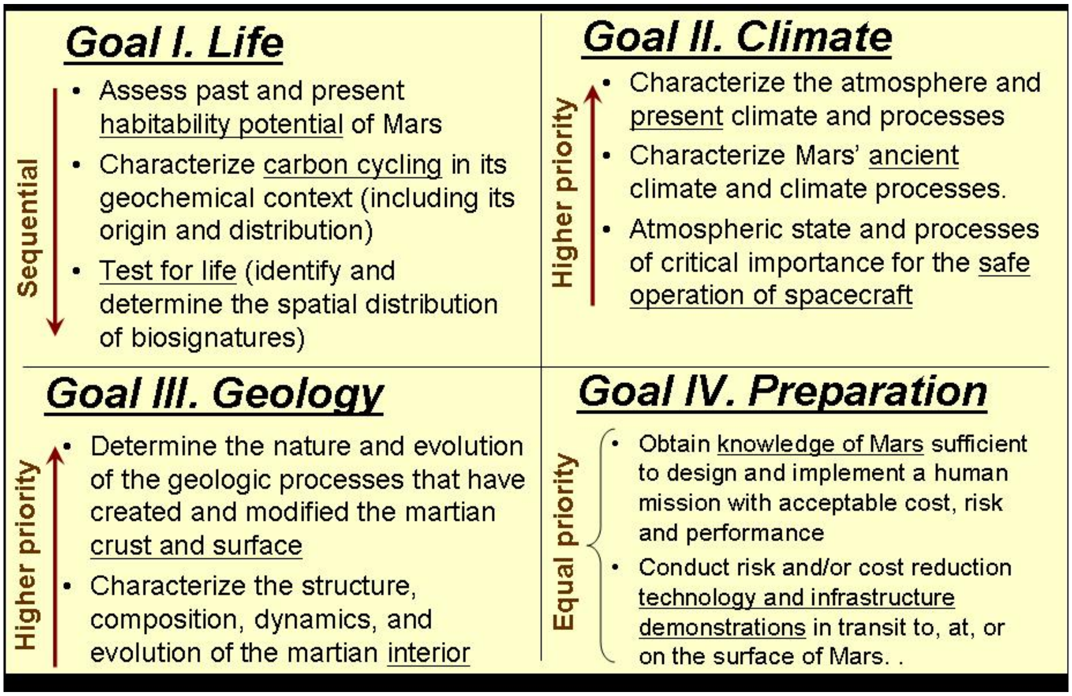


**Figure 2-1.** Scientific objectives for the exploration of Mars.

In planning the scientific objectives of a mission 20 to 25 years from now, we also need to take into account the additional robotic missions that are likely to be scheduled before the first human mission, and the progress that they will make towards these objectives. We need to plan the objectives of a 2030 mission based on our projected state of knowledge as of about 2025, not based on our objectives as of 2008. For the purpose of this planning exercise, between now and 2025 the following missions are assumed to have achieved their objectives: Mars Phoenix, Mars Science Laboratory (MSL) (scheduled for launch in 2009), the Mars Scout Aeronomy Orbiter (scheduled for launch in 2013), ExoMars (scheduled for launch in 2013), at least one additional science orbiter, and Mars Sample Return (MSR). Although other science missions will certainly be considered (most importantly, a network science mission), for the purpose of this planning we have not pre-judged NASA's decision-making, and have only assumed the missions that seem most probable.

The results of the robotic missions between now and 2025 will answer some of the questions on our current marquee, questions that would therefore be removed and be replaced by new questions; this is the way in which scientific investigations always work. Although our ability to predict the results of these future missions, and the kinds of new questions that will come up, is partial, we do know the kinds of data that will be collected and the kinds of questions that data are capable of answering. Thus, we can make some general projections of the state of knowledge as of 2025.

**Goal I. DETERMINE WHETHER LIFE EVER AROSE ON MARS**

By 2025, our assessments of habitability potential will be well advanced for some environments, particularly those that have been visited by the MSR or by major in-situ rovers with life-related experiments. However, it is likely that the habitability of the martian subsurface will be almost completely unexplored other than by geophysical methods. The objective relating to carbon cycling is likely to be partially complete, but in particular as related to subsurface environments. For the purpose of this planning, we assume that the investigations through 2025 have made one or more discoveries that are hypothesized as being related to ancient life (by analogy with the Allen Hills meteorite story, this is a particularly likely outcome of MSR). We should then be prepared for the following new objectives:

- Characterize the full suite of biosignatures for ancient life to confirm the past presence of life. Interpret its life processes and the origin of such life
- Assess protected environmental niches that may serve as refugia for extant life forms that may have survived to the present. Find the life, measure its life processes

- In earliest martian rocks, characterize the pre-biotic chemistry

**Goal II.** **UNDERSTANDING THE PROCESSES AND HISTORY OF CLIMATE ON MARS**
By 2025, our objectives related to characterization of the Mars atmosphere and its present and ancient climate processes are likely to be partially complete. In addition to continuing long-term observations, our scientific questions seem likely to evolve in the following directions. Note in particular that if there is no robotic mission to one of the polar caps, the priority of that science is likely to be significantly more important than it is today because of the influence of polar ice on the climate system.

- Quantitative understanding of global atmospheric dynamics
- Understand microclimates – range of variation, how and why they exist
- Perform weather prediction
- Understand the large-scale evolution of the polar caps including the modern energy balance, links with dust, carbon dioxide ($CO_2$), and water ($H_2O$) cycles, changes in deposition and erosion patterns, flow, melting, age, and links between the two caps

**Goal III.** **DETERMINE THE EVOLUTION OF THE SURFACE AND INTERIOR OF MARS**
As of 2006, there were two primary objectives within this goal: (1) Determine the nature and evolution of the geologic processes that have created and modified the martian crust and surface, and (2) characterize the structure, composition, dynamics, and evolution of the martian interior. These are broadly enough phrased that they are likely to still be valid in 2025. These two objectives, for example, currently apply to the study of the Earth, even after more than 200 years of geologic study by thousands of geologists. Given the anticipated robotic missions leading up to the first human missions, the first objective is likely to evolve in the following direction:

- Quantitatively characterize the different components of the martian geologic system (at different parts of martian geologic history), and understand how these components relate to each other
- Understand the field context of the various martian features of geologic interest at both regional and local scale
- Test specific hypotheses
- Perform comparative planetology

Unless a robotic geophysical network mission is scheduled before the first human mission, our progress on the second objective will be minimal, and this will remain one of most important open questions.

### 2.2.4 Significance of the variation in martian geology in space and time

Some of the most important questions about all three of the above-mentioned goals involve the relationship of $H_2O$ to martian geologic and/or biologic processes as a function of geologic time. Mars has apparently evolved from a potentially “warm and wet” period in its early Noachian history to the later “cold and dry” period of the Amazonian period (figure 2-2). Since rocks of different age are exposed in different places on Mars, understanding this geologic history requires an exploration program that also involves spatial diversity. As one illustration of this point, the MEPAG HEM-SAG team compiled a map showing the sites of high exploration interest as of 2007 (figure 2-3); they are scattered across the surface of the planet. One of the realities of geology-related exploration is that samples and outcrops are typically representative only of a certain geologic environment, and acquiring information about other environments requires going to a different place. (A terrestrial analog would be asking: How much we could learn about Precambrian granite by doing field work in the sedimentary rocks of the Great Plains?)

Given that the engineering of missions to Mars are constrained to be either “short stay” or “long stay” (section 3.3), and assuming that the initial human exploration of Mars consists of a program of three missions, a key tradeoff is the mission duration and whether the missions are sent to the same or different sites. From the perspective of scientific goals, it is clear that progress will be optimized by visiting multiple sites, and maximizing the stay time at those sites. The same argument regarding diversity of sites was raised, and followed, during the Apollo Program. The longer stay time is needed because the geology of Mars, at many sites, has complexities that will take a significant amount of time to resolve. If we are to bring the unique attributes of human explorers to bear, we would need to give them enough time on the outcrops. The essence of this key trade is summarized, from the point of view of our scientific objectives, in figure 2-3. (**Note:** This map is for illustrative purposes and is not exclusive. The three sites in red – Site 1: Nili Fossae (Jezero Crater), Site 26: Arsia Mons, and Site 38: Mangala Valles – each corresponds to a site of great geological interest representing the three different geological periods of Mars: The Noachian (Jezero Crater), the Hesperian (Mangala Valles), and the Amazonian (Arsia Mons). Each of these sites was selected for human science reference missions (HSRMs) to illustrate the great geological diversity of the surface of Mars).

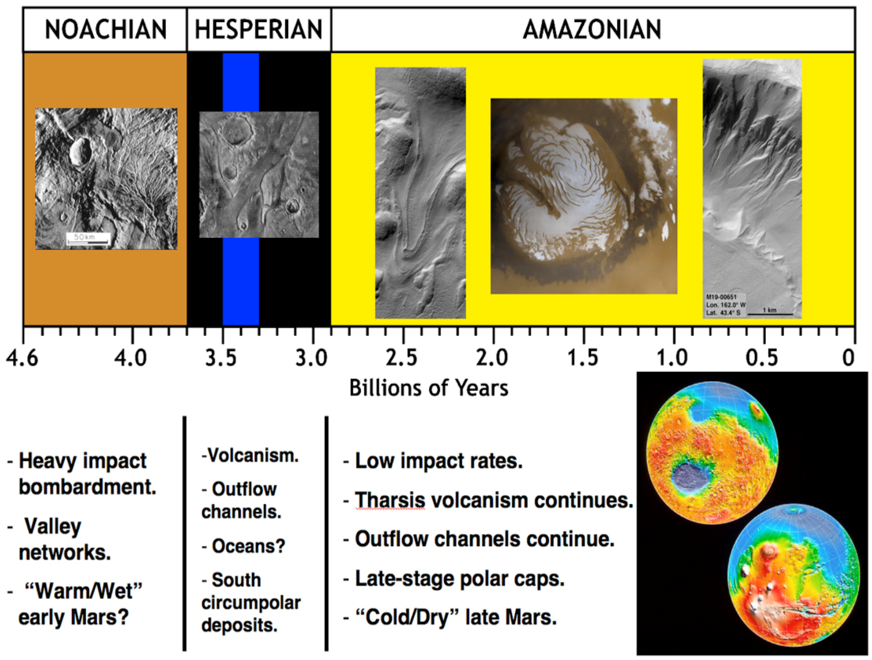


**Figure 2-2.** Geological history of Mars with the major periods and significant events.

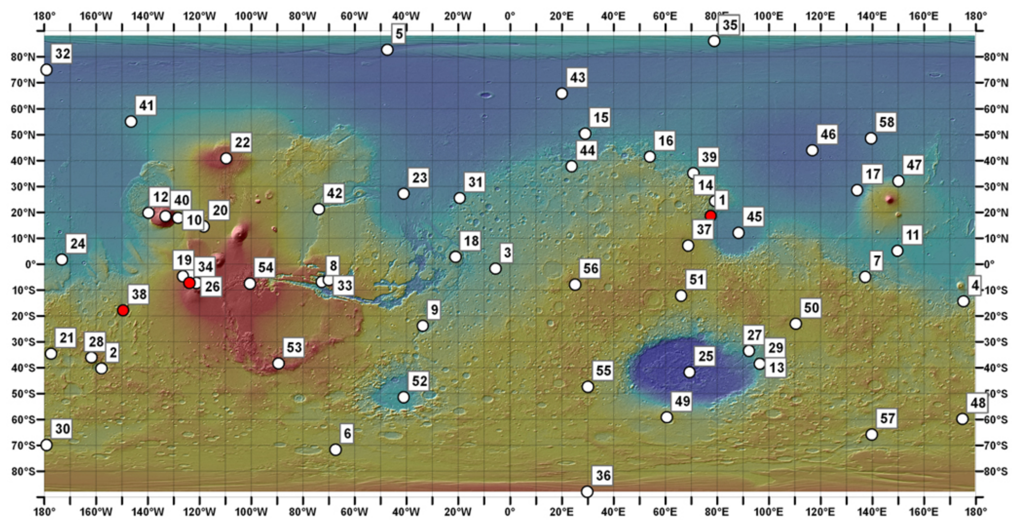


**Figure 2-3.** Map of 58 potential exploration sites on Mars (illustrative purposes only).

### 2.2.5 Introduction to human science reference missions

There is substantial diversity in both geology and topography among the various landing sites of possible interest. This leads to significant differences in the way we would think about the scientific objectives for different places.

To better illustrate the issues, we have found it useful to formulate a spectrum of what we call HSRMs; for additional detail, see MEPAG HEM-SAG (2008). HSRMs allow the opportunity to be more specific at each site on essential parameters such as mobility range, number and kind of objectives, operations times, etc. These HRSMs are not presented as specific recommendations; they are case histories that are designed to illustrate underlying relationships. To illustrate the value and importance of human scientific exploration, HEM-SAG has developed HRSMs for each of the four scientific disciplines considered; i.e., geology, geophysics, atmosphere/climate and biology/life. From the 58 sites that are shown in figure 2-3, we chose several sample for more detailed examination and traverse selection, which we will use as reference missions after further outlining their geological significance and relevance to major science questions.

#### *2.2.5.1 Human science reference missions: geology*

**Interpreting planetary-scale geologic processes using Human Exploration**

The absolute ages of surface units on Mars has been deciphered through indirect methods; samples returned from the Moon in the Apollo Program were used to provide constraints on the crater-size frequency distribution of the lunar surface (Gault, 1970[6]; Hartman, 1972[7]), and this has been applied to Mars, among other terrestrial planetary bodies (Barlow, 1988[8]; Strom, 1992[9]; Neukum, 2001[10]). While this has provided a general history of martian surface processes (figure 2-2), it does not allow for detailed study of specific martian periods, in particular the Hesperian and Amazonian when the impact flux greatly decreased. While martian meteorites have been analyzed and dated (Nyquist et al., 2001[11]), not knowing their geologic context makes their incorporation into the geologic history of Mars difficult. While an MSR mission would potentially yield surface samples with known context, a robotic mission would not yield the array of optimal samples that would address a wide range of fundamental questions. A human mission might allow for greater access to samples that a robotic rover might not get to, and the capacity for real-time analysis and decision-making would ensure that the samples that were obtained would be the optimal samples that are available.

Human explorers would also have greater access to the near-subsurface of Mars, which would yield insights into climate and surface evolution, geophysics, and, potentially, life. Humans would be able to navigate more effectively through blocky ejecta deposits that would provide samples that were excavated from great depth and provide a window into the deeper subsurface. Humans could also trench in dozens of targeted locations and operate sophisticated drilling equipment that could drill to a depth of 500 to 1,000 meters below the surface (The drilling depth range of 500 to 1,000 meters below the surface represents the HEM-SAG team consensus depth, where it is believed that subsurface water may be found. Clearly, additional investigation will be needed to narrow the depth of drilling). Our current understanding of the crust of Mars is limited to the top meter of the surface, so drilling experiments would yield unprecedented and immediate data. Drilling in areas of gully formation could also test the groundwater model by searching for a confined aquifer at depth.

We have analyzed three different exploration sites in detail as reference missions for the first program of human Mars exploration. The sites span the geologic history of Mars (one site for each period of martian history) and allow for exploration traverses that would examine a variety of surface morphologies, textures, and mineralogies to address the fundamental questions that were posed by MEPAG.

**Jezero Crater**

Jezero crater is a, approximately 45-km impact crater that is on the northwest margin of the Isidis impact basin in the Nili Fossae region of Mars (figure 2-4). This region is a very important area for understanding the formation of the Isidis basin, the alteration and erosion of this Noachian basement, and subsequent volcanism and modification

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[6] Gault, D.E. (1970) Saturation and equilibrium conditions for impact cratering on the Lunar surface: Criteria and implications. *Radio Science* **5**, 273-291.
[7] Hartmann, W.K. (1972) Paleocratering of the Moon: Review of post-Apollo data. *Astrophysics Space Science* **17**, 48-64.
[8] Barlow, N.G. (1988) Crater size/frequency distributions and a revised relative Martian chronology. *Icarus*, 74, 285-305.
[9] Strom,R.G., Croft, S.K., Barlow, N.G. (1992) The Martian imact cratering record. In *Mars* (H.H.Kieffer, B.M.Jakosky, C.W.Snyder, and M.S.Matthews eds.), University of Arizona Press, 384-423.
[10] Neukum, G., Ivanov, B.A., Hartmann, W.K. (2001) Cratering records in the inner solar systemin relation to the Lunar reference system. *Space Science Reviews* **96**, 1-4, 55-86.
[11] Nyquist, L.E., Bogard, D.D., Shih, C.-Y., Greshake, A, Stoffler, D., Eugster, O. (2001) Ages and geologic histories of Martian meteorites. *Space Science Reviews* **96**, 1-4,105-164.

(Mangold et al., 2007[12]; Mustard et al., 2007[13]). The rim has been breached in three places: twice where channels from the neighboring highlands to the west drained into the crater from the northwest, and once on the eastern margin where the crater drained eastward towards the Isidis basin. Each input channel deposited deltas on the crater floor that have been preserved to reveal sedimentary structures and clay deposits in high-resolution images and spectral data. Other parts of the crater floor appear to have been resurfaced by lava.

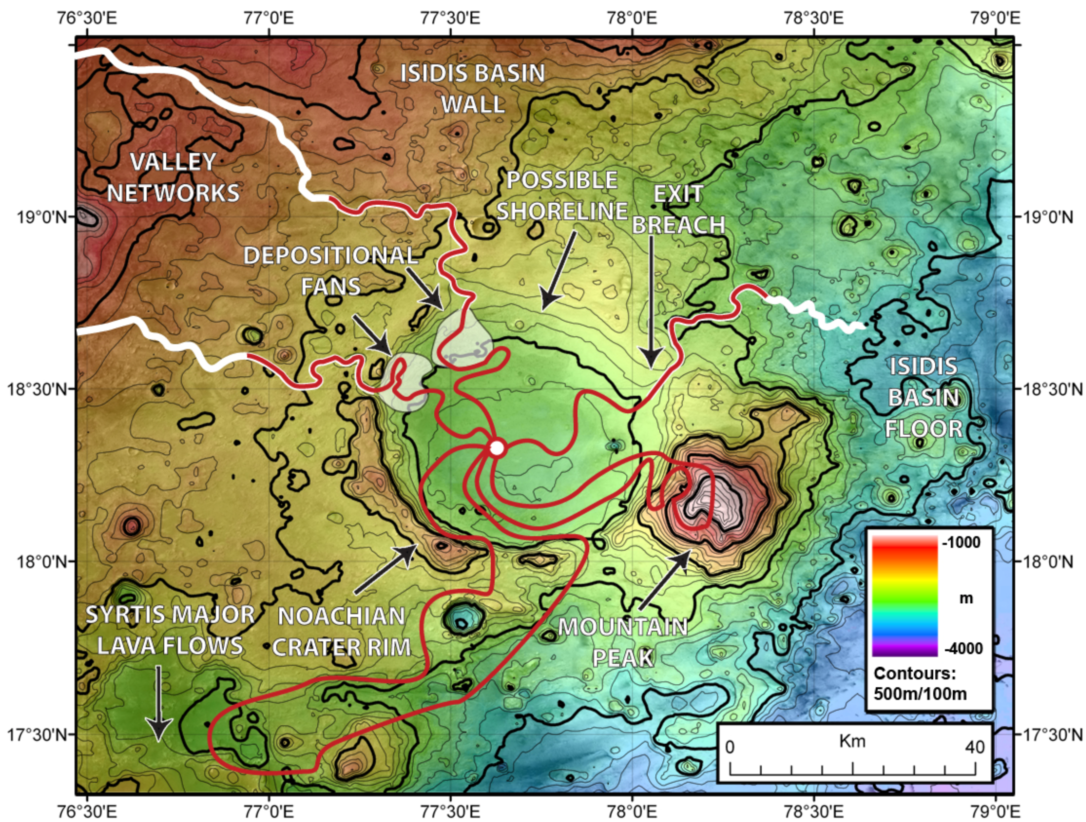


**Figure 2-4.** Potential traverses for human explorers in and around Jezero Crater.

A 500-day mission at this site would give considerable insight into the early martian environment. Jezero Crater itself is Noachian in age, and the preserved rim would provide access to ancient bedrock material that is rich in low-calcium pyroxene and that has been exposed by the impact (Mustard et al., 2007). The delta deposits are likely to be Noachian in age; HiRISE data show that the sedimentary record in the deposit has been preserved as a series of thin layers (Fassett et al., 2007[14]). On the basis of the fact that a standing body of water existed within the crater for an extended period of time, this would be an ideal site to search for extinct life. Humans would also be able to examine the structure and deposits within the channels that are associated with the deltas, which would be applicable to the other vast valley networks on Mars.

Extended traverses would be able to access and study the entire Jezero Crater system. To the southwest of Jezero Crater are Hesperian lava flows from Syrtis Major, which is one of the main volcanoes in the northern hemisphere of

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[12]Mangold, N., Poulet, F., Mustard, J. F., Bibring, J.-P., Gondet, B., Langevin, Y., Ansan, V., Masson, Ph., Fassett, C., Head, J. W., Hoffmann, H., Neukum, G. (2007) Mineralogy of the Nili Fossae region with OMEGA/Mars Express data: 2. Aqueous alteration of the crust.. *Journal of Geophysical Research* **112**, E8, doi: 10.1029/2006JE002835.
[13]Mustard, J. F., Poulet, F., Head, J. W., Mangold, N., Bibring, J.-P., Pelkey, S. M., Fassett, C. I., Langevin, Y., Neukum, G. (2007) Mineralogy of the Nili Fossae region with OMEGA/Mars Express data: 1. Ancient impact melt in the Isidis Basin and implications for the transition from the Noachian to Hesperian. *Journal of Geophysical Research* **112**, E8, doi: 10.1029/2006JE002834.
[14]Fassett C. I., Ehlmann, B. L., Head, J. W., Mustard, J. F., Schon, S. C., Murchie, S. L. (2007) Sedimentary Fan Deposits in Jezero Crater Lake, in the Nili Fossae Region, Mars: Meter- scale Layering and Phyllosilicate-Bearing Sediments. *American Geophysical Union* (abs) P13D-1562.

Mars, that provide a key constraint on the geological timescale of the region. This would also shed light on the evolution of magma compositions on Mars. To the east of Jezero Crater is the floor of Isidis basin, which is topographically connected to the northern plains and allows for detailed study of major impact events. Samples collected from all of these sites would allow for enhanced geochronology and a more detailed understanding of the hydrology, sedimetology, volcanology, and habitablity of the region.

**Mangala Valles**
Mangala Valles is a Hesperian-aged outflow channel that has received considerable attention because of its role in global cryosphere/hydrosphere interactions, as well as the possibility that it contains icy near-surface deposits (figure 2-5) (Zimbleman et al., 1992[15]; Ghatan et al., 2005[16]; Levy and Head, 2005[17]; Basilevsky et al., 2007[18]). Mangala Valles emanates from a graben that is radial to the Tharsis volcanic complex. Massive release of water from the ground at the graben was accompanied by phreatomagmatic eruptions, causing catastrophic flow of water to the north and carving streamlined islands. There are also young glacial deposits along the rim of the graben and evidence for glacial scour having modified the surface of the outflow channel.

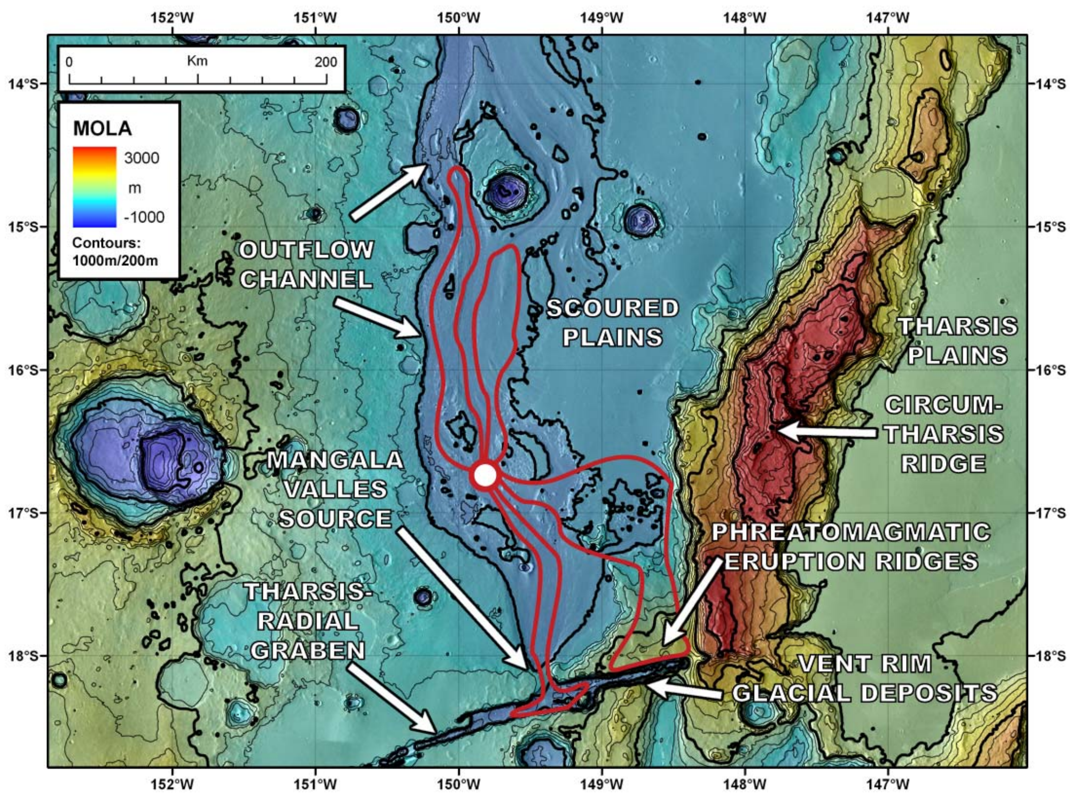


**Figure 2-5.** Potential traverses for human explorers in and around Mangala Valles.

[15]Zimbelman J.R., Craddock, R. A., Greeley, R., Kuzmin, R. O. (1992) Volatile history of Mangala Valles Mars. *Journal of Geophysical Research* **97**, E11, 18,309-18,317.
[16]Ghatan G. J., Head, J. W., Wilson, L. (2005) Mangala Valles, Mars: Assessment of Early Stages of Flooding and Downstream Flood Evolution. *Earth Moon and Planets*, **96**, 1-2, 1-5.
[17]Levy, J.S., and Head, J.W. (2005) Evidence for remnants of ancient ice-rich deposits: Mangala Valles outflow channel, Mars. *Terra Nova* **17**, 6, 503-510.
[18]Basilevsky A. T., Neukum, G., Werner, S. C., van Gasselt, S., Dumke, A., Zuschneid, W., Chapman, M., Greeley, R. (2007) Geological Evolution of Mangala Valles, Mars: Analysis of the HRSC Image H0286. *Lunar and Planetary Science XXXVIII*, Abstract No. 1338.

This site shows evidence for fluvial, volcanic, tectonic, and glacial activity as well as complicated interactions among them. A landing site in the smooth terrain at the center of the outflow channel would provide access to a variety of sites of interest. Traverses to the channel head and the graben would allow direct observation of cryosphere-breaching geological activity. Traverses along the floor of the outflow channel and on the scoured plains would provide insight into outflow flood hydrology and erosion processes, and might also provide an opportunity for sampling ice-rich deposits that may contain ancient flood residue. A traverse to the vent-rim glacial deposits would provide access to landforms that were created by volcano-ice interactions as well as to samples of distal Tharsis volcanic deposits. If life exists on Mars, it is most likely to inhabit the subsurface, and a site such as Mangala offers a unique opportunity to sample for evidence of such activity.

**Arsia Mons Graben**

All three of the major Tharsis Montes shield volcanoes and Olympus Mons exhibit expansive late-Amazonian glacial deposits on their northwestern flanks (figure 2-6). The broadest of these are found on Arsia Mons, which shows glacial deposits that are approximately 400 km to the west of the accumulation zone and covers an area of about 170,000 km$^3$ (Shean et al., 2006[19]). These glacial deposits are found among classic volcanic and tectonic structures, so an extended mission at this location would provide a wealth of information concerning several of the fundamental questions of martian geology during the Amazonian period.

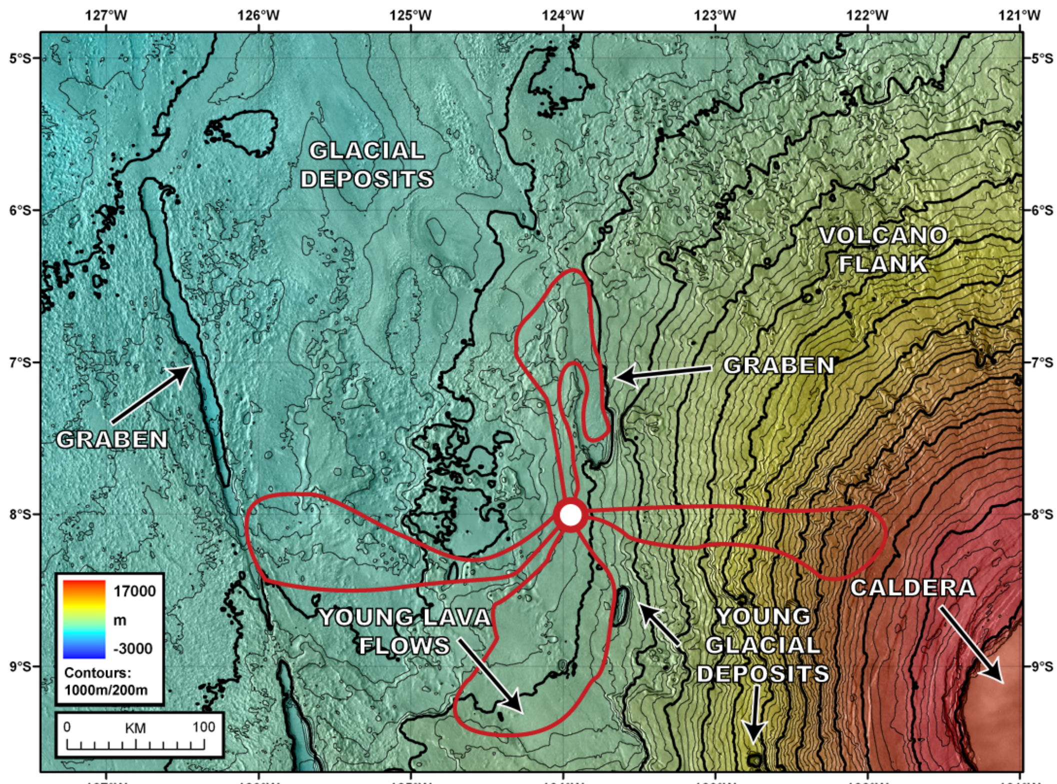


**Figure 2-6.** Potential traverses for human explorers in and around the Arsia Mons glacial deposits.

We envisioned several traverses from a potential base camp set up at 8°S, 124°W that would analyze glacial and volcanic deposits as well as the complicated relationship between them. By using extended rovers, human explorers would be able to ascend the western flank of the shield and systematically obtain targeted samples that elucidate the recent volcanic history of Arsia. Another traverse from the same base camp would provide access to an approximately 5-km-wide graben that appears to have been a major accumulation zone for much of the observed glacial deposits.

[19]Shean D. E., Head, J. W., Fastook, J. L., Marchant, D. R. (2006) Recent glaciation at high elevations on Arsia Mons, Mars: Implications for the formation and evolution of large tropical mountain glaciers. *Journal of Geophysical Research* 112, E3, doi: 10.1029/2006JE002761.

A systematic sampling strategy at this location would provide a history of the flow regime at this site, and drilling at targeted locations can provide the recent climate record for Mars.

A list of geologic processes that have created and modified the martian crust and surface, in priority, is summarized in table 2-2.

**Table 2-2.** List of Geologic Processes That Have Created and Modified the Martian Crust and Surface

*Objective: Determine the nature and evolution of the geologic processes that have created and modified the martian crust and surface (investigations in priority order).*

| Investigation | Geology Approaches |
|---|---|
| 1. Determine the present state, three-dimensional distribution, and cycling of water on Mars. | – Drilling<br>– Surveying for groundwater seeps<br>– Ground-penetrating radar (GPR)<br>– meteorology (MET) stations |
| 2. Evaluate fluvial, subaqueous, pyroclastic, subaerial, and other sedimentary processes and their evolution and distribution through time, up to and including the present. | – Sampling along traverses<br>– MET stations<br>– In-situ composition analysis |
| 3. Calibrate the cratering record and absolute ages for Mars. | – Thorough sampling of diverse rocks<br>– Cosmogenic age dating of samples |
| 4. Evaluate igneous processes and their evolution through time, including the present. | – Extensive sampling traverses<br>– In-situ composition analysis |
| 5. Characterize surface-atmosphere interactions on Mars, including polar, Aeolian, chemical, weathering, mass-wasting, and other processes. | – MET stations<br>– Traverse sampling along glaciers<br>–Sampling of diverse mineralogy |
| 6. Determine the large-scale vertical structure and chemical and mineralogical composition of the crust and its regional variations; this includes, for example, the structure and origin of hemispheric dichotomy. | – Drilling<br>–Seismic stations<br>– GPR<br>– Compositional comparison of fresh/weathered samples |
| 7. Document the tectonic history of the martian crust, including present activity. | – Seismic stations<br>– Observations of graben and other tectonic features |
| 8. Evaluate the distribution and intensity of hydrothermal processes through time, up to and including the present. | – Stratigraphic sample collection<br>– Compositional analysis at multiple sites |
| 9. Determine the processes of regolith formation and subsequent modification, including weathering and diagenetic processes. | – Sample collection at multiple latitudes/ environments<br>– MET stations |
| 10. Determine the nature of crustal magnetization and its origin. | – In-situ magnetometer analysis<br>– Traverses in areas of magnetic anomalies |
| 11. Evaluate the effect of impacts on the evolution of the martian crust. | – Ejecta sampling<br>– Mapping of crater-wall outcrops |

#### *2.2.5.2 Human science reference missions: geophysics*

Mars geophysics science objectives fall into two broad categories: planetary-scale geophysics (1000's of km) and what might be called "exploration geophysics," which addresses regional (10's–100's km) or local scales (<10 km). The first involves characterizing the structure, composition, dynamics, and evolution of the martian interior, while the second addresses the structure, composition, and state of the crust, cryosphere, hydrologic systems, and upper mantle. Here we describe how these objectives might be met through investigations that are carried out on human missions.

We assume here that no robotic missions to Mars before 2025 will address the science issues in a complete way. For example, we assume that no network mission such as $ML_3N$ (National Research Council (NRC), 2006[20]) will be flown. We do this to be conservative, to make as complete a set of human exploration-related geoscience

[20]National Research Council (2006) Assessment of NASA's Mars Architecture 2007-2016, Committee to Review the Next Decade Mars Architecture, Space Studies Board, Division on Engineering and Physical Sciences. The National Academies Press, Washington, D.C.

investigations and activities as possible. Clearly if future robotic missions address Mars geophysics topics, the human mission activities must be reconsidered.

In general, Mars geophysics will be well served by landing sites and traverses that were identified by the Geology panel. Figure 2-3 shows the 58 sites that were considered. (For additional detail, see MEPAG HEM-SAG, 2008.) These sites span the planet, and offer a sampling of Mars' remarkable geologic diversity. The Chasma Boreale site (CB, Site 5) offers access to an immense stratigraphic column of polar layered deposits that presumably stretch far back into the Amazonian (Tanaka, 2005[21]) (This site is at an elevation of several kilometers.) The Nili Fossae site (NF, Site 1) sits on the edge of the hemispheric dichotomy boundary and provides access to Noachian/Hesperian-age fluvial features (Tanaka, 2001[22]). The Centauri Montes site (CM, Site 29), which is on the eastern rim of the giant Hellas impact basin, contains features that range from Noachian basin rim materials to Amazonian/Hesperian outflow channels to Amazonian debris aprons and recent gully changes, hinting at the possibility of near-surface water (Malin et al., 2006[23]). The Arsia Mons site (AM, Site 26) sits on the western flank of the volcano and provides access to putative Amazonian-age glacial deposits and comparatively young lava flows. Each site offers the opportunity to address multiple geophysics investigations. We will revisit these sites and plausible geophysical exploration strategies later.

**Planetary-scale geophysics: structure, composition, dynamics, and evolution of the martian interior**
To characterize the structure and dynamics of the martian interior and determine the chemical and thermal evolution of the planet, physical quantities such as density and temperature with depth, composition and phase changes within the mantle, core/mantle boundary location, thermal conductivity profile, and the three-dimensional mass distribution of the planet must be determined. To determine the origin and history of the magnetic field of the planet, we must discover the mineralogy that is responsible for today's observed remnant magnetization, and understand how and *when* the rocks bearing these minerals were emplaced.

The measurement requirements for planetary-scale geophysics present some drivers for Mars exploration architectures. A key driver is the need to instrument the planet at appropriate scales. For example, global seismic studies rely on widely separated stations so that seismic ray paths passing through the deep mantle and core can be observed. This need translates into multiple, widely separated landing sites for the first human missions. If only a single landing site is selected and revisited, far less information about the interior of the planet will be obtained. As can be seen in figure 2-3, the three low-latitude sites would provide a reasonable planetary-scale network, and would also enable heat flow measurements in diverse crustal/lithospheric settings: the volcanic Tharsis rise, the Isidis wall/dichotomy boundary, and the rim of the Hellas basin.

To characterize the structure, composition, and state of the martian near-surface crust, both local and regional subsurface information must be obtained. A wide variety of exploration geophysics techniques exist that provides such information. For example, sounding for aquifers can be accomplished through electromagnetic techniques, and layering in sedimentary units can be determined through reflection seismology. Magnetic surveys that are carried out at landing sites tell us about the spatial scales of crustal magnetization, and tie in to local and regional geology for context.

Geophysics measurement requirements span three disparate spatial scales, depending on the science that is to be done. At the largest scales (1000's of km), characterizing the interior of Mars requires a widely spaced network of at least three emplaced central geophysics stations, one at each landing site. At regional scales (10's–100's km), characterizing crustal structure, magnetism, and other objectives requires mobility to emplace local networks around a landing site. Finally, at local scales (~10 km), mobility is key to performing traverse geophysics, and in carrying out investigations (such as seismic or electromagnetic (EM) sounding) at specific stations along a traverse. The central geophysics stations and the regional scale networks would be emplaced and left to operate autonomously after the human crew departs. Traverse and station geophysics would be carried out only during the human mission, unless this could be done robotically after completion of the human mission.

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[21]Tanaka, K. L.(2005) Geology and insolation-driven climatic history of Amazonian north polar materials on Mars. *Nature* ***437***, 991-994.
[22]Tanaka, et al. (2001) Catastrophic erosion of Hellas basin rim on Mars induced by magmatic intrusion into volatile-rich rocks. *Geophysical. Research. Letters.* **29,** doi:10.1029/2001GL013885.
[23]Malin, M.C., Edgett, K.S., Posiolova, L.V., McColley, S.M., Noe Dobrea, E.Z. (2006) Present-Day Impact Cratering Rate and Contemporary Gully Activity on Mars. *Science* **314**, 5805, 1573 – 1577.

Central geophysical stations at each landing site would include passive broadband seismic, heat flow, precision geodesy, and passive low-frequency EM instrumentation. Satellite geophysics stations would include the nodes of a regional seismic array and vector magnetometers. Along the traverses, experiments would be performed at sites of interest. These would include active EM sounding for subsurface aquifers, active seismic profiling to establish structure with depth, and gravity measurements. GPR and neutron spectroscopy along the traverse track would help to map out subsurface structure and hydration state/ice content for the near-subsurface.

Based on the geophysics science objectives, multiple sites meet the investigation needs and geologic settings. Three sites, which are found at widely separated locations, are required to address global questions concerning the interior of Mars. These three sites are Centauri Montes, Nili Fossae, and Arsia Mons.

*Centauri Montes Site*: This site provides a location for addressing multiple geophysics objectives (figure 2-7). First, it is one of three sites for global seismic monitoring. Heat flow measurements for this highlands site can be compared to, for example, such measurements in the large volcanic Tharsis province, if the Arsia site is also chosen.

Exploration targets at Centauri Montes include recent gullies (possibly liquid water), ancient Noachian Hellas basin rim constructs, Amazonian debris aprons, and other features that are associated with geologically recent climate change. Figure 2-7 shows several traverses, each requiring an extended period of exploration. During these traverses, specific sites will be selected for in-depth geophysical exploration. The right panel zooms in on one part of the blue traverse, showing two stations (red crosses) where detailed geophysical exploration could be done (MEPAG HEM-SAG, 2008). Active reflection seismology and EM sounding, for example, might be carried out to explore in detail theWS subsurface structure of these lobate debris aprons.

While traversing, some kinds of measurements can be made to map out subsurface structure and state. For example, GPR and neutron spectroscopy would provide cuts of near-surface layering (with sufficient dielectric contrast) and bulk hydrogen estimates as a function of position along the traverse. Perhaps most importantly, geophysical methods can be used to sound the subsurface along the rim of the gullied crater, thus providing information about the presence or absence of an aquifer as a potential gully source. Here the rover team would first explore the crater rim, stopping at promising sites to temporarily emplace geophysics instrumentation such as EM sounding and active seismic systems to characterize the subsurface. Results from these surveys would help to determine the most promising location(s) in which to drill.

*Nili Fossae Site*: The Nili Fossae site is another location for addressing multiple geophysics objectives. Again, it is one of three sites for global seismic monitoring. Heat flow measurements for this dichotomy boundary site, which is far from late Amazonian volcanic activity, provide another important tie-point for interior structure, composition, and dynamics. The stratigraphy of depositional fans, possible lake-bottom deposits, shoreline breaches, and other features could be explored.

*Arsia Mons Site*: The Arsia Mons site opens the exploration of the most important volcanic province on Mars (figure 2-6). Again, it is one of three sites for global seismic monitoring. Heat flow measurements for this Tharsis rise site, where extensive volcanism has occurred since the Noachian, would certainly improve our knowledge of the interior structure, composition, and dynamics of Mars. In this case, the satellite stations may include additional heat flow experiments, searching for evidence of late-stage dike intrusion if cooling time is not too short ($<10^5$ yr). With the inclusion of a local seismic network, seismic velocity anomalies that are associated with deep magmatic bodies will be identified. Active (reflection) seismic studies at many local sites along the traverses will help reveal the history of ash deposits and lava flows. They may also reveal the presence of ice at depths that are consistent with late-Amazonian deposition and subsequent sublimation in current obliquity and climate conditions.

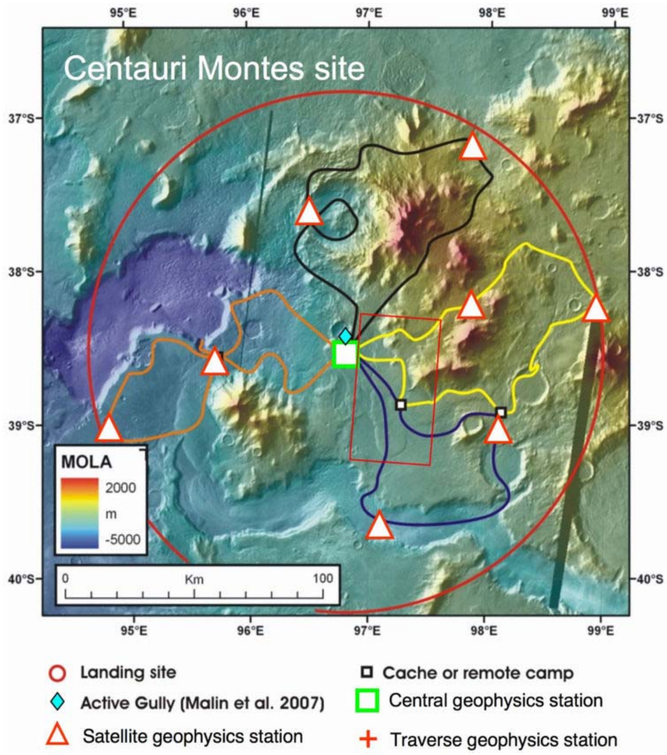


**Figure 2-7.** Centauri Montes mission landing site and traverses.

*Some Geophysics Investigations*: Planetary scale as well as regional scale and local geophysics investigations and approaches are summarized in tables 2-3 and 2-4. In these tables, the first column lists the MEPAG investigations, and the second column identifies the relevant geophysics techniques (in no particular order) that was used to address the objectives of each investigation. For example, a seismic network provides S and P wave travel tSimes from which ray paths and velocities are determined. Models of interior composition and structure must be consistent with these measurables.

**Table 2-3.** Planetary-scale Geophysics: Investigations and Approaches

| Investigation | Geophysics Approaches |
|---|---|
| 1. Characterize the structure and dynamics of the interior. | – Seismology<br>– Heat flow<br>– Gravity<br>– Ultra-low frequency (ULF) EM induction (conductivity profile) |
| 2. Determine the origin and history of the magnetic field. | – High-precision, high-resolution magnetic field measurements<br>– Measurements of the magnetic properties of samples |
| 3. Determine the chemical and thermal evolution of the planet. | – Seimsology<br>– Heat flow<br>– ULF EM induction (conductivity profile)<br>– Gravity<br>– High-precision geodesy<br>– High-precision, high-resolution magnetic field measurements |

**Table 2-4.** Local-scale Geophysics: Investigations and Approaches

| Investigation | Geophysics Approaches |
|---|---|
| 1. Evaluate fluvial, subaqueous, pyroclastic, subaerial, and other sedimentary processes and their evolution and distribution through time, up to and including the present. | – Reflection seimsology<br>– GPR<br>– Gravity<br>– EM induction (conductivity profile)<br>– Neutron spectroscopy |
| 2. Characterize the composition and dynamics of the polar layered deposits. | – Reflection seismology<br>– GPR<br>– Gravity<br>– EM sounding (conductivity profile) |
| 3. Evaluate igneous processes and their evolution through time. | – Reflection seimsology<br>– GPR<br>– Gravity<br>– EM induction (conductivity profile) |
| 4. Characterize surface-atmosphere interactions on Mars, including polar, aeolian, chemical, weathering, mass-wasting, and other processes. | Requires active seismic, EM, neutron spectroscopy |
| 5. Determine the large-scale vertical and horizontal structure and chemical and mineralogical composition of the crust. This includes, for example, the structure and origin of hemispheric dichotomy. | Requires passive, active seismic, gravity, active EM, passive low-frequency EM. |
| 6. Determine the present state, three-dimensional distribution, and cycling of water on Mars. | Requires active seismic, active EM, passive low-frequency EM, instrumented drilling, or wireline sensors |
| 7. Document the tectonic history of the martian crust, including present activity. | Requires gravity, passive and active seismic, active EM, passive low-frequency EM, instrumented drilling, or wireline sensors |
| 8. Evaluate the distribution and intensity of hydrothermal processes through time, up to and including the present. | Requires passive and active seismic, active EM, passive low-frequency EM, instrumented drilling, or wireline sensors (maybe robotic since there is astrobiological potential in hydrothermal systems) |
| 9. Determine the processes of regolith formation and subsequent modification, including weathering and diagenetic processes. | Requires passive, active seismic, active EM, passive low-frequency EM, neutron spectroscopy for hydrogen |
| 10. Determine the nature of crustal magnetization and its origin. | Requires mobile magnetometry, heat flow, passive low-frequency EM (multi-point? a network). |
| 11. Evaluate the effect of impacts on the evolution of the martian crust. | Subsurface mapping via active seismic, EM sounding, etc. |

### *2.2.5.3 Human science reference missions: atmosphere/climate*

**Introduction**

Atmosphere and climate goals and objectives are more varied and less site-specific than geology, geophysics, or life investigations, with the notable exception of climate studies that are associated with polar ice cap drilling. These goals and objectives are summarized in table 2-5. Consequently, we emphasize updated atmospheric and climate objectives and the degree to which they may be advanced by general rather than site-specific human exploration activities on Mars. Meteorological measurements are included in geology, geophysics, and biology reference missions as they are key to characterizing the present-day surface-atmosphere exchange of water and surface weathering. Meteorological measurements, as on Earth, are also expected to be key to planning safe daily human surface operations. For these two reasons, atmospheric reference mission activities are anticipated to be included in all human missions.

Two atmosphere/climate missions are identified: an atmospheric HSRM and an HSRM to the north polar dome for deep drilling to define the more site-specific, human-enabled mission activities that would be necessary to sample the critical volatile records that are contained within the polar ice caps. A third class of activity is associated with the early evolution of climate and would benefit from the return of samples containing gas inclusions to Earth. In the following sections, we will briefly outline and discuss the various atmosphere/climate investigations to be conducted by human explorers on Mars.

**Table 2-5.** Summary of Atmosphere/Climate Objectives

| MEPAG 2030 Goal | Key Objectives for Human Missions |
|---|---|
| Atmospheric objectives | Surface-atmosphere interactions: dynamics, heat and mass balance, non-equilibrium trace gases |
| | Search for sources of volatiles and trace gases |
| Polar Cap objectives | Baseline chronology and characterization of the climate history of the north polar dome (deep core) |
| | Horizontal sampling of the North Polar Layered Deposits (NPLD) |
| Early climate evolution | Long-term climatic evolution of the planet (billion-year temporal scale); implications of early climatic conditions in the emergence of early potential habitats and/or life, which includes inference in the atmosphere chemical state |
| | Sampling of Noachian to Amazonian deposits through soft drilling (~1 meter deep) along outcrops, or deep drilling to capture information in the sedimentary record |

**Overview of updated goals and objectives**

In the human era of exploration, atmospheric measurements at all sites will be seen as important not only to the understanding of the martian atmosphere and climate and to the planning of human surface operations, but also as an environmental characterization that is essential to the interpretation of many life and geology objectives. The trend towards system science called out in MEPAG (2006), a “ground-to-exosphere approach to monitoring the [martian] atmospheric structure and dynamics,” will continue with more emphasis on the mass, heat, and momentum fluxes among the three Mars climate components: atmosphere, cryosphere, and planetary surface.

This systems approach will be enabled by advances in Mars global circulation models (MGCMs), a doubling in length of The global time-series that is derived from monitoring Mars surface and atmosphere from orbit, new atmospheric vertical structure information from Mars Express and Mars Reconnaissance Orbiter (MRO), new anticipated global data sets on aeronomy, atmospheric composition, and winds, and by network science and coordinated lander-orbiter campaigns, such as that planned with Phoenix-MRO. Year 2007 trends in MGCM development are towards coupling of upper and lower atmosphere; coupling with regolith models; integrating models of atmospheric chemistry and dynamics; multiscale, nested models – where small-scale surface-atmosphere interactions can be studied within the context of global transport – and data assimilation. Models have not yet been successful in reproducing the observed martian dust cycle with active dust transport. Temperature and wind profile information from heights between the top of instrumented masts and the free atmosphere will likely remain sparse or nonexistent.

Understanding of the past climate on Mars will benefit from anticipated new knowledge of current atmospheric escape rates from the 2013 Mars Aeronomy Scout. However, a significant advance in the key area of access to the polar stratigraphic record is not expected in the decades before human exploration. In 2030, this will remain one of the highest priorities for MEPAG. On the other hand, the study of the paleoclimatic parameters that are imprinted in the ancient geological record (e.g., Noachian to Amazonian) also concern the high priorities of the MEPAG, which directly relate to unlocking the ancient climatic conditions of Mars through the physical (e.g., geomorphic and/or sedimentary), petrological, mineral, and geochemical (including isotopic) material characterization.

While recognizing that the MEPAG 2006 Goal II objectives are sufficiently general that they will all remain largely valid, some updating relevant to 2030 is captured in the following four subsections.

**Quantitative understanding of Mars atmospheric processes**

The 2006 MEPAG Goal IIA is to characterize what constitutes the basic state and critical processes of the current martian atmosphere. Here we describe the globally active physical processes that determine the basic state and variability of the Mars atmosphere, and so are most important to resolve. These processes are inherently global in character such that relevant measurements may be obtained from human activities at all of the sites that will be visited. There are, however, large-scale atmospheric provinces that exhibit distinctive dynamical, aerosol (dust and clouds), surface, and potential subsurface volatile conditions. Consequently, although site selection is unlikely to be driven by atmospheric science, the specific complement of atmospheric experiments and measurement goals is likely to vary according to site selection.

The emphasis of human atmospheric science measurements will likely focus on processes within the planetary boundary layer (PBL) (surface to ~2 km), where surface-atmosphere interactions impart fundamental influences on the dynamical, chemical, and aerosol characters of the global Mars atmosphere. All spatial scales are important in turbulent exchange, from centimeters to kilometers, in both horizontal and vertical dimensions for the PBL. It is the wide diversity of spatial scales and the driving importance of the near-surface contribution that lead to fundamental limitations of orbital remote sensing; surface field campaigns are still a major thrust of atmospheric boundary layer research on Earth for understanding small-scale variability. Through nonlinear processes, small-scale variability can significantly influence the global climate. Human atmospheric observations can provide optimum in-situ and remote access to the PBL and, in turn, characterize local environmental conditions in support of human operations.

Atmospheric dynamics, in concert with radiative forcing, determine the basic thermal structure of the Mars atmosphere, the global transport of volatiles ($CO_2$, $H_2O$, dust), and the maintenance of Mars polar ice caps, all of which vary on seasonal and inter-annual timescales. Current understanding of Mars atmospheric dynamics is based, to a large extent, on remotely sounded atmospheric temperature profiles, which are analyzed in the context of MGCMs. Recent Mars missions (Mars Global Surveyor (MGS), Mars exploration rover (MER), Mars Express, MRO) have extended the vertical, global, and temporal coverage of atmospheric temperature and aerosol (cloud and dust) distributions towards enhanced constraints on MGCM dynamical simulations. The dynamical state of the upper Mars atmosphere (altitudes above 80 km), which carries additional significance in terms of spacecraft aerobraking and atmospheric escape rates, has been inferred from the in-situ density measurements that are associated with aerobraking (Withers, 2006[24]). Dedicated global observations from the 2013 Mars Aeronomy Scout Mission will greatly expand our understanding of Mars upper atmospheric dynamics. Within the near-surface atmosphere, atmospheric observational constraints remain sparse. This reflects both the limitations of orbital remote sensing and the geological focus of lander/rover operations to date. Viking lander in-situ observations of surface pressure and winds reflect active planetary wave systems and storm fronts (e.g., Barnes,1980[25]; Murphy et al.,1990[26]). MER-based thermal and dust-aerosol profiling within the lower (<5 km) atmosphere also indicates strong PBL variability over local turbulent to diurnal to seasonal timescales (Smith et al.,2006[27]). MSL and Phoenix will conduct limited meteorological measurements as constrained by the primary surface science objectives of these missions. Dedicated observations of surface pressure and temperature-wind-dust profiles of the PBL from distributed surface stations constitute a key priority for human investigations of Mars atmospheric dynamics.

*Atmospheric Dust*: Radiative forcing of the Mars atmosphere may be represented roughly as an energy balance between cooling through $CO_2$ thermal infrared (IR) emission and heating through absorption of solar flux by suspended dust particles. Atmospheric heating that is associated with atmospheric dust intensifies global atmospheric circulation and near-surface winds, which, in turn, increase lifting of surface dust into the atmosphere. A dramatic result of this dust radiative-dynamic feedback is ubiquitous aeolian activity on Mars, with significant dust lofting and transport occurring over a wide range of spatial and temporal scales. These range from nearly continuous dust devil activity, to regional dust storms in every Mars year, to global dust storms that may occur once every 3 or 4 Mars years (Cantor et al., 2001[28]). As a consequence, atmospheric dust plays a major role in the spatial, seasonal, and interannual variability of Mars atmospheric thermal structure and circulation. Global imaging and thermal IR dust abundance observations of Mars atmospheric dust extend from the Mariner 9 mission to Viking, MGS, and the current MER, Mars Express, and MRO missions, thereby providing an accumulating timeline of Mars dust storm activity (McCleese et al., 2007[29]; Wolf and Clancy, 2003[30]). Current mission observations have also substantially advanced vertical profile and dust radiative property definitions. Both of these factors are critical to understanding

[24]Withers, P., (2006) Mars Global Surveyor and Mars Odyssey Accelerometer observations of the Mars atmosphere during aerobraking, *Geophysical Research Letters* **33**, L02201.doi: 10.1029/2005GL024447.

[25]Barnes, J. R., (1980) Time spectral analysis of midlatitude disturbances in the Martian atmosphere, *Journal of Atmospheric Science* 37, 2002-2015.

[26]Murphy, J. R., C. B. Leovy, and J. E. Tillman (1990) Observations of Martian surface winds at the Viking Lander 1 site, *Journal of Geophysical Research* **95**, 14555-14576.

[27]Smith, M. D., M. J. Wolff, N. Spanovich, A. Amitabha, D. Banfield, P. R. Christensen, G. A. Landis, and S. W. Squyres (2006) One Martian year of atmospheric observations using MER Mini-TES, *Journal of Geophysical Research* **111**, E12S13, doi:10.1029/2006JE002770.

[28]Cantor, B. A., P. B. James, M. Caplinger, and M. J. Wolff (2001) Martian dust storms: 1999 Mars Orbiter Camera observations, Journal of Geophysical Ressearch **106**, E10, 23653-23688.

[29]McCleese, D. J., J. T. Schofiled, F. W. Taylor, S. B. Calcutt, M. C. Foote, D. M. Kass, C. B. Leovy, D. A. Paige, P. L. Read, and R. W. Zurek (2006), Mars Climate Sounder: An investigation of thermal and water vapor, dust and condensate distributions in the atmosphere, and energy balance of the polar regions. *Journal of Geophysical Research* **112**, E05S06, doi:10.1029/2006JE002790

[30]Wolff, M. J., and R. T. Clancy (2003) Constraints on the size of Martian aerosols from the Thermal Emission Spectrometer observations, *Journal of Geophysical Research* **108**, E9, doi:10.1029/203JE002057.

the radiative-dynamical relationships that are associated with Mars dust storm activity. A key element that has yet to be addressed regards the particle-size-dependent flux of dust at the surface-atmosphere boundary as a function of atmospheric and surface conditions. Hence, our understanding of dust lifting rates from the Mars surface is characterized by relatively simple surface wind parameterizations, and it remains uncertain as to whether global surface dust distributions limit or are influenced by atmospheric dust transport. In-situ observations of dust surface flux (lifting and deposition), particle sizes, radiative properties, and vertical profiles within the PBL constitute primary objectives for human atmospheric dust studies.

*Atmospheric Water*: Atmospheric water, in the form of vapor and ice clouds, plays significant roles in atmospheric chemistry, dust radiative forcing, and climate balance. The photolysis products of atmospheric water vapor determine Mars trace species abundances (Nair et al., 1994[31]). Water ice clouds have long been associated with major topographic features, autumnal polar hoods, and a variety of cloud wave structures (Kahn, 1984[32]). The existence of an aphelion, low-latitude cloud belt is identified as a significant influence on the vertical distribution of atmospheric dust and water vapor (Jakosky and Farmer, 1983[33]), as well as meridional transport of atmospheric water (Clancy et al., 1996[34]). Atmospheric exchange with polar cap water ice deposits dominates the seasonal variation of atmospheric water vapor, whereas atmospheric exchange with subsurface ice and adsorbed water at lower latitudes remains uncertain. Recent spacecraft observations of atmospheric water vapor (Smith, 2002[35]), subsurface water ice (Feldman et al., 2004), and polar cap water ice (Langevin et al., 2005[36]) from MGS, Odyssesy, and Mars Express have begun to illuminate surface-atmospheric exchanges of Mars water over seasonal, interannual, and possibly longer timescales. The Phoenix Lander (Smith, 2006) will excavate and analyze subsurface water ice on Mars for the first time, and MSL will provide measurements of surface humidity and the water content of surface materials over the course of 1 martian year. HEM studies of atmospheric water are likely to focus on vertical profile measurements within the PBL, which are not easily addressed from orbital remote sensing. Subsurface core sampling of adsorbed water and water ice water deposits, which are site-dependent in this case, also constitutes a key Mars water objective that is uniquely facilitated by human measurements.

*Atmospheric Chemistry*: The trace chemical composition of the current Mars atmosphere reflects photochemical cycles associated with the major atmospheric constituents $CO_2$, $H_2O$, and nitrogen ($N_2$); and perhaps non-equilibrium chemistry that are associated with potential subsurface sources-sinks of methane ($CH_4$), sulfur dioxide ( $SO_2$), and hydrogen peroxide ($H_2O_2$ ) (Levine, 1985[37]; Yung and DeMore, 1999[38]). Some of these compounds can be essential to sustain a Mars cryptic biosphere through direct or indirect (bio)chemical pathways (e.g., atmospheric oxidants can be used as electron acceptors for microbial metabolism, whereas reducing gases –$CH_4$- can be electron donors). Existing measurements of the Mars trace species carbon monoxide (CO), $O_2$, ozone ($O_3$), and $H_2O_2$ appear to confirm the dominant $HO_x$ catalytic cycle that was proposed to prevent buildup of large CO and $O_2$ concentrations from photolysis of the primary $CO_2$ constituent (Parkinson and Hunten, 1972[39]; McElroy and Donahue, 1972[40]). Hence, atmospheric water vapor, as the primary photolytic source of atmospheric $HO_x$ species, plays a dominant role in Mars atmospheric chemistry. Definitions of spatial and seasonal variations in atmospheric trace composition remain tentative, with the exception of Mars ozone, which exhibits large increases towards winter high latitudes (Barth, 1985[41]). The detailed seasonal variation of Mars ozone also suggests that heterogeneous $HO_x$ chemistry may occur

[31]Nair, H., M. Allen, A. D. Anbar, Y. L. Yung, and R. T Clancy (1994) A photochemical model of the martian atmosphere, *Icarus* **111**, 124-150.
[32]Kahn, R. (1984) The spatial and seasonal distribution of Martian clouds and some meteorological implications, Journal of Geophysical Research **89**, 6671-6688.
[33]Jakosky, B. M. and C. B. Farmer (1982) The seasonal and global behavior of water vapor in the Mars atmosphere- Complete results of the Viking atmospheric water detector experiment, Journal of Geophysical Research **87**, 2999-3019.
[34]Clancy, R. T., A. W. Grossman, M. J. Wolff, P. B. James, D. J. Rudy, Y. N. Billawala, B. J. Sandor, S. W. Lee, and D. O. Muhleman (1996) Water vapor saturation at low altitudes around aphelion: A key to Mars climate?, *Icarus 122*, 36–62.
[35]Smith, M. D. (2002) The annual cycle of water vapor on Mars as observed by the Thermal Emission Spectrometer. *Journal of Geophysical Res,earch* **107**, E11, 1-25, doi:10.1029/2001JE001522.
[36]Langevin, Y., F. Poulet, J. P. Bibring, B. Schmidt, S. Doute, and B. Gondet (2005) Summer evolution of the north polar cap of Mars as observed by OMEGA/Mars Express, *Science* **307**, 1576-1581.
[37]Levine, J. S. (Editor) (1985) The Photochemistry of Atmospheres: Earth, The Other Planets, and Comets, Academic Press, Inc., Orlando, 518 pp.
[38]Yung. Y. L. and W. B. DeMore, 1999: Photochemistry of Planetary Atmospheres, Oxford University Press, New York, 456 pp.
[39]Parkinson, T. M. and D. M. Hunten (1972) Spectroscopy and aeronomy of O2 on Mars, Journal of Atmospheric Science **29**, 1380-1390.
[40]McElroy, M. B. and T. M. Donahue (1972) Stability of the Mars atmosphere, *Science* **177**, 986-988.
[41]Barth, C. A. (1985) The Photochemistry of the Atmosphere of Mars. In *The Photochemistry of Atmospheres: Earth, The Other Planets and Comets* (J. S. Levine, Editor), Academic Press, Inc., Orlando, 337-392.

on the surface of Mars water ice clouds (Lefevre et al., 2004[42]). Vertical gradients in trace species abundances, which are associated with a saturation-dependent water mixing profile (Clancy and Nair, 1996) or vertical variations in photolysis rates (Nair et al., 1994), are inferred but not definitively measured. The most problematic trace species measurements, on both observational and modeling grounds, are the recent reported detections of significant atmospheric $CH_4$ abundances (Formisano et al., 2004[43], Krasnopolsky et al., 2004[44]). Methane is not photochemically produced and is not stable in the current Mars atmosphere such that detectable amounts (parts per billion) require a source from the subsurface (Krasnopolsky et al., 2004).. Reported variations in $CH_4$ abundance vs. time and space (Mumma et al., 2007[45]) place further requirements on atmospheric loss rates for $CH_4$, which remain extremely challenging. Subsurface sources for sulfur-bearing gases such as $SO_2$ (Krasnopolsky et al., 2004), and triboelectric sources for enhanced production of peroxide (Atreya et al., 2007[46]) remain unsubstantiated by observations and so unconstrained. MSL, the Mars Aeronomy Scout mission, and MSO should address many of the above questions regarding Mars atmospheric chemistry, including the degree to which subsurface sources of non-equilibrium gases are significant globally. Human observations of atmospheric chemistry are likely to focus on detections of locally enhanced $CH_4$, $SO_2$, hydrogen sulfide ($H_2S$), hydrogen cyanide (HCN), or peroxide concentrations that are associated with confined source regions that are specific to the geology, geophysics, or life site.

*Electrical effects*: Experimental and theoretical investigations of frictional charging mechanisms in both small- and large-scale meteorological phenomena suggest that Mars very likely possesses an electrically active atmosphere as a result of dust-lifting processes of all scales, including dust devils and dust storms. Naturally occurring dust activity is nearly always associated with significant electrification via the process of triboelectricity, which is the frictional charging of dust grains that are in contact with one another or the surface as they are transported by wind or convective circulations. Based on the results of terrestrial experiments and their implications for the presence of electrification processes on Mars, it has been hypothesized that electric fields up to the breakdown potential of 25 kV/m can occur near the martian surface (Delory et al., 2006[47]). A large-scale, electric dipole moment can be generated by nearly any process with a vertical lifting component, as the smaller, negatively charged grains are transported to higher altitudes than the heavier, positively charged grains. In dust devils and dust storms, the vertical stratification of grains based on size and mass will create a stratification of charge, which creates an electric dipole moment with a spatial scale on the order of the storm size.

Electrical effects impact human exploration and the environment of Mars as a source of both continual and episodic energy. Differential charging between separate objects, which are in the presence of electrified dust, that then come into direct contact and cause a discharge will damage electronics or interfere with radio communications. Suspended electrified dust presents a hazard for launch operations (an example of this is the Apollo 12 launch, which was struck by lightning due to the short-to-ground that was caused by the vehicle exhaust trail). Dust adhesion may also be dominated by electrical effects, with implications in terms of its transport into the habitat/human environment where other effects may take over (toxicity, friction in seals/machinery, etc.).

Currently, measurements of electric charging within the Mars atmosphere do not exist, and experiments that are necessary for such measurements are not incorporated in the Phoenix Lander or MSL missions. For operational safety concerns alone, basic measurements of martian surface charging conditions should be obtained prior to HEM activities. Human measurements of atmospheric charging within active dust devils are especially relevant to the dynamic response times that are associated with dust devil occurrences and motions.

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[43]Formisano, V, S. Atreya, T. Encrenaz, N. Ignatiev, and M. Giuranna (2004) Methane on Mars, *Science* **306**, 5702, 1758-1761.
[44]Krasnopolsky, V. A., J. P. Maillard, and T. C. Owen (2004) Detection of methane in the martian atmosphere: evidence of life?, *Icarus* **172**, 537-547.
[45]Mumma, M. J., G. L. Villanueva, R. E. Novak, T. Hewagama, B. P. Bonev, M. A. DiSanti, and M. D. Smith (2007) Absolute measurements of methane on Mars: The current status, *AAS Bulletin* **39**, p. 370.
[46]Atreya, S. K., , A. S. Wong, N. O. Renno, W. M. Farrell, G. T. Delory, D. D. Sentman, S. A. Cummer, J. R. Marshall, S. C. R. Rafkin, and D. C. Catling (2006) Oxidant Enhancement in Martian Dust Devils and Storms: Implications for Life, and Habitability, *Astrobiology* Vol. **6** 3, 439-450.
[47]Delory, G. T., W. M. Farrell, N. O. Renno, A. S. Wong, D. D. Sentman, S. A. Cummer, J. R. Marshall, S. C. R. Rafkin, and D. C. Catling (2006) Oxidant Enhancement in Martian Dust Devils and Storms: Storm Electric Fields and Electron Dissociative Attachment, *Astrobiology* **6,** 3, 451-462.

**Science goals and approach**
These goals require similar investigations; however, a microclimate objective will be more specific, requiring additional planning to optimize site selection for meteorological stations and time-phasing of investigations that are relative to relevant seasonal cycles. Site selection considerations are described under the subsection *Location*.

A proposed baseline is a central station (it could be close to the habitat, but see the constraints on fetch in the *Location* subsection) plus remote stations that is used either to broadly characterize the region (co-sited with major geology/life investigations) or that is arranged to give three-dimensional information on the specific flows that are associated with microclimate. The microclimate objective will also require reference meteorological station(s) to provide regional context. Investigations of atmosphere-surface interactions are summarized in table 2-6.

**Table 2-6.** Investigations of Atmosphere-Surface Interactions and Approaches

| Investigation | Atmosphere/Climate Approaches |
|---|---|
| Monitor basic atmospheric state at reference height above surface (2 m) | MET station:<br>Instrumented mast-sonic anemometer (three heights), temperature (three heights), pressure, humidity, radiation (net, long wave, short wave), dust particle counter<br>Soil heat and conductivity probes<br>Soil temperature profile |
| Monitor the radiation and heat balance for surface-atmospheric exchange and solar forcing | |
| Monitor temperature, wind, dust and cloud through the depth of the boundary layer (two scale heights ~20 km) | Upward-looking thermal IR spectroscopic sounder (water vapor, dust, temperature)<br><br>Tethered balloon, winch, and gondola – sonic anemometer, temperature, pressure, humidity, camera<br>Radiosonde balloon – temperature, pressure, wind, humidity<br><br>Portable Doppler laser imaging detection and ranging (LIDAR) (wind)<br>Portable Raman/imaging lidar (dust)<br>Direct current (DC) electric field sensors<br>Portable differential absorption LIDAR (DIAL) (water vapor)<br><br>Surface accumulation measurements (dust/ice)<br>Microscope analysis of dust |
| Monitor the mass balance for dust and volatile components, especially considering dust-lifting processes and also considering electrical effects | |
| Investigate processes that influence the mass balance for dust and volatile components | |
| Assess the impact of latitude, longitude, season, and local time | Diurnal cycle campaign: tethered balloon sounding each 2 hours, portable instrumentation deployment<br>Seasonal cycle campaign: 3 days of diurnal cycle campaigns, six times over mission, radiosonde release midday/midnight |
| Measure atmospheric composition (trace, isotopes) | Isotope mass spectrometer<br>(2–100 amu [atomic mass unit])<br><br>Sample processing system<br>pH, wet chemistry, microscope |
| Measure physical and chemical properties of the regolith | |
| Measure the deposition of chemically active gases, such as $O_3$ and $H_2O_2$, to the Mars surface. | |
| Search for gases of biogenic ($CH_4$, ammonia (NH), etc.) and volcanic ($SO_2$, $H_2S$, etc.) origin and determine their source(s) | Portable laser diode system or Fourier transform infrared spectrometer (FTIR) for ~100 parts per trillion detection limits |
| Search for sources of atmospheric water vapor | |
| Assess the impact of latitude, longitude, season, and local time on atmospheric composition and the photochemistry of trace atmospheric gases | Chemistry campaign: 3 days, six times over mission |

### *2.2.5.4 Nominal deep drilling polar reference mission*
The polar regions pose unique technical challenges due to cold temperatures and polar night, which may be somewhat offset by access to a ready supply of water and radiation shielding material (ice).

The north polar dome is the target for the first human mission. A deep drilling phase is described followed by traverse to lower latitudes for launch at the onset of polar night. This reference mission does not include explicit life investigations that are related to access to ancient ice or basal melting, or geology investigations that are related to crossing the NPLD, both of which are anticipated to be of high interest. Polar drilling investigations and approaches are summarized in table 2-7.

**Table 2-7.** Polar Deep Drilling Investigations and Approach

| Investigations | Atmosphere/Climate Approaches |
|---|---|
| Deep core and baseline chronology and characterization of major climatic events in past 5 million years | Continuous flow analysis equipment (see habitat capabilities) for dating and composition |
| | Borehole instrumentation: Multi-spectral imager <0.1mm resolution; microscopic imager; thermometer |
| | Returned samples of dust from significant lag deposits |
| Polar cap mass and energy balance for current climate state and seasonal cap formation processes | Requirements as for nominal atmospheric mission |
| Shallow cores to investigate heterogenuity | Hand auger |
| Emplacement of geophysical sensors | Heat probes, seismic sensors |

#### *2.2.5.5 Human science reference missions: biology/life*

Human-enabled biological investigations on Mars will focus on taking samples and making measurements to *determine whether life ever arose on Mars*. This goal is consistent with the 2006 MEPAG goals and priorities, and we do not see this goal changing in the next 30 years.

The search for life on Mars can be generally broken into two broad categories: (1) the search for evidence of past life on Mars (which may or may not still be alive); and (2) the search for present (extant) life on Mars. Both have been, and will continue to be, based on a search for water, since all life on Earth requires water for survival. Abundant evidence on the martian surface of past water activity (e.g., rivers, lakes, groundwater discharge) has led to Mars becoming a strong candidate as a second planet in our solar system with a history of life. With increasing knowledge of the extremes under which organisms can survive on Earth, especially in the deep subsurface, whether martian life is still present today has become a compelling and legitimate scientific question.

The NRC was recently commissioned to do a study to develop "an up-to-date integrated astrobiology strategy for Mars exploration that brings together all the threads of this diverse topic into a single source for science mission planning." This NRC report, which was published in 2007, is entitled, "An Astrobiology Strategy for the Exploration of Mars (NRC, 2007[48]). This report did not consider how to do science with humans, but we nevertheless rely heavily on it and earlier MEPAG documents here as snapshots of current community thinking on astrobiological investigations on Mars.

As pointed out by NRC (2007), the search for life on Mars requires a very broad understanding of Mars as an integrated planetary system. Such an integrated understanding requires investigation of the following:

1. The geological evolution of Mars
2. The history of Mars' volatiles and climate
3. The nature of the surface and the subsurface environments
4. The temporal and geographical distribution of liquid water
5. The availability of other resources (e.g., energy) that are necessary to support life
6. An understanding of the processes that controlled each of the factors that is listed above

Many of these investigations are well under way robotically, and will be much further advanced through additional robotic missions and sample return.

[48]National Research Council (2007) An Astrobiology Strategy for the Exploration of Mars. The National Academies Press, Washington, DC, 118 pp.

## 2.3 The Search for Extant Life

The NRC (2007) suggests a number of high-priority targets that are based on evidence for present-day or geologically recent water near the surface:

1. The surface, interior, and margins of the polar caps
2. Cold, warm, or hot springs or underground hydrothermal systems
3. Source or outflow regions that are associated with near-surface aquifers that might be responsible for the "gullies" that have been observed

MEPAG SR-SAG (2006[49]) noted that sites in which recent water may have occurred might also include some mid-latitude deposits that are indicative of shallow ground ice. Conditions in the top 5 m of the martian surface are considered extremely limiting for life. Limiting conditions include high levels of ultraviolet radiation and purported oxidants as well as most of the surface being below the limits of water activity and temperature for life on Earth. For these reasons, finding evidence of extant life near the surface will likely be difficult, and the search will almost certainly require subsurface access. This was also a key recommendation of NRC (2007).

## 2.4 The Search for Past Life

The NRC (2007) lists sites that are pertinent to geologically ancient water (and, by association, the possibility of past life), including the following:

1. Source or outflow regions for the catastrophic flood channels
2. Ancient highlands that formed at a time when surface water might have been widespread (e.g., in the Noachian)
3. Deposits of minerals that are associated with surface or subsurface water or with ancient hydrothermal systems or cold, warm, or hot springs

## 2.5 Human Science Reference Mission to Address Biological Goals: Centauri Montes

As a demonstration of how HEM-SAG envisions carrying out the biological goals, an HSRM was designed to the Centauri Montes region.

**Why Centauri Montes?**
The Centauri Montes region has drawn attention from astrobiologists as a result of the discovery by Malin et al. (2006) that a flow feature (gully) inside a crater wall has apparently been active in the last decade, thereby providing the intriguing possibility of episodic liquid water at or near the surface. This region has also been well documented for its concentration of young, volatile-rich deposits and figures that feature prominently in recent MGCM simulations at different obliquities, which indicates that the eastern-Hellas region should be receiving significant amounts of water-ice from the south pole (Forget et al., 2006[50]). Centauri Montes is also at the head of major Amazonian/Hesperian outflow channels.

The indicators of ice deposits and liquid water today, as well as the region being associated with outflow channels, provide ample local targets for the search for extant and extinct life. For geological investigations, this region has the attraction of all three primary martian epochs being represented in close proximity. Proposed investigations at Centauri Montes are summarized in table 2-8.

[49]MEPAG SR-SAG (Special Regions Science Analysis Group) (2006), Findings of the Mars Special Regions Science Analysis Group, Astrobiology 6, 677-732. The document can also be accessed at http://mepag.jpl.nasa.gov/reports/index.html.
[50]Forget, F.; Haberle, R. M.; Montmessin,F.; Levrard,B.; and Head, J. W. (2006) Formation of Glaciers on Mars by Atmospheric Precipitation at High Obliquity. *Science* 311, 368-371.

**Table 2-8.** Investigations at Centauri Montes and Approach

| Investigation | Biology Approaches |
|---|---|
| 1. Characterize complex organics | – Use of Raman and gas chromatograph mass spectrometer<br>– Screening for thousands of biomolecules (electron transport molecules, various key proteins, phospholipids, etc.) using life marker chips and other lab-on-a-chip assays<br>– Equipment for nucleic acid extraction (assuming that the crew have it) and some sequencing capability<br>– Basic staining for biomolecules |
| 2. Characterize the spatial distribution of chemical and/or isotopic signatures | – Use of isotope mass spectrometers and GCs<br>– Tabletop scanning electron microscope (SEM) |
| 3. Characterize the morphology or morphological distribution of mineralogical signatures | – x-ray diffraction (XRD), laser-induced breakdown spectroscopy (LIBS)<br>– Bright field microscopy |
| 4. Identify temporal chemical variations requiring life | – Basic metabolic analysis (Viking style experiments, but using nonorganic redox couples) |

**Location**
Centauri Montes is located on the rim of Helles Basin. The landing site would be adjacent to the crater where a recently active gully was discovered (hereafter referred to as the active gully crater). This crater is located near 38.7°S, 263.3°W (see figure 2-8).

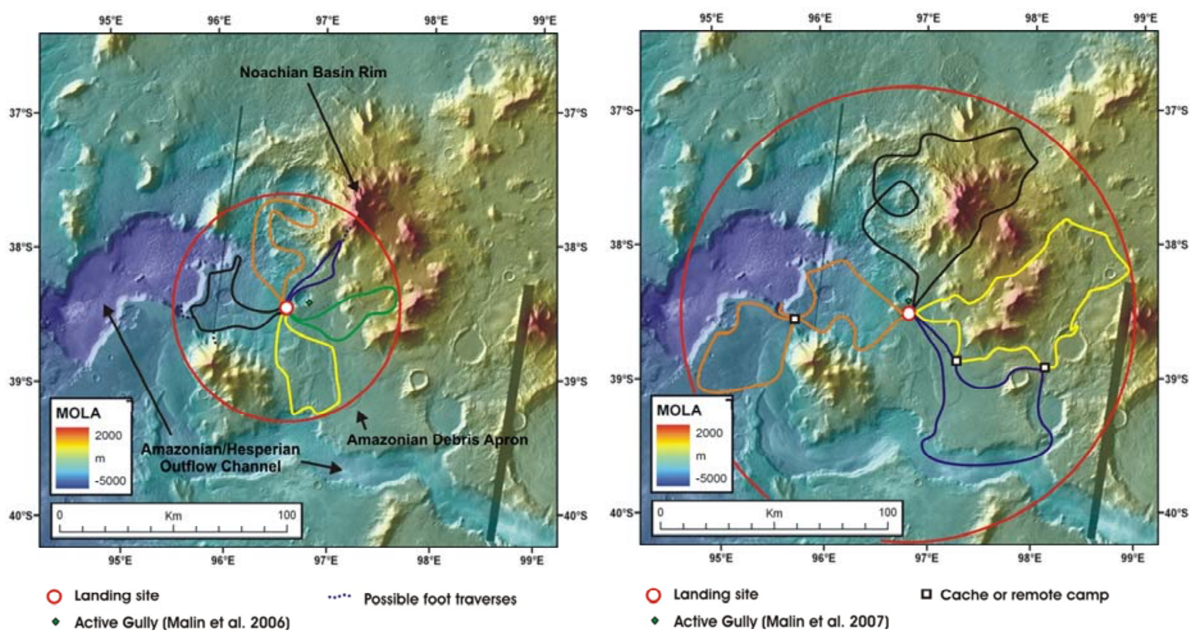



**Figure 2-8.** Comparison of possible traverses from base camp.

*Research plan at Centauri Montes*
Two modes of research would be carried out at CM. These are detailed below.

**Mode 1 — Active gully investigations and local drilling**
This mode of research is primarily focused on assessing the recently active gully and other fresh gullies as potential sites of recently water activity and, hence, extant life.

1. **Drilling.** For this activity, horizontal mobility would be minimal because it is largely dependent on how close to the active gully crater a suitable landing ellipse can be placed. The drill rig would need to be portable enough to be moved from the landing site to the drill site. Alternatively, it could be moved from the landing site on the rover in pieces and assembled at the drill site. We envision the drill site and landing site to be close enough that daily commuting could occur between the two. Drill samples (cores and/or cuttings) would need suitable on-site storage to keep them protected (as close to their ambient conditions as possible) until such time as they can be moved back to base (presumably at the end of each work shift). Once back at base, cataloging, sub-sampling, and analyses would be done in the habitat laboratory. If there is suitable interest from other disciplines (e.g., geophysics, geology), other, but not necessarily as deep, holes could be drilled in the local area for specific goals of geology, geophysics, and climate studies.

2. **Direct measurements and sampling from the active gully.** Based on available data, this seems to be achieved most easily by descending to the gully site from above.

3. **Sampling of sediments on the crater floor.** The available imagery of the active gully crater suggests a history of fluid flow through this crater, possibly associated with the gullies. Drilling on the crater floor into some of these sediments, even to shallow depths, would be useful for seeking out evidence of past life.

We have allotted approximately half of the mission time to this detailed investigation of local and potentially active gullies.

**Mode 2 — SAMPLING TRAVERSES**
The second half of the expedition would be spent traversing out to a radius of 50 km away from the landing site to access materials from the three different epochs and collect samples for investigation of past life. For the astrobiology work, we would only do minimal analysis in the field and would return many samples to the base lab for detailed analysis.

**Planetary protection issues at the human science reference mission locality and potential mitigation**
To achieve the life goals, especially the search for extant life, we will almost certainly need to enter special regions (e.g., gully sites and the subsurface) with humans. We feel that a biologically focused mission needs to include a search for extant life, so technological developments are needed to prevent forward contamination and provide a safe barrier for astronauts who are working on samples. Detailed procedures and protocols for the mitigation of forward contamination must be developed prior to human exploration on Mars.

### 2.5.1 Some summary implications for Goals I–III
The following questions provide, in summary, implications for Goals I–III:

- *For the first three human missions: three different sites or the same site?*
  Three independent sites.

- *For the first three human missions, short stays (~30 days) or long stays (~500 days)?*
  Three long stays (~500 days) to maximize scientific return (figure 2-9).

- *How much surface mobility, in terms of radial distance from the landing site, is required to perform the required science?* (**Note:** In this section, the HEM-SAG team provides quantitative estimates for several key mission parameters; e.g., the range of human mobility, the subsurface depth for drilling, and the mass of Mars samples to be returned to Earth. The estimates for these parameters are based on discussions of the HEM-SAG team and represent consensus estimates that are consistent with the team's goal of maximizing scientific discovery by humans on Mars).
  To determine the radial distance from the landing site that humans will explore, HEM-SAG estimated a radial distance range requirement in the range of 250–500 km. This estimate was based on the great diversity of features on the surface of Mars, as indicated in the Mars surface terrain maps that were developed for the HSRM scenarios. While the radial distance is an estimate, HEM-SAG concluded that human explorers on Mars should be able to traverse radial distances on the order of several hundreds of

kilometers, as opposed to radial distances of several tens of kilometers. The important consequence of this human mobility range is that it requires the use of a pressurized vehicle.

- *What are the subsurface access requirements?*
  To determine the depth of vertical subsurface access (drilling depth), HEM-SAG estimated a drilling depth in the range of 100 to 1,000 m, depending on the drilling site and the scientific goal of the drilling; e.g., subsurface liquid water zones or recoverable polar coring. For example, drilling into a gulley to search for subsurface water may require less depth than drilling to obtain a continuous core sample through the thick ice at the poles to investigate past atmospheric composition and climate.

- *How do we implement a search for extant life vs. a search for fossil life (assessing the value of human explorers on the search for extant life)?*
  Very carefully, because the search for extant life is an important scientific goal/objective of human exploration. The use of human explorers in the search for extant life on Mars will require that we address the following issues:

  - Human in in-situ analyses on Mars vs. returning samples to Earth for analyses (mass of instrumentation/equipment transported from Earth to Mars for in-situ analyses on Mars vs. amount of sample mass to be returned to Earth)
  - Human habitat/workstation: In-situ sample analysis and cataloging; performing analyses that cannot be performed on Earth (e.g., tests for extant life)
  - Samples include: rocks, drill cores, surface/atmospheric dust, ice, atmospheric gas
  - Sample conditioning and preservation essential
  - Human habitat laboratory instruments for multiple objectives: geology, atmosphere/climate, and life
  - Emplacement of network stations for geophysics, atmosphere/climate, and even life is essential beyond initial landing site (250–500 km radial from landing site) to be operated during and after humans return to Earth (see above discussion on human mobility)

- *How much sample mass should be returned to Earth?*
  The mass of samples that will be returned to Earth will be >250 kg.

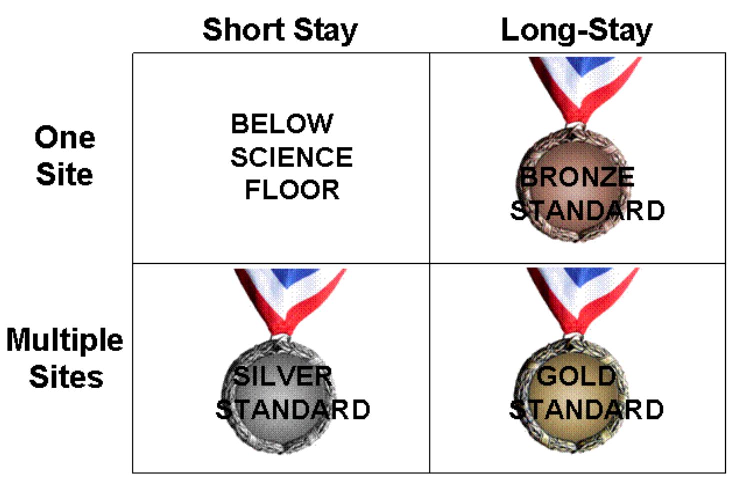



**Figure 2-9.** Value from the perspective of our scientific goals of stay time and landing site diversity.

### 2.5.2 Conclusion

Over the last decade, the exploration of Mars by robotic orbiters, landers, and rovers has shown Mars is a planet of great diversity and complexity. The great diversity and complexity of Mars offers a unique opportunity for humans who are on the surface of Mars to obtain data and measurements that could not be obtained by robotic probes alone. Due to the great diversity and great complexity of Mars, HEM-SAG strongly recommends that the first three human missions to Mars should be to three different geographic sites, and should be long-stay missions of about 500 days' duration on the surface. HEM-SAG has addressed the key scientific questions about Mars as detailed in "Mars

Scientific Goals, Objectives, Investigations and Priorities," 2006 (MEPAG, 2006), has assessed the questions that cannot or will not be addressed or fully addressed by the year 2025, and has developed a series of HSRMs to be pursued by human explorers on the surface of Mars to answer these key scientific questions. HEM-SAG concludes that human mobility is key for the human exploration of Mars, and strongly recommends a pressurized mobility system for human exploration over greater surface distances. HEM-SAG believes that deep drilling is an important function that humans will perform on the surface of Mars and strongly recommends that deep drilling equipment be included on the first three human missions. HEM-SAG also strongly recommends that a well-equipped scientific laboratory for the human analyses of rocks, dust, ice, atmospheric gases, etc. on the surface of Mars be a key aspect of the human exploration of Mars. This on-site laboratory will permit analysis of martian material in its natural environment, and will significantly reduce the mass of material that will have to be transported back to Earth for analyses. HEM-SAG also believes that the impact of human explorers and potential "human contamination" of the Mars environment in the search for present-day life on Mars is a problem that requires more study and evaluation, but will be solved prior to the first human landing on Mars.

## *2.6 Objectives Related to Preparation for Sustained Human Presence (Goal IV+)*

### 2.6.1 Introduction

The MEPAG has developed several goals for Mars exploration, the fourth of which (i.e., Goal IV) is to "prepare for human exploration." The goals and objectives activity of the MAWG defined the MEPAG Goal IV as "preparation for later sustained human presence." This MEPAG goal, which is referred to as Goal IV+, specifically focuses on Mars human habitability, exploration systems development, and long-duration space mission operations. The purpose of the Goal IV+ study was to identify the objectives for the first three human Mars missions that would support the performance of human Mars missions four through 10. The scope of the representative scenarios for missions four through 10 includes developing the knowledge, capabilities, and infrastructure that are required to live and work on Mars, with a focus on developing sustainable human presence on Mars.

### 2.6.2 Ground rules and assumptions for later sustained human presence

The results of the Goal IV+ study are based on specific GR&As. The first states that the initial three human missions will demonstrate the transportation of humans from the surface of Earth to the surface of Mars. Missions one through three will also have Mars surface-stay times of at least 30 days and potentially greater than 450 days.

The second states that there are two potential campaign paths after completion of the third human mission: sortie mode and outpost mode. A sortie mode would involve individual missions that are self-contained and do not rely on elements or materials that were used by a previous mission. A sortie does, however, allow for pre-positioning of supplies and elements. The outpost mode assumes that each of the human missions to Mars will integrate its elements and materials with those from previous missions, building up to a greater capability for future missions. The Goal IV+ study assumed that human missions four through 10 will focus on an outpost mode of operations at either one or several sites. This reflects a conservative approach to the need for new capabilities. If missions one through three were sorties and the campaign was to continue in sortie mode, the first three missions will have demonstrated the capabilities that are required for such operations and, therefore, missions four through 10 would not need any new capabilities. If missions one through three were to develop an outpost, missions four through 10 would continue to support the outpost buildup. The continued outpost buildup would be the stressing campaign for objective development due to the possible need for new capabilities for these later missions where the outpost can continue to change and evolve as it grows.

### 2.6.3 Study process

The Goal IV+ study included several steps over the course of the MAWG Phase 1 and Phase 2 efforts, spanning from March to August 2007. MAWG Phase 1 determined potential objectives for human missions four through 10, starting with the lunar exploration objectives that were developed during 2006 and released at the 2nd Space Exploration Conference (December 2006; Houston). The lunar objectives list was reviewed, and those objectives that were applicable to Mars missions were identified and appropriately edited. An agency-wide team of subject matter experts was formed to review the resulting list of Mars objectives. This team was tasked with modifying, adding, and/or deleting objectives as needed. The goal of this team was to develop a comprehensive data set of possible objectives for missions to Mars (without gaps or implementation specifics) that would enable all applicable future technology trades. The Phase 1 efforts concluded with the list of objectives, which was finalized by a team of Mars multidisciplinary experts. A final assessment of the data set was completed in which the grouping of the objectives into higher-level categories was reviewed, and any remaining gaps were identified and filled with appropriate objectives.

In MAWG Phase 2, the Mars multidisciplinary expert team categorized the lower-level objectives resulting from MAWG Phase 1 into four major objective areas and confirmed the 21 sub-objective categories. Using this updated list, the team then provided input to the Mars Strategic Integration Group regarding the surface-stay duration of a human mission. Surface-stay duration impacts the length of the transit between Earth and Mars as well as the capabilities and infrastructure that are needed for a crewed mission. The team recommended a long surface stay, as this extended time would enable more testing of proof of capabilities for extended human presence. Next, the team drafted recommendations of objectives for human missions one through three. These were based on synthesis of information from each of the lower-level objectives, such as importance towards the completion of Goal IV+, whether they would be best satisfied by long or short surface stays, and whether their completion would be aided by repeated visits to the same site. Finally, a Red Team was formed to review these final recommendations, ensuring accuracy and completeness.

### 2.6.4 Major Goal IV+ objectives decomposition

The four major Goal IV+ objective areas are: Mars Human Habitability/ISRU, Exploration Systems Development, Operational Capabilities, and Other. Within each of these areas are multiple categories of lowe- level objectives, as shown in table 2-9.

**Table 2-9.** Goal IV+ Objective Decomposition

| Mars Human Habitability/ISRU | Exploration Systems Development | Operational Capabilities | Other |
|---|---|---|---|
| Human Health | General Infrastructure | Crew Activity Support | Planetary Protection |
| Environmental Characterization | Operational Environmental Monitoring | | Historic Preservation |
| Environmental Hazard Mitigation | Life Support | | Commercial Activities |
| Mars Resource Utilization | Habitation Systems | | Global Partnership |
| | EVA Systems | | Public Engagement |
| | Power | | |
| | Communications | | |
| | Position, Navigation, and Time | | |
| | Transportation | | |
| | Surface Mobility | | |
| | Operations, Testing, and Verification | | |

### 2.6.5 Study findings

#### *2.6.5.1 Projected Mars exploration objectives for human missions one through three*

Within each of the four major Goal IV+ objective areas, the Goal IV+ study team defined projected Mars Exploration Objectives for human missions one through three. There are three resulting objectives in the Mars Human Habitability/ISRU area. The first is to develop the capability of providing crew needs from local resources; an example of this is in-situ food production. The second objective is to develop the capability of extracting power and propulsion consumables from local resources. This could be accomplished through ISRU processing of the martian atmosphere or regolith to produce $CH_4$ or other chemicals that are needed for power and propulsion technologies. The third objective is to develop and test the capabilities that are needed for in-situ fabrication and repair. This could be accomplished by fabricating infrastructure element replacement parts on the martian surface from raw materials brought from Earth, or by reusing parts from other infrastructure elements that are no longer in use (e.g., a descent stage that is used only for landing on the surface).

The Exploration Systems Development area includes three objectives, all of which relate to the establishment of reliable and robust space systems that will enable gradual and safe growth of capabilities. The first such capability is the number of individuals that can be supported by the infrastructure on Mars. The exploration systems that are developed will also work to increase the duration of time that individuals can live safely on the planet. Another thrust will be the gradual increase in the range of mobility that is provided to visiting crews. As each of these capabilities is realized and matured, the potential for even greater exploration, science, discovery, and new technology is greatly enhanced.

The level of self-sufficiency of operations for Mars missions also must increase and, hence, is the objective in the Operational Capabilities area. Due to the complexity of procedures and the communications delay among other factors, a crew that is operating on the surface of Mars will need to be independent from the support personnel who are located back on Earth. These Earth-based teams will, of course, be available to offer assistance in nonemergency situations. However, the new complications of a martian mission warrant consideration of a day-to-day level of autonomy that is not currently present in space shuttle and ISS missions.

The study resulted in three "Other" objectives addressing: planetary protection concerns, partnerships, and public engagement. Planetary protection concerns were well described by the Space Studies Board (SSB) of the NRC in 2006: "Increased scientific understanding of the [martian] environment and the ability of microorganisms to survive in severe conditions have important implications for the planetary protection of Mars. … Anticipated Mars missions will likely travel to locations with greater potential for the survival and possibly the growth of Earth microbes. The science and engineering community needs to ensure, on an ongoing basis, that planetary protection policy and practices reflect current scientific and technical understanding and capabilities." [Preventing the Forward Contamination of Mars (2006), p. 111] Therefore, special care must be taken not to contaminate the natural environment, where scientific measurements will require pristine samples, as well as any areas to which the human crews will be exposed so as to protect their health. Sustained human presence on Mars will require the development of partnerships. Promoting agreements and collaboration among government, international, commercial, and other entities will be a necessary challenge. Another objective in this area is to provide and sustain public engagement. The exploration of and sustained human presence on Mars will obviously be a grand undertaking, requiring long-term, continual public support. This objective is in direct alignment with NASA's "commitment to communicate with key partners and stakeholders, including elected public officials, the media, the public, academia, other government agencies, and international space agencies, to enhance understanding of the agency's programs, policies, and plans and to advance the nation's space program agenda," as well as to continue "the [agency's] tradition of investing in the [nation's] education programs and supporting the country's educators who play a key role in preparing, inspiring, exciting, encouraging, and nurturing the young minds of today who will manage and lead the nation's laboratories and research centers of tomorrow." [2006 NASA Strategic Plan pg. 29].

#### *2.6.5.2 Recommendation based on Goal IV+ objectives*

The Goal IV+ team has two primary recommendations. First, the team recommends long surface stays for human Mars missions. Longer stays allow for a more comprehensive characterization of certain environmental parameters and a longer baseline of measurements. This specific and long-duration knowledge will be essential in the development of health monitoring and hazard mitigation strategies for the crew and infrastructure elements. The systems that are required for long stays are also more supportive of the longer-term missions that will achieve sustained human presence in the future than those that would be used on a short surface stay mission. Examples of some of these systems that will benefit from the longer durations early in the overall Mars campaign include habitation, life support, EVA, mobility, and ISRU. Longer-stay missions are also preferred from an operations viewpoint. Crewmembers are expected to be in a de-conditioned state for up to 7 days following landing due to the length of transit from Earth to Mars and the effects of the zero- or microgravity environment during this journey. On a mission with a short surface stay, only 3 weeks of surface operations would remain after this recuperation time, limiting the support of Goal IV+ objectives.

A Goal IV+ team's second recommendation pertains to the number of landing sites for human missions to Mars. The Goal IV+ objectives lend themselves best to repeated visits to a specific site on Mars. Repeated site visits enable a buildup of infrastructure that would benefit the longer-term missions of the Goal IV+ objectives. This buildup would provide more systems for use by the crews such as a habitable volume, mobility aids, and science equipment. These systems and the potential for spares could also potentially reduce the amount of logistics required for the long-term missions.

The Goal IV+ team ranked combinations of surface-stay duration and number of sites as shown in figure 2-10. Short surface stays to either one or multiple sites were ranked as the Bronze Standard. The rationale for the lower ranking of these options is cited above. Long surface stays at multiple sites is the Silver Standard. The long stays elevate this option to a higher level. The Gold Standard is defined as long surface stays to one site, with the single landing location being the enhancing discriminator over the previous case. Multiple long-stay human missions to one site enable the benefits mentioned in the rationale for repeated site visits to be added to the long-duration benefits, hence the gold designation. The team has concluded that long-surface-stay missions at a Mars base with repeated site visits would be most advantageous towards preparation for later sustained human presence.


**Figure 2-10. Value from the perspective of Goal IV+ of stay time and landing site diversity**

## *2.7 Objectives Related to Other Classes of Science (Goal V)*

### 2.7.1 Heliophysics

Heliophysics is the study of the mechanisms and processes of the solar system as driven by the sun. This science encompasses solar activity and stellar cycles, space plasmas, cosmic rays, particle acceleration, space weather, radiation, magnetic reconnection, and magnetic fields. Heliophysics embraces science that safeguards the journey of exploration by developing the capability to forecast both hazardous and safe working conditions in space for human and robotic explorers. To address the issues that are relevant to human exploration of Mars in this science discipline, the MAWG requested the following assessment by a subpanel of the NASA Advisory Council Heliophysics Subcommittee.

The martian system is of tremendous interest to the science of Heliophysics, both as an archive of solar evolution and as a case of planetary interfaces responding to solar influences. These influences range from solar irradiance and high-energy particles pounding the surface of the planet, to solar wind and magnetic fields interacting with the martian atmosphere and ionosphere. Mars also represents an important key instance of fundamental Heliophysical processes that influence the habitability of planetary environments. Because the space environment matters to the safety and productivity of humans and their technological systems both at Mars and in transit, it is essential that we monitor Heliophysical conditions between Earth and Mars, and understand the solar effects in Mars orbit and on the atmosphere and surface environment.

Our solar system is a fascinating nested system that is so closely connected that an explosive event on the sun produces measurable effects that span the entire solar system and heliosphere. Through judicious use of a number of operating missions, the international community has achieved system surveillance over parts of the heliosphere and has been able to examine causal linkages between its elements. Our nation’s challenge to establish a sustained presence on the Moon and enable human exploration of Mars and beyond presents great opportunity and sobering demands for Heliophysics. A host of interconnected physical processes, which are strongly influenced by solar variability, affect the habitability of alien environments and the requirements for the health and safety of travelers in space. Heliophysics is poised to develop the quantitative knowledge that is needed to help assure the safety of the new generation of human and robotic explorers.

Using recommendations of the 2003 NRC decadal survey, "From the Earth to the Sun: A Decadal Survey for Solar and Space Physics," and the 2006 community roadmap "Heliophysics: The New Science of the Sun-Solar System Connection," NASA designed a Heliophysics Science Plan with objectives that represent a science community consensus on priorities. The following sections highlight those top-level Heliophysics objectives that relate to human exploration of Mars – the Science Frontier, Planetary Habitability, and Safeguarding the Journey of Exploration – and identify relevant research focus areas and specific investigations. A final section lists the methods of investigation that will accomplish these objectives as part of the Mars reference architecture.

#### *2.7.1.1 Understanding the fundamental processes that control Mars' space environment*

Planetary upper atmospheres, the sun, our solar system, and the universe consist primarily of plasma, resulting in a rich set of interacting physical processes and regimes, including intricate exchanges with the neutral environment. Human explorers can anticipate encounters with hazardous conditions stemming from ionizing radiation on our return to the Moon and journey to Mars. We must develop mitigation strategies and a complete understanding of the many processes that occur with such a wide range of parameters and boundary conditions within these systems. We must be able to predict the behavior of the complex systems that influence the hazardous conditions crews will encounter.

The processes of interest occur in many locations, although with vastly different magnitudes of energy, size, and time. The same processes rule the seething atmosphere and interior of our sun, the supersonic wind of particles that our star flings outward into space, Mars' limited magnetosphere, the variability of the martian ionosphere and the tenuous upper atmosphere, and even the fantastically energetic spinning pulsars that spray out beams of x rays.

By quantitatively examining similar phenomena occurring in different regimes through a variety of measurement techniques, we can identify important controlling mechanisms and more rigorously test our developing knowledge. Both remote sensing and in-situ observations must be used to provide the three-dimensional, large-scale perspective and the detailed small-scale microphysics view that are necessary to see the complete picture.

On Earth, the lower atmosphere is periodically pumped and heated, giving rise to a spectrum of small-scale gravity waves and longer-period oscillations. These waves can propagate into the mesosphere and thermosphere, depositing momentum. The atmospheric mean circulation is thereby modified, resulting in changes to the temperature structure and redistribution of radiation absorbers and emitters. The mean wind and temperature structures in turn influence the propagation of the waves and the manner in which they couple the lower and upper atmosphere. Similar processes are also key to understanding the upper atmosphere weather and climate on Mars.

The following research focus area and investigations are necessary to obtain knowledge of these fundamental processes:

*Understand the role of magnetic fields, plasmas, and neutral interactions in the nonlinear coupling of regions*

- What governs the coupling of neutral and ionized species at various spatial and temporal scales?
- How do energetic particles and the solar wind modify planetary environments and their chemical and isotopic composition?
- How do the martian magnetosphere, ionosphere, and atmosphere interact with one another?
- How does the neutral environment in the martian system affect the global morphology of the planet through charge exchange and mass-loading processes?
- How do planetary dynamos function, and why do they vary so widely across the solar system?

#### *2.7.1.2 Understanding the influence of planetary magnetic fields*

Plasmas and their embedded magnetic fields affect the formation, evolution, and destiny of planets and planetary systems. The heliosphere partially shields the solar system from galactic cosmic radiation (GCR). Our habitable planet is shielded by its magnetic field and atmosphere, protecting it from solar and cosmic particle radiation. The atmosphere is protected from solar wind erosion by the magnetosphere. Planets without a shielding magnetic field, such as Mars and Venus, are exposed to those processes and evolve differently. Moreover, on Earth the magnetic field changes strength and configuration during its occasional polarity reversals, altering the shielding of the planet from external radiation sources. How important is a magnetosphere to the development and survivability of life? Properties of solar activity and energetic particle transport conditions in the heliosphere change dramatically over timescales ranging from days to millenia. How do these short- and long-term changes affect life and its sustainability in our solar system? Does Mars contain space weather climate archives of the near and distant past?

Magnetic fields play a central role in the formation of planetary systems from disks of gas and dust around young stars. Stellar ultraviolet emission, winds, and energetic particles influence this process by ionizing matter and making it electrically conducting. This in turn allows magnetic coupling of rotation and transfer of momentum and energy, with the side effect of ablating volatile materials that are present. Mars is a key member of the planetary zoo for studies of these effects because it has apparently lost a magnetic dynamo and much of its atmosphere. Through comparison of Mars with the other terrestrial and giant gas planets, we may better understand the types of planets that can form in stellar systems, and how they later evolve in terms of habitability.

The following research focus areas and investigations address the question of planetary habitability:

**Specify, predict, and mitigate ionospheric effects**

- What role does the electrodynamic coupling between the martian ionosphere and the magnetosphere play in determining the response to solar disturbances?
- How do magnetic and electric fields as well as currents evolve in response to solar disturbances?
- How do the coupled middle and upper atmospheres respond to external drivers and to each other?

**Depict magnetic influences on planetary system evolution and habitability**

- What is the role of planetary magnetic fields for the development and sustenance of life?
- What can the study of planetary interactions with the solar wind tell us about the evolution of planets and the implications of past and future magnetic field reversals at Earth?
- Does Mars harbor unique, long-term climate records that are equivalent to ice cores on Earth; and what can these records tell us about past levels of solar activity and its influences on the atmosphere and surface of Mars?

#### *2.7.1.3 Maximizing safety and productivity for human explorers*

Hazards in planetary environments must be understood, characterized, and mitigated. We must understand how space weather impacts planetary environments in ways that affect exploration activities, from spacecraft staging in LEO to transfer orbits, on through entry, descent, and landing (EDL) at Earth and Mars. Reliable communications and navigation for spacecraft and surface crews will require improved understanding of Earth’s and martian ionospheres. Although the sun and its variability drive these environments, many internal processes must also be understood.

Energetic particles from the sun propagate along the normal spiral magnetic field that is embedded in the solar wind. However, coronal mass ejections (CMEs) routinely disrupt the field lines and alter particle transport paths. Future spacecraft in transit to Mars will undergo 6 to 9 months in cruise phase far from either Earth or Mars. This phase will require characterization, “now-casting,” and forecasting capability based on measurements at the spacecraft itself, as well as diverse other observing points. Measurements from a wide range of heliospheric longitudes and remote sensing of the sun will be required to accurately characterize and, ultimately, predict the extremes of space weather that will be encountered during cruise phase. Continuing measurements and study of magnetospheric conditions are also necessary to characterize and predict the highly variable radiation belt environment through which astronauts will depart from, and arrive back at, Earth.

**Characterize space environmental variability and extremes**

- What are the variability and extremes of the radiation and space environment for exploration at Mars?
- How does the radiation environment vary in space and time, and how should it be sampled for situational awareness during exploration?
- What is the relative contribution from solar energetic particles and cosmic radiation behind the various shielding materials that are used and encountered, and how does this vary?

**Understand and characterize space weather effects**

- What level of characterization and understanding of the dynamics of the atmosphere is necessary to ensure safe aerobraking, aerocapture, and EDL operations at Mars?
- To what extent do ionospheric instability and seasonal and solar-induced variability affect communication system requirements and operation at Earth and Mars?
- What are the effects of energetic particle radiation on the chemistry, isotopic composition, and energy balance of the martian atmosphere?
- What are the dominant mechanisms of dust charging and transport on Mars that impact human and robotic safety and productivity?

#### *2.7.1.4 Reference investigation measurements for Heliophysics at Mars*

The methods of research that are enabled and required by human exploration of the surface of Mars can be summarized concisely as follows, with a notation tracing to the top-level objectives of Science Frontier (F), Planetary Habitability (H), and Safeguarding the Journey (J):

1. **Planetary coronal imaging for aeronomy from orbit**: This includes any of the available techniques for remotely sensing and analyzing the properties of the global extended atmosphere of Mars. (F, J)
2. **Ionospheric and planetary current system magnetometry from orbit and from the surface**: Ionospheric currents and surface magnetic features are of great scientific interest, and can be investigated with standard tools of high heritage. (F)
3. **Entry and descent accelerometry**: All orbital or lander vehicles are potential sources of knowledge about the dynamics and structure of the upper atmosphere, which is applicable to the planning of aerobraking. (J)
4. **Global electric circuit and ionospheric linkages**: This is approached with orbiting and ground-based radiowave receivers and field probes. (F, H, J)
5. **Ionospheric radio propagation effects**: The plasma structure of the ionosphere will introduce certain constraints on communications and are also of great scientific interest. This, too, can be addressed using radio receivers and transmitters that are placed on the surface and in orbit. (F, J)
6. **Surface and in-situ (CEV) measurement of cosmic rays and solar energetic particles**: Cruise-phase and surface-real-time monitoring of in-situ ionizing radiation, which is essential for astronaut radiation safety, provides insight into charged- and neutral-transport processes through the heliosphere and Mars atmosphere. (H, J)
7. Suborbital exploration of atmospheric dynamics and composition: Balloons, uncrewed aerospace vehicles (UAVs), and even small sounding rockets would make many useful measurements of atmospheric structure and dynamics. (F, H)

*Surface and core sample return for long-term space climate studies*: Interdisciplinary investigation between Heliophysics and planetology may reveal the history of the interaction of cosmic rays and solar energetic particles with the martian atmosphere. (F, H)

### 2.7.2 Astrophysics on Mars

The architecture for human exploration of Mars creates additional opportunities for realizing the science priorities in astrophysics. The goal of astrophysics is to discover the origin, structure, evolution, and destiny of the universe, and to search for Earth-like planets. While observations from free space offer the most promise for significant progress in broad areas of astrophysics, the Mars DRA presents an opportunity to consider investigations that are uniquely enabled by the infrastructure and capabilities of a human mission to Mars.

A similar opportunity occurred during the 2006 development of the architecture to return humans to the Moon. In November 2006, a workshop on “Astrophysics Enabled by the Return to the Moon” was organized by the Space Telescope Science Institute (STScI), in collaboration with Johns Hopkins University, the Association of Universities for Research in Astronomy, and NASA. That workshop identified astrophysical observations that could either directly or through the capabilities that are developed by the lunar architecture, provide opportunities for significant progress toward answering questions in astrophysics. Among the most promising in this respect are laser ranging experiments to test a certain class of alternative theories (to general relativity) of gravity. Such experiments become even more valuable when considered in the context of a humans-to-Mars architecture. Mars provides a unique capability that would otherwise not be enabled by free space implementations or via a lunar architecture.

The placement of small laser-ranging transponders on Mars would provide several superb results in astrophysics and planetary science. Tests of Einstein's general theory of relativity (GR) could be performed with unparalleled accuracy, at least an order of magnitude better than currently exists. The range to Mars would be improved from current meter-level accuracy down to the centimeter level, thereby improving the ephemeris of Mars by more than an order of magnitude. The mass of Jupiter could also be more accurately determined. Improved measurements of Mars' rotational dynamics could provide estimates of its core size. The elastic tidal Love number is expected to be less than 10 cm – within reach of laser ranging.

#### *2.7.2.1 Laser ranging*

With the first placement of retroreflectors on the Moon by the Apollo astronauts in 1969, lunar ranging was enabled at the centimeter level of accuracy. Laser pulses from ground stations on the Earth illuminate the lunar arrays of retroreflecting cubes, and the timing of the reflected signals provides a range measurement. By analyzing the variation of signal timing from a few positions on the Moon, the range, orbit, libration, and other quantities (such as tests of general relativity) are determined. A well-known aspect of lunar reflectors is that the return signal strength goes as $1/r^4$, not $1/r^2$, due to the fixed area of the reflector surface. This $1/r^4$ dependence prevents passive reflectors on Mars from being of any practical use.

By contrast, if the incoming light pulse from Earth were detected and responded to by an *in-situ* laser that was directed towards the Earth, the return signal strength would go as $1/r^2$. Such *laser transponders* (Merkowitz et al., 2006[51]) would enable the same quality of ranging data from Mars that we currently achieve with the Moon, with spectacular science return. Although the mass and power requirements of such transponders have not yet been firmly established, they are expected to be "suitcase" sized.

#### *2.7.2.2 Martian laser ranging science*

The manifold science returns of installing laser transponders on Mars would include:

- Improved Mars ranging from current meter-level accuracy to centimeter-level accuracy, with dramatic improvement in the ephemeris of Mars (important for interplanetary navigation); specifically, the ephemeris of Mars is now known to within meters in the plane, but only to hundreds of meters off the plane. Out-of-plane accuracy would improve by one or more orders of magnitude with laser ranging
- Improved mass of Jupiter by virtue of its measured effect on the Sun-Earth-Mars system.
- Measurement of the warping of spacetime by the Sun as Mars goes into superior conjunction, via the "Shapiro time delay" test; specifically, the $\gamma$ parameter, which is a measure of the curvature of space by a massive body, is given by GR to be $\gamma \equiv 1$. The Cassini mission gave a limit to deviations from unity at the level of $2 \times 10^{-5}$, while martian laser transponders would yield an accuracy roughly 10 times better, at approximately $10^{-6}$. This would be the strongest test to date of Einstein's GR.
- A test of the Strong Equivalence Principle (SEP), which is the equivalence of the gravitational mass of a body with its inertial mass) of GR. Violations of SEP by the sun and planetary bodies result in measurable deviations from Keplerian planetary motion. This has already been searched for with lunar ranging via the "polarization effect" (Anderson et al., 1995[52]) in the Earth-Moon-Sun system. This polarization effect, however, is approximately 100 larger for Earth-Mars orbits. Should ranging to Mars be enabled at centimeter-level accuracy, our precision in testing Einstein would be improved by two orders of magnitude. (In particular, this would measure the nonlinearity of self-interactions of gravity.)
- Centimeter-accuracy ranging to Mars, in concert with ranging data for the Moon, would allow better limits on the time variation of the gravitational constant.
- With transponders placed at several (>3) locations on Mars, the planetary rotational dynamics can be measured, yielding estimates of the core size of the planet. These would also enable measurement of Mars' tidal Love number (a measure of the elastic properties of the martian interior).

[51]Merkowitz, S.M., et al. (2006) International Journal of Modern Physics D.
[52]Anderson, J.D. et al., astro-ph/9510157 (1995).

While it is unknown whether this kind of laser transponder science could be done robotically, it certainly could and should be done on any human mission to Mars. In addition, the emplacement of transponders at multiple locations on the planet (e.g., at the same multiple sites that are advocated for planetary science) would greatly improve the science return. Astrophysical science would be best served by planning for a "suitcase-sized" astrophysics package on all three of the initial human missions that are envisioned in the reference architecture, with the capability to emplace the packages at geographically dispersed sites that are separated by large distances (i.e., 1,000s as opposed to 100s of kilometers apart).

# 3 ARCHITECTURAL ASSESSMENTS

One primary focus of the Design Reference Architecture 5.0 development was the identification and systematic assessment of principal key challenges that are associated with human exploration of Mars. A top-down systems engineering approach was established to identify, assess, and systematically eliminate unattractive options from further consideration. This process was facilitated by the development of an architecture trade tree (figure 3-1). This trade tree provides a graphical representation of key technical linkages and architectural challenges associated with future human exploration missions to Mars. The trade tree was established to place those key decisions or "architectural branches" that have the most overall leverage on the resulting architecture as high in the trade tree as possible. Providing a structured approach allowed the study team to systematically eliminate complete branches, thus placing effort on those branches that provided the best balance of the key figures of merit: safety, cost, and performance. The architecture trade tree was a very effective tool that allows the team to strategically address the overall architectural approaches, and to concentrate on those that provided the highest overall architectural leverage early in the study. The overall study approach was structured to begin with this high-level architectural "trade tree trimming" followed by a series of architecture refinement activities with the purpose of better optimizing the overall architectural approach.

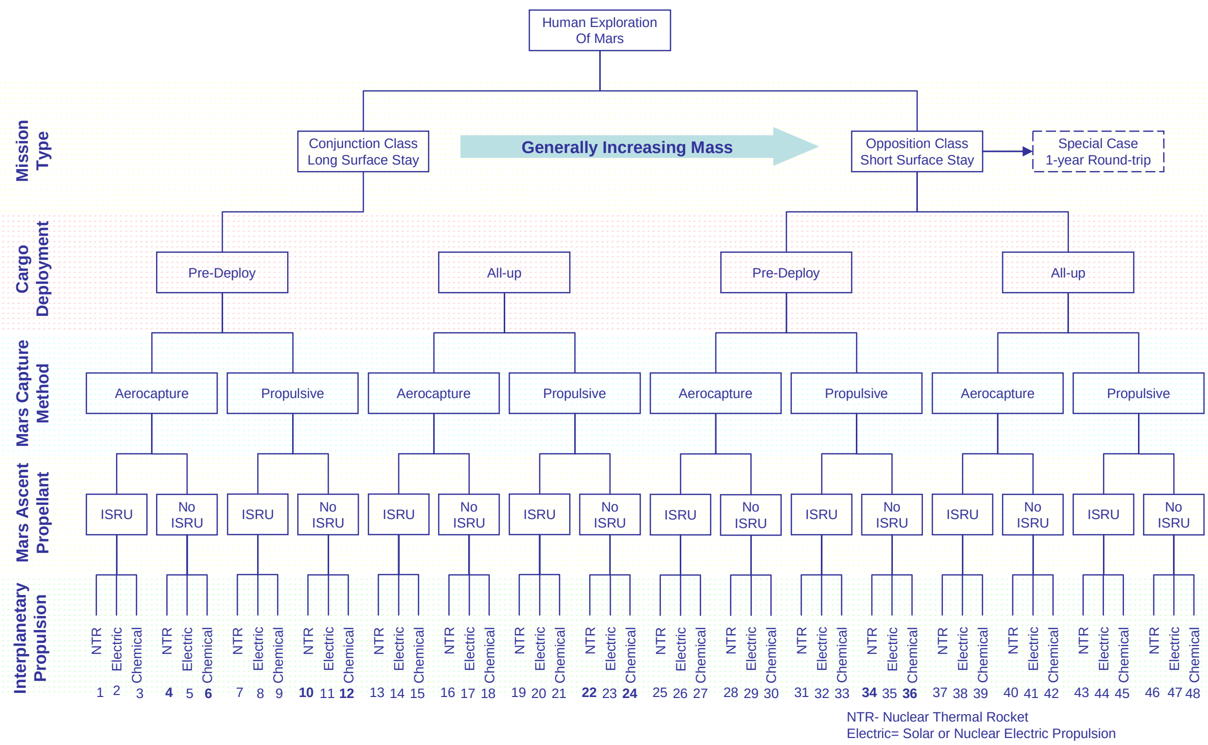



**Figure 3-1.** Design Reference Architecture 5.0 trade tree.

The emphasis of the first phase of the study activity focused on trimming the trade tree by developing specific decision packages associated with each key architectural branch of the trade tree. Each decision package used a common set of integrated performance tools that included an estimate of the overall architecture performance, risk, and cost. In addition, each decision package was formulated around a common set of FOMs with common key measures of effectiveness. Each decision package was then reviewed by the Joint Steering Group for concurrence on the results of the assessment. This iterative approach allowed an appropriate hierarchy of decisions to be addressed in a very systematic manner. Since the emphasis of this initial phase of assessments was on the key decision points of the trade tree, emphasis was placed on the relative comparison of the architectural approach associated with that specific decision comparison. That is, emphasis was placed on ensuring that a proper relative comparison between the two branches was achieved as opposed to optimization of a specific branch. Emphasizing the relative architectural comparisons allowed the study team to develop rapid, high-level comparative models rather than spending too much time refining specific design details. Optimization was reserved for the second phase of the study within a narrower set of architectural options or branches. Emphasis during the first study phase was placed on establishing the proper level of details that was associated with the decision at hand to ensure that important design or operational details were not overlooked that could sway the decision in a different direction. To aid in this process, previous models and design details that were developed by various subject matter experts who participated in the many previous human exploration of Mars efforts were used to the greatest extent possible. Throughout this process, emphasis was placed on consistency and commonality of all ground rules, assumptions, and modeling approaches to ensure that proper relative comparisons were being made.

During the development of Design Reference Architecture 5.0, emphasis was equally placed on assessing overall architectural risk and cost as well as performance. Integrated risk and cost models were developed based on the technical details that were developed by the various subject matter experts. These risk and cost concepts were then combined into an overall mission model for assessment of the overall architectural risk and cost. Assessment of the resulting integrated model allowed for identification of the key cost and risk drivers that were associated with each represented branch of the architecture trade tree. An overview of this iterative design and decision process is provided in figure 3-2.

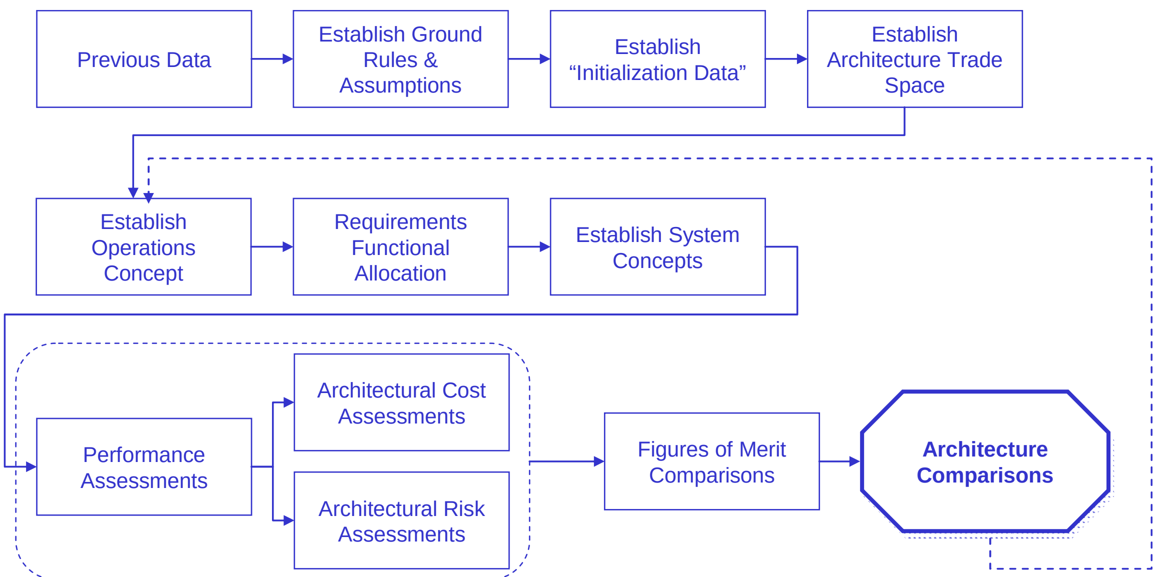



**Figure 3-2.** Architecture definition evaluation process.

## *3.1 Figures of Merit, Ground Rules, and Assumptions*

### 3.1.1 Figures of merit

During the study, key FOMs were used to help the analysis team develop an understanding of the implications of the various decisions under consideration. FOMs were used to measure the benefit of one approach as compared to other alternatives. Using standard categories, a consistent set of measures makes it possible to compare alternatives in addition to providing insight into the performance sensitivities of the alternatives and variations due to different assumptions and inputs. Specific measures of effectiveness associated with each FOM were established based on the specific decision that was on hand. The MAWG used the following FOMs in the development of the various Decision Packages that were under consideration:

*Safety and Mission Success*: Measures of effectiveness that are associated with safety and mission success focus on determining the degree to which a mission concept or technology option ensures safety and reliability for all mission phases. To be sustainable, future space exploration systems and infrastructure, and the missions that are pursued when using them, must be reliable and, when astronauts are involved, be as safe as reasonably achievable. Emphasis is placed on understanding the comparative values of safety-related measures of performance discussed below:

- *Risks* – An assessment of the events that could result in loss of crew, loss of vehicle, and mission failure. These could include launch failure or failure during other mission events. The confidence levels of known and unknown aspects of the mission concept or technology choices should be addressed. Key FOMs for the risk category included crew safety (probability of loss of crew) and mission success (probability of loss of mission). The risk models that were developed to assess the risks included all redundancy, reliability, and contingencies as known about the systems to date. These risk estimates will improve as the design maturity of the systems improves.

- *Hazards* – An assessment of the mission and technology risks that have the potential to cause a mishap. This includes hardware, software, and operational issues that could result in loss of crew, personnel, vehicle, or mission. Hazard measures of effectiveness include crew radiation exposure, trajectory hazards such as close passage to the sun, etc.

- *Aborts* – An assessment of the ability of the mission concept or technology choice to provide for the survival of the crew during various mission phases due to anomalies that result in early mission termination. Aborts could include early vehicle return or safe havens, but must result in the eventual safe return of the crew to Earth. For the most part, aborts were considered as part of the overall risk measure, such as the ability for the crew to return to orbit due to systems failures on the surface of Mars.

- *Development* – A key FOM for the Mars architecture is development risk. There is development risk for new technologies. There is also development risk associated with the design and testing of hardware and software, beyond just the risk of successfully developing new technologies. This includes not only the flight elements, but the fabrication, test, and operations facilities that are needed to support missions. Some factors in development risk are complexity, maturity of the technology, performance margins, manufacturability, and schedule. There are also risk factors that are not directly technical, such as public approval of any nuclear technologies that are used, acquiring existing facilities, environmental approval for new facilities or modifications, planetary protection issues for Mars and Earth, potential international cooperation issues, not being able to deliver some products for the cost estimates that were committed to, and variability in the funding environment.

*Effectiveness*: Measures of performance that are associated with effectiveness focus on determining the degree to which the mission concept, or technology option, effectively meets mission needs. Future space exploration systems and missions must be effective. In other words, the capabilities of a new system or infrastructure must be worth the costs of developing, building, and owning them. The goals and objectives that are achieved by the missions using those systems and infrastructures must be worth the costs and risks that are involved in operating them. Effectiveness must be determined case-by-case, based on the specific design objectives of the system or infrastructure, and on the detailed mission objectives (e.g., science objectives) that may be achieved.

- *Mission objectives* – Assessment of capability of the mission approach or technology choice to satisfy exploration objectives, including the ability to meet science objectives and flexibility in mission planning and execution. This FOM includes items such as number of launches, spacing between launches, time available to support key operations, etc.

- *Mass* – Total mass required to be delivered to LEO to support initial mission (includes pre-deployed infrastructure, if any) and the required mass for each subsequent mission. Also includes an assessment of the total number of launches that are required to emplace the necessary infrastructure as well as for each recurring mission. Mass measures of effectiveness also include architecture sensitivity to change in mass.

*Affordability*: To be sustainable, future space exploration systems and infrastructures, and the missions that are pursued using them, must be affordable. In other words, the costs for design, development, test, and engineering for the systems must be consistent with projected future-year NASA budgets. (The same is true for the recurring costs of additional copies of all exploration systems). Similarly, the costs that are associated with operating these systems in future space exploration missions must be consistent with projected future-year NASA budgets. Assessments of affordability include the degree in which the proposed mission or technology option is expected to provide an affordable approach. Assessments in this focus area include both total expected costs as well as affordability assessments regarding expected funding profiles and phasing.

- *First mission* – Total cost for the design, development, test, and evaluation of the required systems and facilities that constitute the element or mission concept for the first human mission. This includes all necessary flights, cargo, and crew that are necessary to conduct the mission. First-mission cost includes total program, infrastructure, and facility costs that are necessary for execution of the mission concept (e.g., sustaining engineering, hardware production, ground and mission operations, etc.).

- *Third mission* – Total annual program, infrastructure, recurring element, and facility costs that are necessary for execution of three complete human missions to Mars.

### 3.1.2 Ground rules and assumptions

The following GR&As were used as top-level guidance for the study. These were derived based on management guidance, internal and external constraints, design practices, and existing requirements. Items that are Ground Rules are denoted with “GR” and were considered to be held constant for this study. Assumptions, which are denoted with “A,” were assumed parameters for initializing the study and may be traded if sufficient time and resources are available during the study. Table 3-1 provides the safety and mission assurance GR&As, table 3-2 contains the operations and general GR&As, table 3-3 lists the technical GR&As, table 3-4 itemizes the cost and schedule GR&As, and table 3-5 presents the testing and verification GR&As.

**Table 3-1.** Safety and Mission Assurance Ground Rules and Assumptions

| **Safety and Mission Assurance Ground Rules and Assumptions** | | | |
|---|---|---|---|
| **No** | **Type** | **Description** | **Rationale** |
| 001 | GR | NASA Procedural Requirements (NPR) 8705.2, Human-Rating Requirements for Space Systems, will be used as a guideline for all architecture design activities. Required deviations from NPR 8705.2 will be noted in the applicable requirements documentation. | Various levels of in-flight maintenance and repair will be studied to determine the proper balance of risk, cost, and overall system performance. |
| 002 | GR | Abort opportunities will be provided throughout all mission phases to the maximum extent possible. | The feasibility of aborts during various mission phases will be studied as time allows. Strategies such as "remain in orbit" or "abort to surface" may be used as opposed to the more traditional “abort to Earth.” |
| 003 | GR | Planetary Protection requirements shall be considered and implemented during all architecture design activities, as required by the Outer Space Treaty of 1967 as well as by applicable NASA and Committee on Space Research (COSPAR) documents. Protection of the Earth from harmful contamination shall be assured absolutely. | Implementation guidelines addressing Planetary Protection requirements for human missions to Mars are currently under development by COSPAR and NASA. A draft set of guidelines, representing current international consensus on Planetary Protection implementation for human missions to Mars, has been made available by the Planetary Protection Officer (cassie.conley@nasa.gov). A NASA Procedural Requirements document is in preparation. |

**Table 3-2.** Operations and General Ground Rules and Assumptions

| Operations and General Mission Related Ground Rules and Assumptions | | | |
|---|---|---|---|
| **No** | **Type** | **Description** | **Rationale** |
| 103 | GR | Previous studies and planning activities will form the foundation for this study and will be used to the greatest extent possible. | Previous studies as well as updates that were made since the last DRM will be used after proper adjustments to account for differing GR&A and study requirements. In addition, the current Constellation elements (Areas and Orion) will be incorporated. |
| 104 | A | The architecture will allow for pre-deployment of cargo to Mars orbit and Mars surface. | Pre-deployment of mission cargo in advance of the crew mission is one way of reducing overall mission mass, but this must be weighed in context of the overall cost and risk of the architecture. |
| 105 | GR | The architecture will support any mission opportunity to Mars. | The propulsive energy requirements vary from opportunity to opportunity. Since it is not clear when the first human mission to Mars will be conducted, it is important to protect for all opportunities across the synodic cycle. |
| 106 | A | Architectures will be designed to minimize the length of time that the crew is continuously exposed to the interplanetary space environment. | Minimizing the exposure of the crew to the zero-g and radiation environment is a key element of risk reduction. |
| 107 | GR | In-space EVA assembly will not be required. | Crew risk is minimized by reducing complex assembly operations in space. This ground rule is not intended to restrict operations such as automated rendezvous and docking of one or more large elements in LEO, but rather limit ISS-type assembly operations. |
| 108 | GR | In-space EVA will only be performed as a contingency operation. | Contingency EVA is preserved for risk mitigation. |
| 109 | A | Campaign assessments will include a minimum of three consecutive missions to Mars. | The initial investment to send a human crew to Mars is sufficient to warrant frequent mission opportunities. This assumption is being used as an initial variable that will be traded within anticipated budgets. Timing of the missions as well as spacing between missions will be studied. |
| 110 | A | The crew size for each human mission to Mars will be six. | Consistent with the result from previous crew and skill mix studies. Assumption will be traded as time allows. |
| 111 | GR | The CEV will be used to deliver crew to the Mars transfer vehicle (MTV) in Earth orbit. | Consistent with the Vision for Space Exploration and a key element in cost reduction. |
| 112 | GR | A block upgrade of the CEV will be used to return the crew from the Mars-Earth return trajectory. | Consistent with the Vision for Space Exploration and a key element in cost reduction. Capturing the MTV back into Earth orbit has been shown to be prohibitively expensive. |
| 113 | GR | Launch operations will be performed at the NASA Kennedy Space Center (KSC) through clearing of the launch pad structure. | Consistent with the Vision for Space Exploration and current Constellation Program (CxP) planning. |
| 114 | GR | On-orbit flight operations and in-flight operations for crewed missions will be performed at NASA Johnson Space Center (JSC). | Consistent with the Vision for Space Exploration and current CxP planning. |
| 115 | GR | Crew and cargo recovery operations from the crew and cargo launches will be managed by KSC with assistance from other NASA and non-NASA personnel and assets as required. | Consistent with the Vision for Space Exploration and current CxP planning. |
| 116 | GR | Rely on the advances of automation and robotics to perform a significant amount of routine activities throughout the mission. | Crew workload can be optimized by providing a proper mix between humans and robots. |

**Table 3-3.** Technical Ground Rules and Assumptions

| General Technical Ground Rules and Assumptions | | | |
|---|---|---|---|
| **No** | **Type** | **Description** | **Rationale** |
| 203 | GR | Earth return trajectories will be limited to Earth entry speeds of a maximum of 13.5 km/s. | Consistent with previous studies that balance return time of flight, entry corridor width, and crew g-loads. |
| 205 | GR | Zero percent dry weight contingency for existing vehicle elements with no planned specification change and no anticipated modifications (this includes current Constellation elements, which already have appropriate contingencies applied to the current best estimate) | Consistent with ESAS and best practice. |
| 206 | GR | Five percent dry weight contingency on existing systems requiring minimal modifications | Consistent with ESAS and best practice. |
| 207 | GR | Ten percent dry weight contingency on new elements with direct heritage | Consistent with ESAS and best practice. |
| 208 | GR | Thirty percent dry weight contingency on new in-space elements with no heritage | Consistent with ESAS and best practice. |
| 209 | GR | Thirty percent margin for average power | Consistent with ESAS and best practice. |
| 210 | GR | Two percent margin for reserves and residuals mass | Consistent with ESAS and best practice. |
| 211 | GR | Two percent propellant tank ullage fractions for lunar vehicle (LV) stages | Consistent with ESAS and best practice. |
| 212 | GR | A structural 2.0 factor of safety for crew cabins | Consistent with ESAS and best practice. |
| 213 | GR | A 1.5 factor of safety on burst pressure for fluid pressure vessels | Consistent with ESAS and best practice. |
| 214 | GR | A 1.4 ultimate factor of safety on all new or redesigned structures | Consistent with ESAS and best practice. |
| 215 | GR | A 1.25 factor of safety on proof pressure for fluid pressure vessels | Consistent with ESAS and best practice. |
| 216 | GR | Ten percent margin for rendezvous delta-Vs | Consistent with ESAS and best practice. |
| 217 | GR | One percent ascent delta-V margin on launch vehicle and ascent stage to account for dispersions | Consistent with ESAS and best practice. |
| 219 | GR | Five percent additional payload margin on CaLV delivery predictions to account for airborne support equipment (ASE). | Consistent with ESAS and best practice. |
| 220 | GR | Technologies will be Technology Readiness Level-6 (TRL-6) or better by Preliminary Design Review (PDR). | Consistent with ESAS and best practice. |
| 221 | GR | Twenty percent launch vehicle payload delivery margin | Provides planning margin to account for the difference between payload current best estimates and launch vehicle minimum guaranteed performance. |

**Table 3-4.** Cost and Schedule Ground Rules and Assumptions

| Cost and Schedule Ground Rules and Assumptions | | | |
|---|---|---|---|
| **No** | **Type** | **Description** | **Rationale** |
| 303 | GR | All life cycle cost (LCC) estimates will include estimates of "full-cost" impacts. The Mars Architecture Team (MAT) cost lead will issue guidelines that will include full-cost categories and recommended percentages to ensure consistency in application across architecture elements. | Includes corporate General and Administrative (G&A) at 5%, center G&A, center Civil Service salaries, travel, overhead, and center service pool costs. |
| 304 | GR | Architecture elements will not include reserves. Reserves will be applied at the cost integration (architecture) level based on a strategy that complies with the Agency 70% confidence level policy. | Agency policy is to budget at the 70% confidence level. CxP has approval for the 70% to be applied to the program rather than project level. |
| 305 | GR | Cost of technical margins cited in Section 3 will be included in the cost estimates. | |
| 307 | A | There is a goal of performing the first human Mars landing by 2030, or as soon as practical | This is an assumption to initiate the study and will be a significant variable considered. |
| 308 | GR | PDR assumed to occur (TBD) years and Critical Design Review (CDR) assumed to occur (TBD) years prior to first launch. | The timing that is associated with each element is dependent on the complexity of the system involved. These dates will be addressed during Phase 2. |
| 310 | A | For the purpose of planning the scientific and risk-reduction objectives, the following minimum set of robotic missions is assumed to have been completed by 2030: Mars Phoenix, MSL, AFL, Upper Atmosphere Orbiter, MSR, and ML3N. | These missions are being assumed as a going-in position for the scientific community. The associated risk mitigation that each mission provides in the overall context of human exploration of Mars and associated budget will be addressed during the latter half of the study. |

**Table 3-5.** Testing and Verification Ground Rules and Assumptions

| Test and Verification Ground Rules and Assumptions | | | |
|---|---|---|---|
| **No** | **Type** | **Description** | **Rationale** |
| 403 | GR | Elements will have ground qualification tests to demonstrate readiness for human flight. | Consistent with ESAS and best practice. |
| 404 | GR | Multi-element integrated ground tests will be performed to demonstrate readiness for human flight. | Consistent with ESAS and best practice. |
| 405 | GR | Human flight elements require a minimum of one qualification flight demonstrating full functionality prior to crewed flights. | Consistent with ESAS and CxP accepted level of risk. |
| 407 | GR | Qualification of the Mars transfer stages for firing while mated to a crewed element requires a minimum of two flights to demonstrate full functionality prior to crewed flight. | Consistent with ESAS and CxP accepted level of risk. |

## *3.2 Initial Architecture Assessments: Trade Tree Trimming*

The initial activities of the MAWG focused on key architectural drivers for future human exploration of Mars. Emphasis was placed on providing a systematic top-down systems engineering process whereby architectural options that provide the greatest leverage in terms of satisfying key FOMs are addressed first, with subsequently lower priority trades and decisions conducted later. This top-down process allowed the study team to trim Unsatisfactory options from the trade tree early in the initial phase of the study. Due to the limited scope and time allocated for the study, not all potential options were considered. For the DRA 5.0 activity, emphasis was placed on understanding the advantages and disadvantages of the following architectural options:

1. Mission Type: Long-stay (Conjunction Class) vs. short-stay (Opposition Class) missions
2. Mars Cargo Deployment: All-up vs. pre-deploy

3. Mars Orbit Capture Method: Aero vs. propulsive capture
4. Mars Ascent Propellant: In-situ resources?
5. Mars Surface Power: Solar vs. nuclear

## *3.3 Mission Type*

The choice of the overall exploration mission sequence and corresponding trajectory strategy has perhaps the greatest single influence on the resulting architecture. The ideal mission would be one that: (1) provides the shortest overall mission to reduce the associated human health and reliability risks, (2) provides adequate time on the surface to maximize the return, and (3) provides low mission mass that, in turn, reduces overall cost and mission complexity. Unfortunately the "ideal" mission does not exist, and tough choices must be made between design options. Thus, the first decision that was dealt with by the MAWG addressed a key architectural component that is tied to the orbital mechanics of human exploration missions, namely long surface stays vs. short surface stays. Human missions to Mars are classified into these two primary approaches as governed by orbital mechanics that are described in this package.

The mission type trade tree is shown in figure 3-3. The branches of the trade tree that were considered are those numbered 10, 12, 34, and 36 in the trade tree. These branches were chosen because experience has shown that the cases that were chosen represent typical approaches, and the trends will be similar for the other branches of the trade tree.

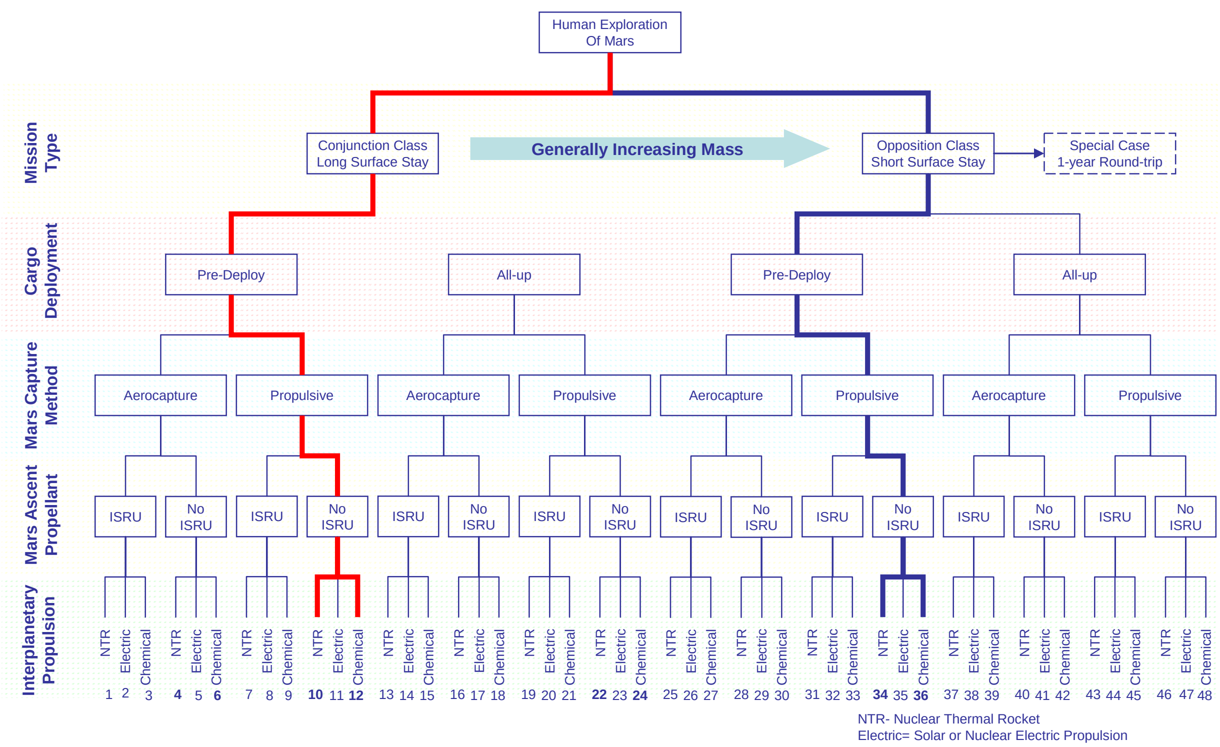



**Figure 3-3.** Tree with emphasis on mission type branches.

### 3.3.1 Trajectory options

Trajectories from Earth to Mars are well understood and have been used by NASA for over 4 decades. Round-trip missions to Mars and back, however, are more complex in that the outbound and inbound legs must be synchronized into an optimal mission plan. For the lower-energy outbound trajectories, upon arrival at Mars the Earth is in a relatively unfavorable alignment (phase angle) for an energy-efficient return. This unfavorable alignment results in two distinct classes of round-trip Mars missions: Opposition Class missions, which are also referred to as short-stay missions, and Conjunction Class missions, which are also referred to as long-stay missions. Practical considerations – e.g., total propulsive requirements, mission duration, surface objectives, and human health considerations – must be considered in the mission design process when choosing between these mission classes. The period of time necessary for the phase angle between Earth and Mars to repeat itself varies. The mission repetition rate for identical Earth-Mars phasing and, therefore, launch opportunities for similar mission classes is on the order of every 26 months. The mission characteristics such as mission duration, trip times, and propulsive requirements vary to due to the eccentricity of Mars' orbit.

Opposition Class missions are typified by short surface stay times (typically 30 to 90 days) and relatively short round-trip mission times (400 to 650 days). The exploration community has adopted the terminology "short-stay" missions for this class. The trajectory profile for a typical short-stay mission is shown in figure 3-4. This mission class has higher propulsive requirements than the long-stay missions, and often uses a gravity-assisted swing-by at Venus or performs a deep-space propulsive maneuver to reduce total mission energy and constrain Mars and Earth entry speeds. Short-stay missions always have one short transit leg, either outbound or inbound, and one long transit leg, which requires a close passage by the sun (0.7 astronomical unit (AU) or less). After arrival at Mars, rather than waiting for a near-optimum return alignment, the spacecraft initiates the return after a brief stay, and the return leg cuts well inside the orbit of the Earth to make up for the "negative" alignment of the planets that existed at Mars departure. Distinguishing characteristics of the short-stay mission are: (1) short-stay at Mars, (2) medium total mission duration, (3) the vast majority of the round-trip time is spent in interplanetary space, 4) perihelion passage inside the orbit of Venus on either the outbound or inbound legs, and (5) large total energy (propulsion) requirements.

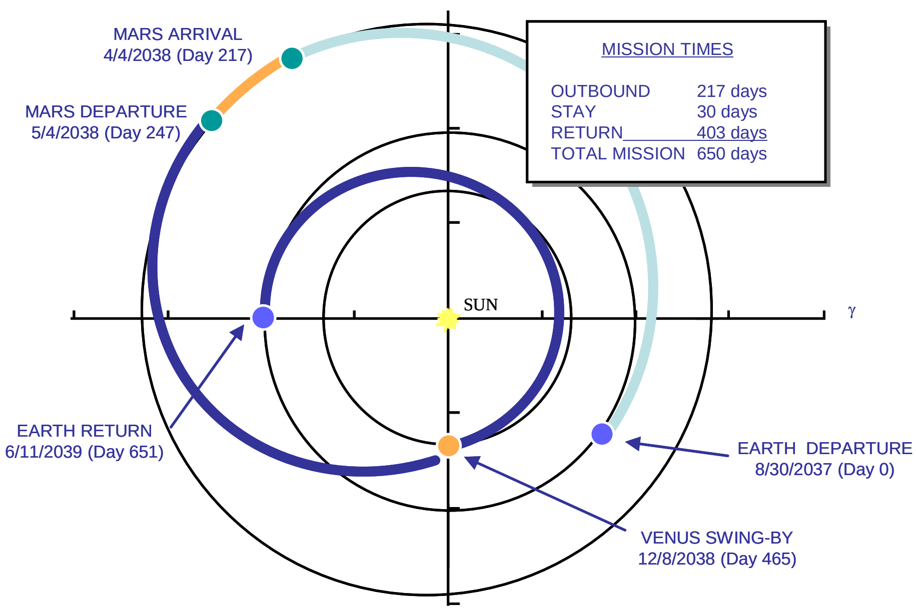



**Figure 3-4.** Typical Opposition Class short-stay mission.

The close perihelion passage for the short-stay missions presents risks in thermal design, radiation exposure, and crew safety. The thermal design would most likely require one side of the vehicle to be completely shaded by a large deployable ceramic fabric sunshade. In addition, deployable radiators and active cooling loops may be required. Off-nominal pointing, which placed the vehicle in direct solar illumination, could probably not be survived for more than a few minutes. For a solar-powered spacecraft, the arrays would have to be articulated to maintain an exact angle to the sun to keep the arrays from being destroyed and to maintain the needed power output. Off-nominal pointing could quickly result in destruction of the arrays.

Additional shielding mass would be required for close perihelion passage to protect from solar flares, especially during the solar maximum periods. Since the strength of the radiation dose is inversely proportional to roughly the square of the distance ($1/R^{2.5}$), close perihelion passage can have a profound affect on the radiation shielding (solar storm) and radiation dosage to the crew. Unpredictable solar flares during a close perihelion passage present a greater risk than for a mission that stayed outside of 1 AU.

Conjunction Class missions are typified by long-duration surface stay times (500 days or more) and long total round-trip times (~900 days). This mission type has adopted the terminology "long-stay" mission. These missions represent the global minimum-energy solutions for a given launch opportunity. The trajectory profile for a typical long-stay mission is shown in figure 3-5. Unlike the short-stay mission approach, instead of departing Mars on a non-optimal return trajectory, time is spent at Mars waiting for more optimal alignment for lower energy return. Distinguishing characteristics of the long-stay mission include: (1) long total mission durations, (2) long stays at Mars, (3) relatively little energy change between opportunities, (4) bounding of both transfer arcs by the orbits of Earth and Mars (closest perihelion passage of 1 AU), and (5) relatively short transits to and from Mars (< 180–210 days).

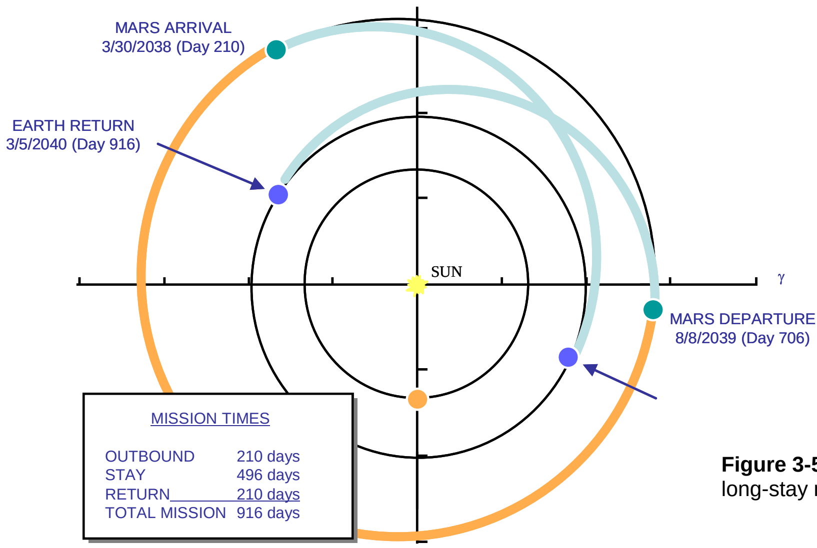



**Figure 3-5.** Typical Conjunction Class long-stay mission.

When considering "fast" Mars missions, it is important to specify whether the reference is to a fast round-trip or a fast transit mission. Past analyses have shown that decreasing round-trip mission times for the short-stay missions does not equate to fast transit times (i.e., less exposure to the zero-g and space radiation environment) as compared to long-stay missions. Indeed, fast transit times are available only for the long-stay missions. This point becomes clear when looking at figure 3-6, which graphically displays the transit times as a function of the total round-trip mission duration. Although the short-stay mission has approximately half the total duration of either of the long-stay missions, over 90% of the time is spent in transit as compared to 30% for the fast-transit mission.

The risk to crews on fast transit missions may be even less than the risk to crews on short-stay missions, not only because of minimized exposure to GCR but also reduced probability of exposure to solar proton events (SPEs) (flares) in interplanetary space. A similar analysis of mission classes is involved in considering the crew's exposure to the zero-g environment during transits to and from Mars. The martian surface stay will have reduced dosage relatively to an equivalent period in interplanetary space, due to the $2\pi$ protection that is afforded by the planet and the thin martian atmosphere, which can provide 10–20 $cm^2$ Al-equivalent depending on latitude and season.

Upon arrival on the martian surface, the crew will need to spend some currently unknown, but probably short, time re-adapting to a partial-g field. This may be of concern for the short-stay missions where a substantial portion of the surface stay time could be consumed by crew adaptation to martian gravity. Conversely, ample time will be available for the crew to regain stamina and productivity during the long surface stays that are associated with the minimum-energy, faster-transit missions.

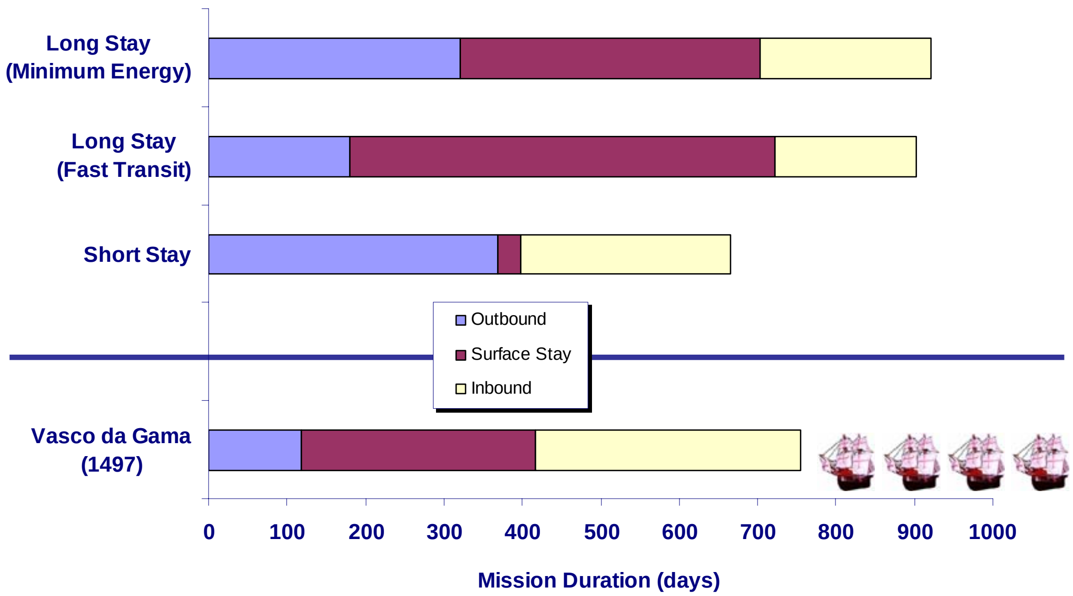



**Figure 3-6.** Comparison of Mars mission transit times.

There are other factors that require consideration in choosing a particular trajectory, launch date, and mission strategy. These include seasonal dust storms and solar conjunctions and oppositions with the Earth that affect the communication links between the mission elements and the ground system. Figure 3-7 shows a timeline for some of the mission opportunities and how critical mission phases line up with some of these environmental effects.

### 3.3.2 Mission design strategy

Given a set of trajectories, there are key mission design parameters that must be considered to optimally address meeting the Science goals and objectives, minimizing risk and cost, and being implemented robustly on a timetable that is consistent with the Vision for Space Exploration. Consideration must also be given to the number of launches required, the timetable for the launches, and the ability of the ground operations infrastructure to deliver those launches in a credible and cost effective manner.

### 3.3.3 All-up vs. pre-deploy

Under nominal conditions, not all mission assets are used by the crew during the outbound phase of a mission. Examples of these include all of the systems that are used on the surface (including habitation), the vehicle that is used for entry and landing as well as launch from the surface, and any ISRU equipment (if used). This makes possible the strategy of sending these items on an earlier, typically more energy-efficient trajectory and, thus, the delivery of more of these assets (mass) for the same amount of propellant (as used by the crew) or of the nominal assets (mass) for less propellant and associated launch vehicles. This approach has become known as the “split” or “pre-deploy” mission approach.

For the short-stay mission sequence, the only cargo to be pre-deployed is the descent/ascent vehicle (DAV), which is sent to Mars on the first minimum-energy trajectory prior to the crew launch on an opposition trajectory. The DAV arrives at Mars before the crew launches from Earth, allowing time to confirm that it is in its proper orbit and functioning normally. The DAV is then placed into a minimal operating configuration and remains in this state more than 1 year before the arrival of the crew. While the first crew is in transit to Mars, the launch campaign for the second crew’s DAV begins. This DAV is in transit to Mars while the first crew carries out its Mars surface mission and begins a return to Earth. This second DAV arrives at Mars and is similarly positioned and checked prior to the departure of the second crew. This DAV waits in its orbit for approximately 2 years prior to the arrival of the second crew. This is a significantly longer wait than experienced by the first DAV, but the variability from mission to mission is typical of the short-stay mission opportunities. Each crew relies on its own DAV for completion of its mission. In addition, the

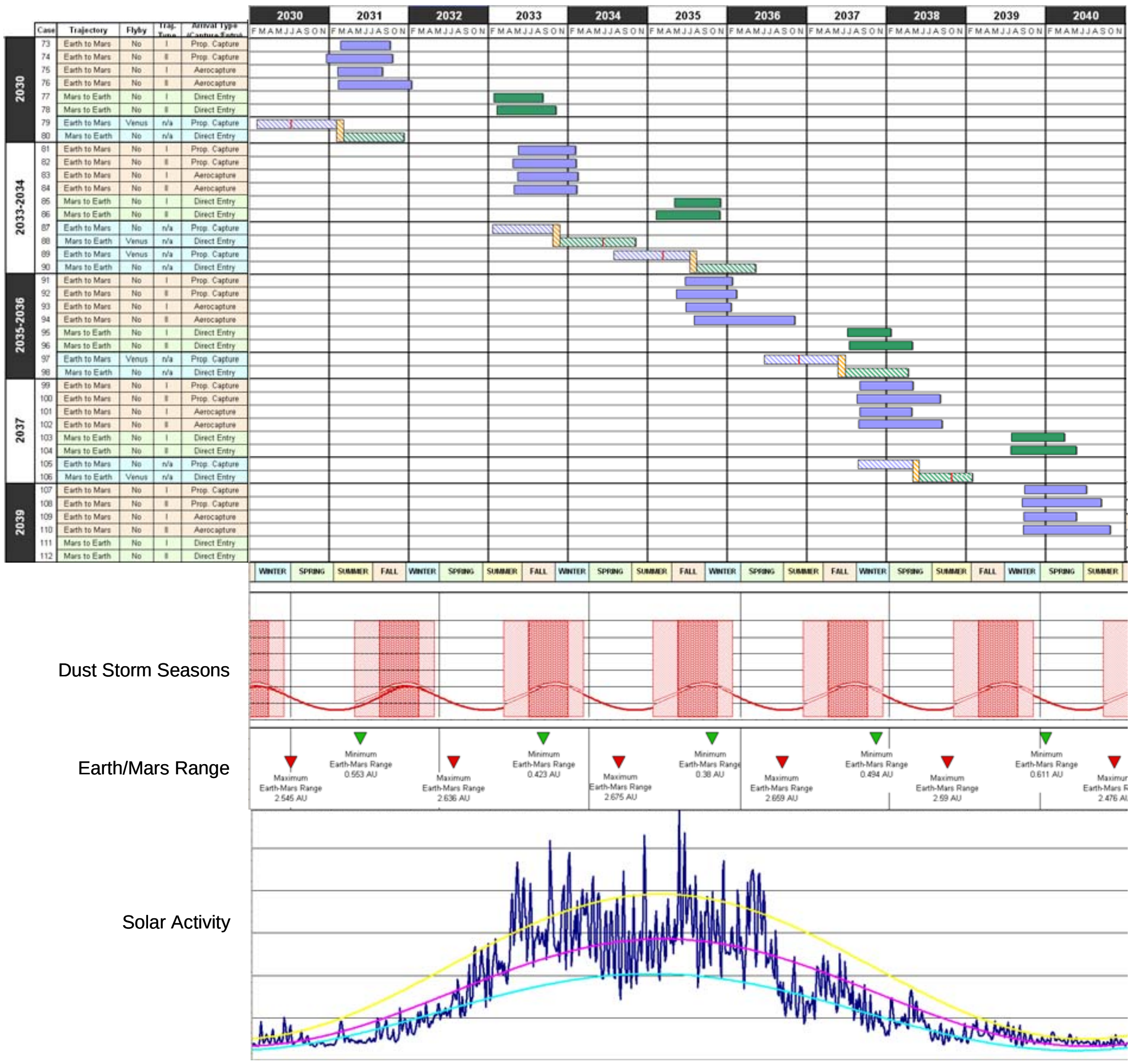



**Figure 3-7.** Mars mission opportunities timeline.

launch windows for this combined use of Conjunction Class and Opposition Class trajectories are such that there is no overlap in the launch campaign at KSC.

For the long-stay mission sequence, two cargo elements are pre-positioned to support the crew's surface mission: the DAV and a surface habitat (SHAB) with other surface equipment. Both of these elements are launched in the same minimum energy opportunity just over 2 years prior to the launch of the crew. The launch campaign for the first two cargo elements begins several months prior to the opening of the launch window. The cargo elements arrive at Mars approximately 8 months later and are placed into the appropriate orbit (for the DAV) or at the surface location (for the SHAB). They are checked for proper function before they are placed into a minimal operating configuration to remain in this state for over 2 years before the arrival of the crew. The next minimum-energy window (for the next cargo elements) opens shortly before the fast transit trajectory window for the first crew, but these launch windows are still close enough that a combined launch campaign at KSC is required. This launch campaign for the second crew's cargo and for the first crew begins as much as 1 year before either windows open so that all of these elements are ready for their respective departures. The first crew arrives before the cargo elements for the second mission and nominally uses the assets launched over 2 years previously. However, should either the DAV or the SHAB suffer a failure between the time the first crew launches from Earth and when it leaves Mars to return to Earth, the second set of cargo elements can be used, thus potentially preventing loss of the mission or of the crew. This is a unique feature of the pre-deployment strategy when applied to the long-stay missions. This overlap of assets is not available for any of the short-stay options or for the all-up strategy.

Figure 3-8 shows the implications of a pre-deployment strategy for both the short- and long-stay missions. A first human mission in the 2030 timeframe along with a subsequent human mission illustrates campaign-level implications of this strategy as well.

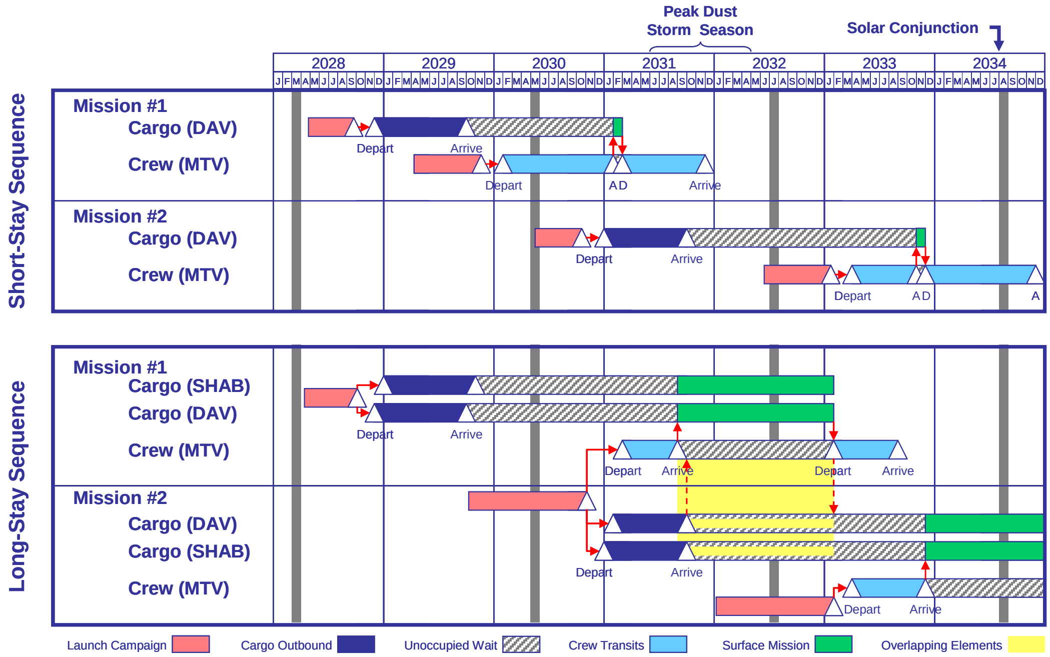



**Figure 3-8.** Pre-deployment timelines for short- and long-stay strategies.

The alternative to the pre-deploy strategy is the all-up strategy, whereby all of the required mission elements would be launched in the same Mars opportunity, although not necessarily on a single launch vehicle, which is effectively impossible, or as a single departure stack at trans-Mars injection (TMI), which would be unrealistically massive. For the short-stay mission sequence, the only cargo element that is required is the DAV. This element is launched on the same trajectory as the crew in its MTV. The combined launch campaign at KSC for both of these elements begins approximately 1 year prior to the launch window, based on the launch rate mentioned above. Both elements arrive at the same time and the nominal surface mission is carried out. KSC initiates the launch campaign for the second mission approximately 2 years after it completes the first campaign, although there is some variability in this interval due to the natural spacing between the short-stay trajectory opportunities.

For the long-stay, all-up mission sequence, two cargo elements are required to support the crew's surface mission: the DAV and an SHAB with other surface equipment. All of these elements are launched on a fast-transit trajectory so that they all arrive at Mars at the same time. While it is conceivable that all of these elements could be integrated into a single stack while in LEO, the total mass of such a stack would be quite significant (i.e., in some cases equivalent of several ISSs) and likely difficult to control. The total thrust that is required to avoid significant gravity losses during departure also makes this approach less desirable. The alternative – three closely spaced departures from LEO during the same launch window followed by a rendezvous (but not necessarily docking) in interplanetary space – is also not trivial but considered manageable and, thus, would be the preferred approach for this option. The KSC launch campaign begins approximately 1 year before these elements depart for Mars; this is similar to the situation described for the pre-deploy strategy. The launch campaign for the next mission begins approximately 1 year after completion of the first campaign. There is no overlap at Mars of the two crews or their equipment.

Figure 3-9 shows the implications of an all-up strategy for both the short- and long-stay missions. A first human mission in the 2030 timeframe along with the subsequent human missions illustrates campaign-level implications of this strategy as well.

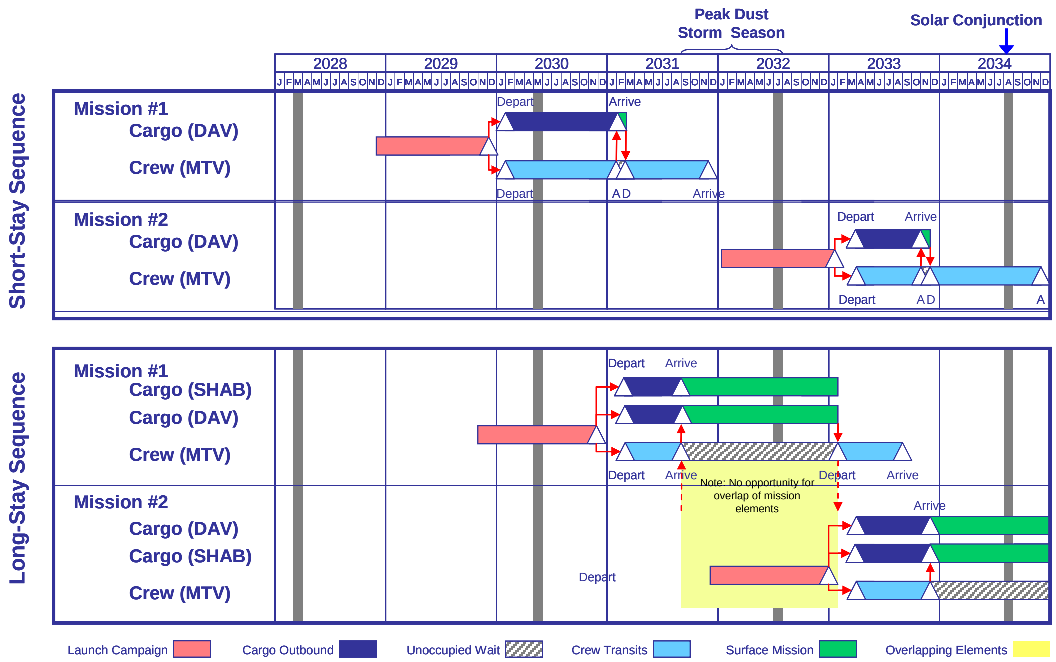



**Figure 3-9.** All-up mission timelines for short- and long-stay strategies.

### 3.3.4 Addressing key risks

Human health is a key mission design factor. The two biggest threats to human health on a Mars mission are radiation exposure and the long-term effects of zero g. Section 3.4 provides a detailed discussion of these issues. Mission design must address both of these concerns by minimizing the periods of continuous exposure to zero g and the total radiation exposure by the crew.

System reliability is a key risk due to the complexity of systems and the long lifetimes that are needed for human missions to Mars. While the long-stay missions require longer system lifetimes than the short-stay missions, this is somewhat offset by the system reliability challenges of close passage to the sun on short-stay missions. Pre-deploy missions require significantly longer operating lifetime for pre-deployed elements. This is offset by the redundancy that the pre-deployed elements afford in a sustaining program where humans are launched to Mars at every available opportunity along with the pre-deployed elements for the next opportunity. In this scenario, the redundancy in habitats and DAVs is provided.

### 3.3.5 Close perihelion passage considerations

For short-stay Opposition Class missions, mission timing can be generally be set up to use Venus during the outbound transit, inbound transit, and sometimes both to help shape the trajectory that is necessary for this class of mission. The Venus swing-by has the same result as a “free” deep-space maneuver and is, thus, more propulsively efficient. This requires that the mission sequence, timing, and relative phase angles between Earth and Mars be in specific relative geometry.

As can be seen from the plots in figure 3-10, the trajectories that are associated with Opposition Class missions, irrespective of the use of a Venus swing-by, require passage within the orbit of Venus. A representative (2037) Opposition Class mission is shown in the trajectory plot. In addition, as can be in the plots, the closest approach to the sun varies by mission opportunity and surface stay. For example, the 2037 Venus swing-by mission passes within 0.49 AU of the sun, spending 108 days within 0.8 AU.

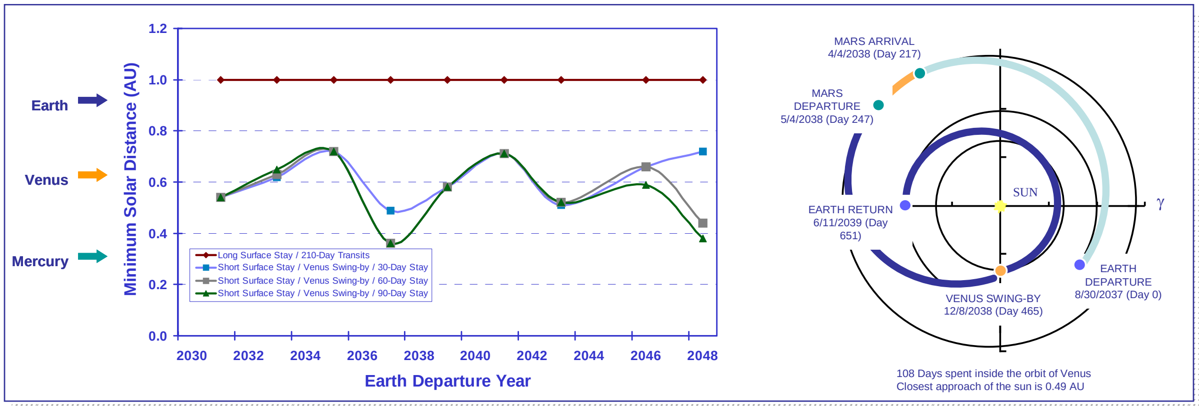



**Figure 3-10.** Venus swing-by perihelion passage.

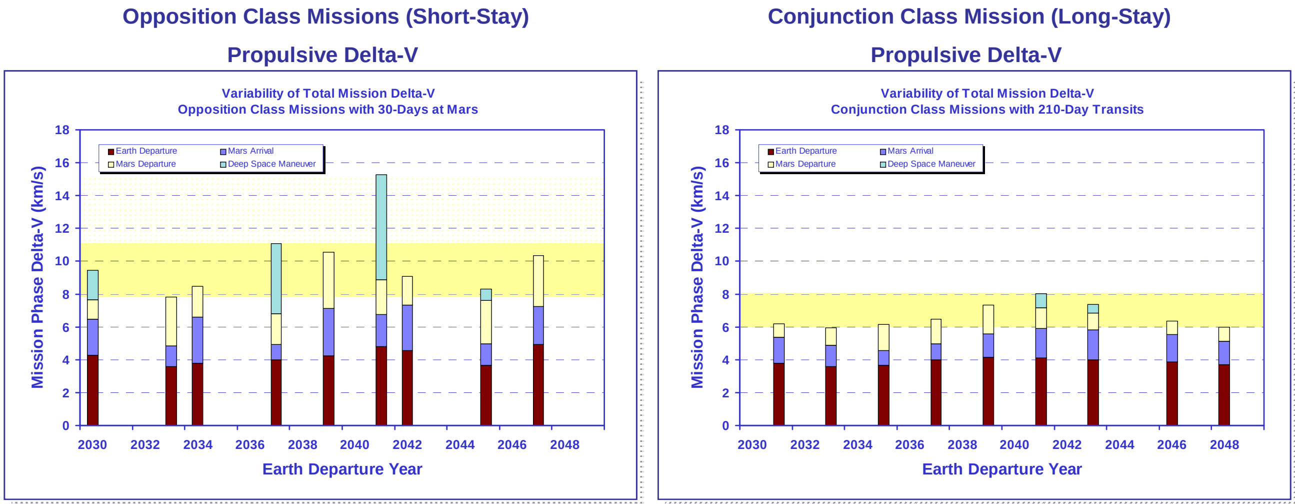
Passing within 1 AU of the sun poses some significant mission, vehicle design, and human health issues that must be adequately considered in the overall context of the mission approach. For instance, additional shielding mass is required to protect from solar flares during solar maximum. Since the strength of the radiation dose is roughly proportional to the square of the distance ($1/R^{2.5}$), close perihelion passage can have a profound effect on radiation dosage to the crew and subsequently to the radiation shielding required (may need additional parasitic shielding for protection against solar storms). Thermal control will be needed for both long- and short stay missions, but the heat load to the vehicle will increase with decreasing perihelion passage. Deployable sunshades are probably required for the short-stay missions to shadow critical vehicle components and areas. In addition, deployable radiators and additional active cooling loops may be required. Due to the increased thermal and solar influence, vehicle systems including solar arrays and sunshades must be positioned relative to the sun with tighter control to prevent overheating. Since Conjunction Class missions rely on favorable phasing between Earth and Mars, the trajectory does not require close perihelion passage; the vehicles thus remain at distances greater than 1 AU throughout the mission.

### 3.3.6 Total interplanetary propulsion requirement considerations

The variability of total interplanetary propulsive delta-V across the synodic cycle for both Opposition Class (short-stay) and Conjunction Class (long-stay) missions are provided in figure 3-11. As can be seen from this figure on the left, Opposition Class missions require greater total propulsive delta-V in addition to resulting in significant variation of propulsion requirements across synodic cycle. As can be seen from the left graph, the variation of delta-V across the synodic cycle is nearly 100% with an average total delta-V of 10 km/s ± 3.7 km/s. This variability significantly impacts the space vehicles since they must be designed to provide the propellant capability and design attributes that allow for a wide range of propellant loads or the capability to delivery a wide range of payloads to Mars.

It can also be seen that there are some mission cases where the total interplanetary delta-V is so excessive that they are outliers and, thus, usually eliminated from consideration. This is clearly evident in the 2041 mission opportunity, which is twice the magnitude of the best 2033 opportunity. Skipping mission opportunities results in a minimum of a 26-month “stand down” before resuming the normal mission sequence.

The variability of total interplanetary propulsive delta-V across the synodic cycle for Conjunction Class missions is provided in the right graph in figure 3-11. As can be seen in this graph, the total, as well as the variation from opportunity to opportunity is fairly small, on the order of 35%, while also providing for overall lower delta-V; the average total delta-V was approximately 7 km/s ± 1 km/s. This small variation of propulsive requirement across the synodic cycle allows the use of a common vehicle and payload design for each opportunity. This common strategy also allows the vehicle systems to be flown in any opportunity, thereby reducing the potential of either skipping harder years, as in the case of Opposition Class missions, or allowing systems to be flown at a later date if necessary due to technical or schedule difficulties.




**Note:** Optimized trajectories assuming 407 km circular LEO departure orbit, propulsive capture at Mars into a Mars 1-Sol orbit of 250 km x 33,793 km. 30 sols stat at Mars. Direct entry at Earth with an entry speed limit of 13 km/s.

**Note:** Optimized trajectories assuming 407 km circular LEO departure orbit, propulsive capture at Mars into a Mars 1-Sol orbit of 250 km x 33,793 km. 210 day transits to and from Mars. Direct entry at Earth with an entry speed limit of 13 km/s.

**Figure 3-11.** Total interplanetary propulsive requirement comparison.

The sensitivity of the total interplanetary propulsion requirements as a function of time spent in the vicinity of Mars for Opposition Class missions is shown in the left side of figure 3-12. As can be seen from this figure, the time that is spent in the vicinity of Mars has a profound affect on the total interplanetary delta-V. This increased delta-V translates directly to more IMLEO. It can also be seen that the sensitivity to stay time varies by mission opportunity ranging from a 15% variance in 2033 to 67% in 2047. Thus, to minimize the overall mission mass for Opposition Class missions, emphasis is placed on minimizing the amount of time spent at Mars that is counterproductive from a mission strategy point of view; reducing the time at Mars limits the mission objectives and goals that can be achieved. It should be noted that a vehicle that is designed for a 30-sol stay for a relatively hard opportunity, such as 2037, can extend the surface stay to 90 sols for easier opportunities, such as 2033. Extending the stay time beyond 90 sols becomes prohibitively expensive from a delta-V and mission mass perspective.

The sensitivity of total propulsive delta-V to the transit times to and from Mars for Conjunction Class missions is provided in the right graph. Minimum energy transfers occur with trip times in excess of 200 days where the savings of total delta-V are decreased as trip time is increased. Since it is important from a human health and performance perspective to reduce the transit times to the greatest extent possible, it can be seen that reductions in total trip time begin to become excessive with times less than 200 days and in some opportunities on the order of 180 days. The design team has chosen to establish the total delta-V capability of the interplanetary transportation system across all opportunities and then use that common system to shorten the trip times to the greatest extent possible.

### 3.3.7 Total mission duration considerations

The breakdown of trip times for the outbound, surface stay, and inbound portions of both the Opposition Class (short-stay) and Conjunction Class (long-stay) missions is provided on the left side and right side, respectively, of figure 3-12. Total mission durations for the short-stay missions range from 550–650 days with 30 sols in the vicinity of Mars. For the short-stay missions, more than 95% of the total mission time is spent in the deep-space zero-g interplanetary environment with the balance of 5% spent in the vicinity of Mars. Duration of the transit legs ranges from a minimum of 190 days to a maximum in excess of 400 days.

The corresponding trip time breakdown for the long-stay mission is provided in the left graph. The total mission durations range from 890 to 950 days with a range of corresponding surface stay times ranging from 475 to 540 sols in the vicinity of Mars. For long-stay missions, approximately 55% of the total mission duration is in the vicinity of Mars with the balance of 45% spent in transit. The time that is spent in orbit vs. the time that is spent on the surface of Mars is open to further refinement as the relative trade-offs between mission return and crew risk are conducted. The radiation dose will vary depending on the location of the mission within the 11-year solar cycle, which is shown on top of both sides of figure 3-13. Although SPE-associated exposure risk is higher during solar max (in red), the GCR dose is generally lower, due to the presence of more profound solar wind. The GCR dose will be correspondingly higher during solar minimum (in green).

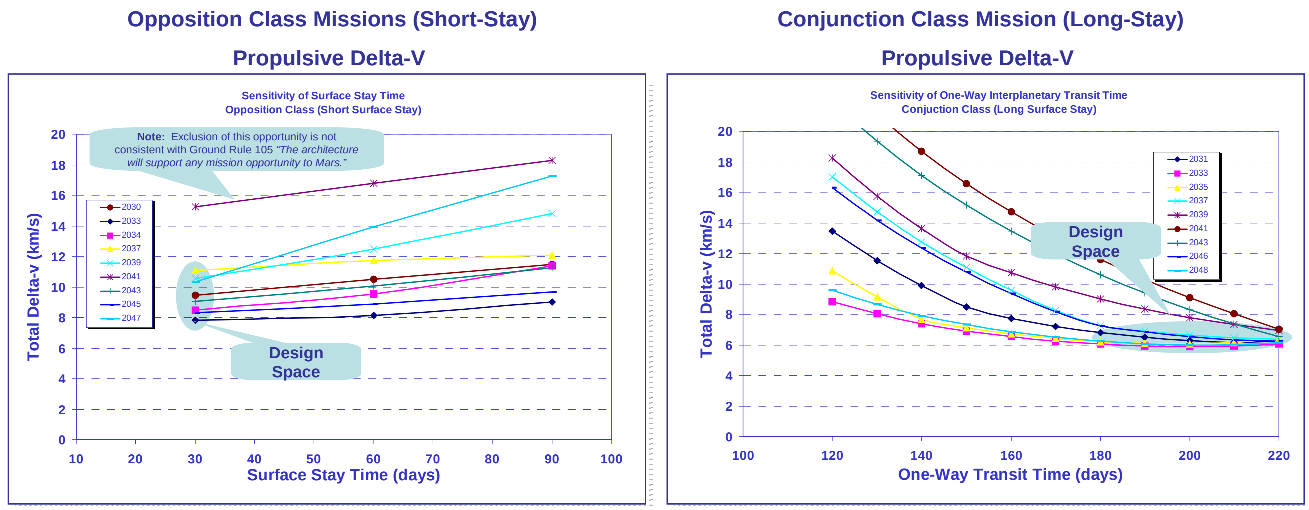



**Figure 3-12.** Total interplanetary propulsive sensitivity comparison.

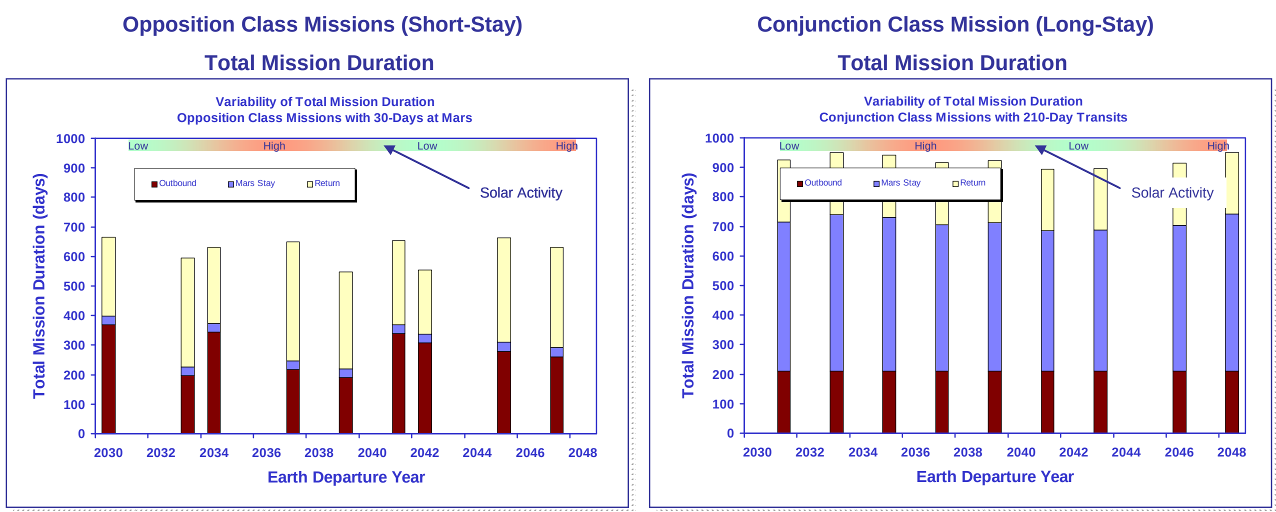



**Figure 3-13.** Total mission duration comparison.

### 3.3.8 Mars vicinitiy operational considerations

The operations that will be conducted in the vicinity of Mars are yet another important consideration when choosing between mission types. The complexity of operations, timing, and sequences as well as considerations that are associated with the health and performance of the crew must be included in the decision process.

#### *3.3.8.1 Mars capture and rendezvous*

Most mission strategies rely upon the pre-deployment of mission cargo to Mars orbit or the martian surface prior to arrival of the crew to reduce mission mass. Since the cargo elements are pre-deployed many months ahead of the crew, there is sufficient time to adjust the orbits prior to crew arrival to ensure optimal co-planar conditions. The crew vehicle will perform the orbital capture maneuver, capturing into a proper phasing orbit that is necessary for the subsequent rendezvous maneuver. Assuming that the cargo elements are placed in a 1-sol (250 km × 33,793 km) parking orbit, the phasing and rendezvous maneuver can take as little as 1 day, but could be longer if the relative phase between the target cargo vehicle and the crew vehicle is greatly out of phase after arrival in Mars orbit. Rendezvous and docking might also be delayed in the case of an off-nominal event.

#### *3.3.8.2 Landing*
After rendezvous with the lander, systems are checked out and verified operational, which is assumed to be at best 1 day. Additional time must be taken into account for any additional orbital loiter that is necessary for proper phasing with the landing site or to wait out Mars environmental factors such as dust storms.

#### *3.3.8.3 Crew acclimation*
After arrival, the crew transit vehicle systems are placed in a safe condition while the lander systems are transitioned to an operational condition and checked out. If artificial gravity (AG) transits to Mars are not used, the crew will be in a deconditioned state due to the lengthy 6- to 7-month zero-g transit from Earth to Mars. Present estimates for crew acclimation are on the order of 1 to 2 weeks based on current U.S. and Russian experience.

#### *3.3.8.4 Mars orbit departure*
During a short-surface duration, there will be very little apsidal and nodal regression. To meet the departure trajectory conditions, a multi-burn departure will be necessary to align with the departure asymptote. This multi-burn departure will require up to a few days, including a small departure window to account for contingencies.

#### *3.3.8.5 Mission type comparison*
Due to the short time spent in the vicinity of Mars for the Opposition Class (short-stay) missions and number of required operations, the short-stay mission will provide at best 1 to 2 weeks of surface exploration with 30 sols in the vicinity of Mars. With higher-performance propulsion systems (e.g., NTP), easier opportunities can extend the time at Mars up to 90 sols. A short-stay mission will require a scripted operational approach that is very similar to the Apollo lunar missions with limited exploration range from the landing site. There is also very little ability to handle any off-nominal events and still conduct a viable surface mission. This mission approach only requires a lander for the surface phase, which provides the potential for overall cost reduction and lower risk for the surface phase of the mission.

The long-stay mission architecture lends itself to a flexible surface exploration strategy. The crew has approximately 18 months in which to perform the necessary surface exploration activities; the strategy thus follows a less rigorous, less scheduled approach. Ample time is provided to plan and re-plan the surface activities, respond to problems, and readdress the scientific questions that were posed early in the mission. In addition, the long-surface mission duration maximizes mission and scientific return, enabling a robust exploration strategy with the ability to reach ranges at a greater distance from the landing site, explore a greater number of sites, and conduct more complex exploration such as deep drilling. Extended surface operation does pose additional risk to the crew, depending on the specific tasks and frequency. In addition, the long-surface stay imposes additional system reliability and maintainability requirements.

### 3.3.9 Mission sequence for the Opposition Class (short-stay) mission
The focus of the Opposition Class mission is to strike a proper balance between length of the overall mission and the total mass that must be launched. As the mission duration is shortened, the total mission mass grows exponentially. For this mission, a split mission approach is used whereby mission cargo is delivered to Mars one opportunity before the crew. This provides a significant advantage in reducing total mission mass. In fact for harder mission opportunities, pre-deployment of mission assets is required to obtain reasonable initial masses. The first phase of the short-stay mission architecture begins with the pre-deployment of the Mars DAV to Mars orbit. The DAV, along with its in-space propulsion system, is launched, assembled, and checked out in LEO. After all of the systems have been verified and are operational, the vehicles are injected into minimum-energy transfers from Earth orbit to Mars. Upon arrival at Mars, the vehicles are captured into a high-Mars orbit and remain in a semi-dormant mode, waiting for the arrival of the crew approximately 24 months later. Periodic vehicle checks and orbital maintenance are performed to place the vehicles in the proper orientation for crew arrival. The specifics of the Earth departure and Mars arrival scenarios are dependent on the transportation technologies that are chosen. The overall mission sequence that is used for the short-stay strategy is depicted in figure 3-14.

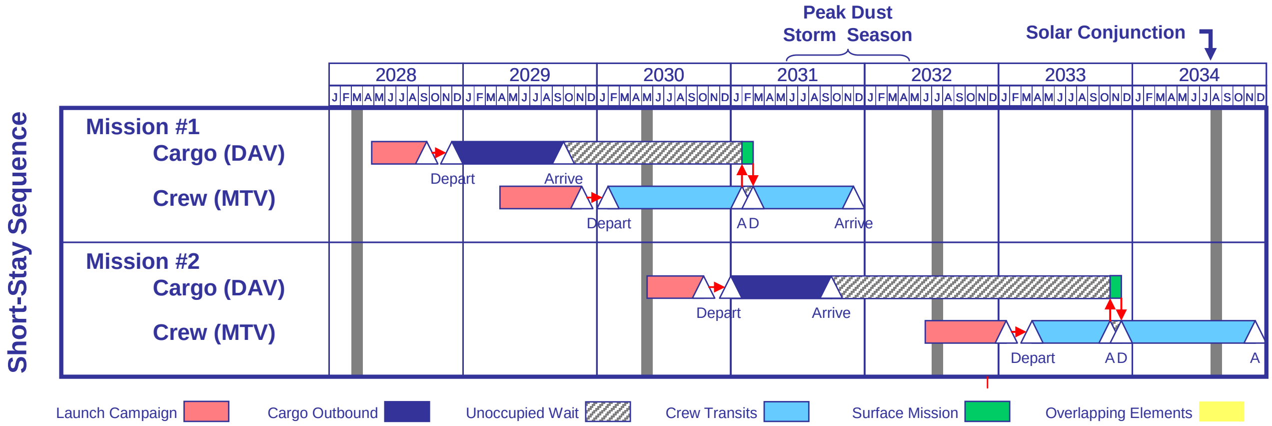



**Figure 3-14.** Short-stay mission sequence.

The second phase of this architecture begins with the launch, assembly, and checkout of the MTV during the next injection opportunity. The MTV serves as the interplanetary support vehicle for the crew as well as the outbound transportation system. A vehicle checkout crew is delivered to the MTV in Earth orbit to perform vital systems verification and any necessary repairs prior to departure of the flight crew. After all of the vehicles and systems, including the Mars DAV, the Orion Earth return vehicle, and the MTV, are verified as operational, the flight crew is injected on the appropriate short-stay trajectory. The length of the outbound transfer to Mars is dependent on the injection opportunity. Upon arrival at Mars, the crew must rendezvous with the DAV. After arriving at Mars, the crew has up to 30 to 90 days (depending on the mission opportunity and propulsion choice) to make all of the necessary orbital adjustments for the return trajectory and conduct the surface mission.

The DAV serves as the primary transportation and crew support element for the planetary exploration phase of the mission. This vehicle is designed to transport the mission crew from a high Mars orbit to the surface of Mars, support the crew for up to 30 days while on the surface, and return the crew from the surface to the high Mars orbit whereby it performs a rendezvous with the MTV. The functional capabilities of the DAV must accommodate the ability to operate in a fully automated mode since it is anticipated that the crew will be incapable of performing complicated tasks due to the long exposure to microgravity while in transit. Vehicle terminal phase targeting/control, post-landing safing, initial flight-to-surface transition, and appendage deployments must occur without crew exertion. Thus, the vehicle must provide adequate time for the crew to re-adapt to 0.38 G on Mars. During this period, no strenuous activities (e.g., EVA) will be scheduled for any crew members, and the focus of the operations will be on developing adequate crew mobility and maintaining systems operability.

The focus of the surface exploration phase is to conduct scientific investigations of the local landing vicinity. Of the 30 days on the surface of Mars, as many as 21 potential EVA sorties can be conducted. This strategy provides time for the crew to acclimate to the martian environment as well as perform the closeout and vehicle checks that are necessary at the end of a surface mission prior to ascending back to orbit. During the science investigations, a 10-m radius has been established as a reasonable traverse radius about the landing zone. This radius is derived from the maximum unassisted walk-back distance of a suited crew member due to rover failure. This radius also considers the rate life support consumables within the EVA system to ensure that they are not depleted the crew members are returning to the SHAB.

After completion of the surface mission, the crew performs the necessary closeout and shutdown operations of the vehicles. Surface elements, including science instruments, are placed in an automated operations mode for Earth-based control. The crew then ascends in the Mars ascent vehicle and performs a rendezvous with the waiting Earth return vehicle. This vehicle is used to return the crew from Mars, ending with a direct entry at Earth.

For this architectural comparison, it was assumed that the length of stay would be limited to 30 days, which is consistent with the capabilities of the DAV. If surface durations in excess of 30 days were required, the architecture team strongly encouraged the introduction of an additional SHAB. Since the addition of this habitat was not included in the comparison, the surface stay was limited to 30 days total.

### 3.3.10 Mission sequence for the Conjunction Class (long-stay) mission

The philosophy of the long-stay mission architecture approach is to minimize the exposure of the crew to the deep-space radiation and zero-g environment while at the same time maximizing the scientific return from the mission. This is accomplished by taking advantage of optimum alignment of the planets for both the outbound and return trajectories by varying the stay time on Mars, rather than forcing the mission through nonoptimal trajectories, as in the case of the short-stay missions. This approach allows the crew to transfer to and from Mars on relatively fast trajectories, on the order to 6 to 7 months, while allowing them to stay on the surface of Mars for a majority of the mission, on the order of 18 months.

The surface exploration capability is implemented through a split mission concept in which cargo is transported in manageable units to the surface or Mars orbit and checked out in advance of committing the crews to their mission. The split mission approach also allows the crews to be transported on faster, more energetic trajectories, minimizing their exposure to the deep-space environment while the vast majority of the material that is  sent to Mars is sent on minimum-energy trajectories. The trajectory analysis that was discussed earlier was used to ensure that the design of the space transportation systems could be flown in any opportunity. This is vital to minimize the programmatic risks that are associated with funding profiles, technology development, and system design and verification programs. The overall mission sequence that was used for the long-stay strategy is depicted in figure 3-15.

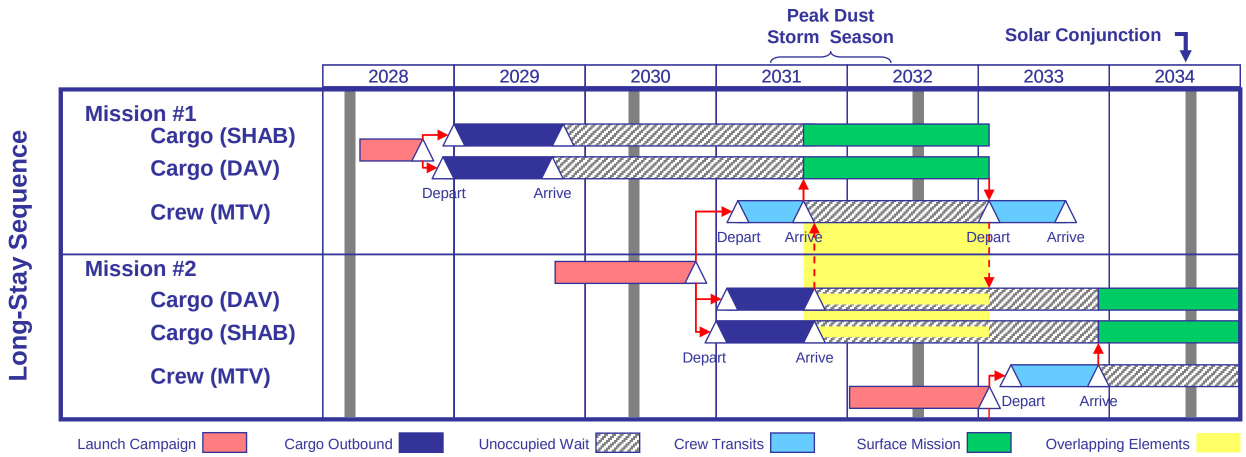



**Figure 3-15.** Long-stay mission sequence.

The first phase of the long-stay mission architecture begins with the pre-deployment of the first two cargo elements, the DAV and the SHAB. These two vehicle sets are launched, assembled, and checked out in LEO. After all systems have been verified and are operational, the vehicles are injected into minimum-energy transfers from Earth orbit to Mars. Upon arrival at Mars, the vehicles are captured into a high-Mars orbit. The specifics of the Earth departure and Mars arrival scenarios are dependent on the transportation technologies that are chosen. The DAV remains in Mars orbit in a semi-dormant mode, waiting for arrival of the crew 2 years later. The SHAB is captured into a temporary Mars orbit and then performs the entry, descent, and landing on the surface of Mars at the desired landing site. After landing. The vehicle is remotely deployed, checked out, and all systems are verified to be operational. Periodic vehicle checks and remote maintenance are performed to place the vehicles in proper orientation prior to crew arrival.

A key feature of the long-stay mission architectures is the deployment of significant portions of the surface infrastructure before the human crew arrives. This strategy includes the capability for these infrastructure elements to be unloaded, moved significant distances, connected to each other, and operated for significant periods of time without humans present. In fact, the successful completion of these various activities will be part of the decision criteria for launch of the first crew from Earth. Pre-deployed and operated surface elements include the SHAB, power system, thermal control system, communications system, robotic vehicles, and navigation infrastructure.

The second phase of this architecture begins during the next injection opportunity with the launch, assembly, and checkout of the MTV. The MTV serves as the interplanetary support vehicle for the crew for a round-trip mission to Mars orbit and back to Earth. Prior to departure of the flight crew, a separate checkout crew is delivered to the MTV to perform vital systems verification and any necessary repairs prior to departure of the flight crew. After all of the vehicles and systems – including the Mars DAV, SHAB, and MTV – are verified operational, the flight crew is injected on the appropriate fast-transit trajectory towards Mars. The length of this outbound transfer to Mars is trajectory Dependent, and ranges from 180 to 210 days. Since the crews are delivered to Mars on their round-trip vehicle includeing the return propellant, the crew does not have to perform any rendezvous or other complicated orbital maneuvers to return from Mars back to Earth. Upon arrival at Mars, the crew performs a rendezvous with the Mars DAV, which serves as their transportation leg to and from the Mars surface. After arriving at Mars, the crew has ample time (up to 18 months) to make all of the necessary orbital adjustments for the return trajectory and conduct the surface mission.

The DAV serves as the primary transportation element for the crew in the vicinity of Mars. The vehicle is designed to transport the mission crew from a high Mars orbit to the surface of Mars, support the crew for the initial post-landing acclimation period (up to 30 days), and return the crew from the surface to the high Mars orbit whereby it performs a rendezvous with the MTV. The functional capabilities of the DAV must accommodate the ability to operate in a fully automated mode since it is anticipated that the crew will not be capable of performing complicated tasks due to the long exposure to microgravity while in transit. Vehicle terminal phase targeting/control, post-landing safing, initial flight-to-surface transition, and appendage deployments must occur without crew exertion. Thus, the vehicle must provide adequate time for the crew to re-adapt to 0.38 G on Mars. During this period, no strenuous activities (e.g., EVA) will be scheduled for any crew members, and the focus of operations will be on developing adequate crew mobility and maintaining systems operability.

Current human health and support data indicate that it may take the crew up to 1 week to acclimate to the partial gravity of Mars. After the crew has acclimated, the focus of the initial surface activities is on transitioning from the lander to the SHAB. This includes performing all remaining setup, checkout, and maintenance that could not be performed remotely from Earth. The crew has as many as 30 days after landing to perform all of the necessary startup activities of the SHAB. During this period, local science is also conducted to ensure that the initial science objectives can be met if early ascent from the surface is required. Lastly, the Mars ascent vehicle is connected to the SHAB power system and placed in a semi-dormant mode since it will not be needed again until ascent from the surface is required. Although the lander is in a semi-dormant mode, emergency abort to orbit (ATO) is available throughout the surface exploration phase of the mission.

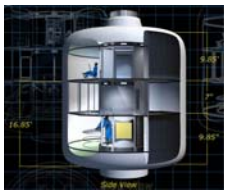
The long-stay mission architecture lends itself to a very robust surface exploration strategy. Since the crew has approximately 18 months in which to perform the necessary surface exploration activities, the strategy follows a less rigorous, less scheduled approach. Ample time is provided to plan and re-plan surface activities, respond to problems, and readdress the scientific questions that were posed early in the mission. The focus during this phase of the mission will be on primary science and exploration activities that will change over time to accommodate early discoveries. A general outline of crew activities for this time will be provided before launch and updated during the interplanetary cruise phase. This outline will contain detailed activities to ensure initial crew safety, make basic assumptions as to initial science activities, schedule periodic vehicle and system checkouts, and plan for a certain number of sorties. Since much of the detailed activity planning while on the surface will be based on initial findings, it cannot be accomplished before landing on Mars. The crew will play a vital role in planning specific activities as derived from more general objectives that are defined by colleagues on Earth.

Before committing the crew to Mars ascent, full systems checkout of the ascent vehicle and the MTV is required. Because both vehicles are critical to crew survival, sufficient time must be provided prior to launch to verify systems and troubleshoot any anomalous indications prior to crew use. In addition, the SHAB will be placed in a dormant mode for potential re-use by future crews. This includes stowing any nonessential hardware, safing critical systems and their backups, and performing general housekeeping duties. Lastly, surface elements, including science instruments, are placed in an automated operations mode for Earth-based control. The crew then ascends in the Mars ascent vehicle and performs a rendezvous with the waiting MTV. This vehicle is used to return the crew from Mars, ending with a direct entry at Earth. It should also be noted that the MTV will also contain all necessary contingency supplies in the event that the crew must depart early from the martian surface and wait in orbit for the return opportunity to open up.

An overview of the mission payloads that were used for the Phase 1 decision packages is depicted in figure 3-16.

**Crew Support During Transit**

**Mars Transit Habitat**



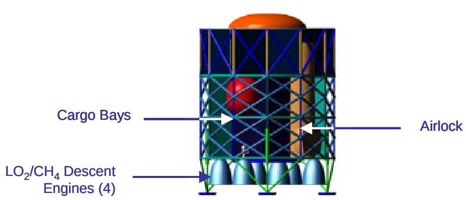
- Function: Support mission crew of six for up to 600 days in deep-space (900 in a contingency)
- Basis of estimate: TransHab, DRM 3.0, 4.0, DPT, NEXT
- Consumables for nominal and contingencies
- Advanced technologies assumed (e.g. closed life-support, inflatable structures, composites, etc.)
- Nominal mass assumed: ~ 27.5 t

**Orion (Block 3)**

- Function: Deliver six crew to MTV in LEO and direct entry return of crew at end of mission
- Basis of estimate: Current Orion estimate
- Entry speed limited to 13 km/s
- Mass assumed: ~ 10 t
- Mass revision needed for:
  - TPS enhancements for higher entry speed
  - Small tailored service module

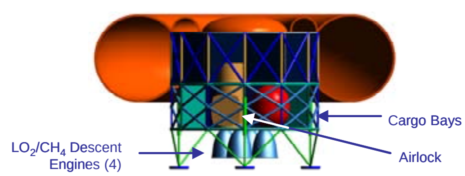
**Crew Support on Mars Surface**

**Mars Lander**




- Function: Transport six crew from Mars orbit to the surface and return to Mars orbit
- Basis of estimate: DRM 3.0, 4.0, DPT, NEXT
- Provides contingency abort-to-orbit capability
- Vehicle supports crew for 30 sols
- Vehicle capable of utilizing locally produced propellants
- Advanced technologies assumed (composites, O2/CH4 propulsion, etc.)
- Wet mass assumed: ~ 68 t

**Mars Surface Habitat**




- Function: Support mission crew of six for up to 550 sols on the surface of Mars. Included in architectures with surface stays in excess of DAV (30 days) capabilities.
- Basis of estimate: DRM 3.0, 4.0, DPT, NEXT
- Provides robust exploration and science capabilities
- Descent vehicle capable of landing ~40 t
- Advanced technologies assumed (e.g. closed life-support, inflatables, composites, O2/CH4 propulsion, ISRU for EVA makeup, etc.)
- Wet mass assumed: ~ 68 t

**Figure 3-16.** Mission payloads used for Phase 1 decisions.

### 3.3.11 Mission-type special considerations: science goals and objectives

During the mission-type deliberations, the MAWG solicited the help of the MEPAG to provide an assessment of the relative advantages and disadvantages of the two mission types under considerations. The MEPAG sponsored the creation of a special assessment group, the HEM-SAG, which reviewed the proposed surface exploration strategies that are associated with both the long- and short-stay mission concepts. The MAWG specifically asked the HEM-SAG to provide an assessment of the relative advantages and disadvantages of not only mission concepts that are driven by length of stay, but also those that are associated with potential return to the same exploration site or conducting subsequent missions to different exploration sites. The general conclusions resulting from the HEM-SAG deliberations are provided below and are graphically depicted in figure 3-17.

| | Short Stay | Long Stay | Comments |
|---|---|---|---|
| **One Site** (Same each mission) | ▪ **LEAST FAVORED**<br>▪ Below the "science floor" (but there is science that could be done, especially via samples back to Earth | ▪ **3rd Most favored (of 4)**<br>▪ 100's km mobility<br>▪ Ideally 1000 km mobility (pressurized)<br>▪ May require unique landing site (with extreme local diversity) | ▪ Long stay may require 1000's km surface mobility if to same site each time (otherwise not scientifically favored) |
| **Multiple Sites** | ▪ **2nd Most Favored**<br>▪ 10's km mobility<br>▪ 100 kg samples to Earth (total Apollo-class)<br>▪ MER-class analytical?<br>▪ Robotic "fetch" rovers?<br>▪ Leave-behind robotic systems (autonomous drill?)<br>▪ Favors sample collection over in situ analysis | ▪ **MOST FAVORED**<br>▪ 100's km surface mobility<br>▪ 100 m vertical (drilling)<br>▪ TBD analytical capability (in situ, possibly MSL class)<br>▪ 100's kg to Earth (Apollo-class)<br>▪ Extensive lab for sample high-grading key for many science issues (astrobiology) | ▪ Highest Science yield requires Diversity (time, space) for optimization<br>▪ Requires ~ 100's km horizontal mobility (all cases)<br>▪ ISSUE: Drilling? (100m desire, 1 m pits req'd)<br>▪ Mass to Earth? (likely to be 100kg minimum) |

**Figure 3-17.** Science goals mission-type considerations.

#### *3.3.11.1 Opposition Class missions (short-stay): scientific position*

Short-surface duration missions, while offering the potential for breakthrough, human-enabled science, are not favored for science-driven exploration for several reasons: Short-stay human surface missions cannot make best use of mobility to optimally explore a region due to time available for EVA (and for subsurface access system operation, such as a deep drill). Nor do they optimize the "iteration cycle time" that is associated with in-situ field investigations on the basis of time available (too few cycles for adapting to the unexpected scientific context that is likely to emerge). Finally, short-stay human surface missions do not allow time for sample high-grading to ensure a best subset of the materials that are returned for detailed analysis on Earth. This limits the discovery potential that is intrinsic to field sampling.

#### *3.3.11.2 Conjunction Class missions (long-stay): scientific position*

Conjunction Class missions that provide extended duration on the surface while maximizing the exploration range from the landing site are most favored to optimize the scientific yield. Long surface stay allows maximal use of human "on-site" observational and intuitive scientific capabilities, even if EVA is restricted to approximately 25% of available time. By maximizing opportunities for adapting scientific investigations to a given region, the probability of paradigm-busting discoveries increases exponentially over focused, robotic surface investigations such as those that are presently in operation with the MERs. Long-surface stay also maximizes the human opportunities for using mobility (horizontal and vertical) to more completely explore a compelling region at scales that are commensurate with processes that preserve evidence of past life on Earth. In addition, the long-surface-stay scenario allows the humans "on site" to make best use of their non-EVA time to employ general analysis "tools" to investigate sampled materials and, hence, to best select the optimized subset (so-called splits) for return to Earth. It should be noted that long surface stays at three independent and different human exploration sites is the most favored option.

### 3.3.12 Mission-type special considerations: human health and performance

The Crew Health and Performance (CHP) Team of the MAWG evaluated both the short- and long-stay mission architectures for the human mission to Mars in the NASA CxP: a long-stay scenario of 18 months on Mars with approximately 6 months transit both out to Mars back to Earth, and a short-stay scenario of about 1 month on Mars, 6 to 10 months outbound and 10 to 13 months inbound. When all of the human health and performance disciplines were considered, no clear advantage of either option was identified on the basis of crew health, safety, and performance. A summary of the key human health and performance findings is provided in figure 3-18. It is important to note that the risk assessment that is provided by the radiation discipline indicates that both the short-stay (Opposition Class) and long-stay (Conjunction Class) mission options pose a high risk that crew members will exceed current permissible radiation exposure limits. This assessment is discussed in further detail later in this section.

| HHP Component | Short Stay (Opposition-class; 22 months total) | Long Stay (Conjunction-class, 30 months total) |
|---|---|---|
| Physiological Countermeasures | ▪ Extended 0-g transits at limits of human spaceflight experience base<br>▪ Preferred option only if AG available | ▪ 0-g transit phases well within experience base<br>▪ 3/8-g surface phase outside experience base, will be partially mitigated by Lunar Outpost experience |
| Human Factors & Habitability | ▪ Not preferred option without access to Surface Habitat | ▪ Preferred option with access to Surface Habitat |
| Radiation | ▪ Higher risk of carcinogenesis, acute syndromes, CNS effects and degenerative effects due to longer transits (SPE & GCR) and close perihelion passage (SPE effects)<br>▪ Option is well outside current permissible exposure limits | ▪ Slightly preferred option<br>▪ Prolonged exposure to poorly-understood surface mixed-field (neutrons and charged particles) environment. Mars surface radiation environment may be more severe than previously estimated.<br>▪ Option is well outside current permissible exposure limits |
| Behavioral Health & Performance | ▪ Preferred option due to shorter overall duration<br>▪ Possible risk due to higher acute radiation exposure within 0.7 AU | ▪ Increased risk due to longer overall duration |
| Medical Capabilities | ▪ Slightly preferred option due to less risk exposure of shorter duration | ▪ Slightly increased risk due to longer overall duration |

**Figure 3-18.** Summary of human-health-mission-type considerations.

A number of significant knowledge gaps and technologies to be developed were identified by the CHP disciplines, which concluded that no legitimate discrimination between the two scenarios would be valid, based on that analysis with current knowledge and space flight experience, because higher-order details of the scenarios have not been fully developed. However, any Mars exploration option that is selected by ESMD can be implemented concomitant with acceptance of all residual human health and safety risks identified by the CHP disciplines and their parent organizations.

Both the long-stay (Conjunction Class) and short-stay (Opposition Class) missions were analyzed from a human health and performance perspective. The assumed mission characteristics for this analysis are listed in table 3-6.

**Table 3-6.** Mission Characteristics

| Factor | Short-stay Mission | Long-stay Mission |
|---|---|---|
| Travel time in transit to Mars | 313 days (~10 months) | 180 days (~6 months) |
| Travel time in transit to Earth | 308 days (~10 months) | 180 days (~6 months) |
| Total transit time in deep space<br>% total mission duration | 621 days (~21 months)<br>94% | 360 days (~12 months)<br>40% |
| Surface stay time<br>% total mission duration | 40 days (~1 month)<br>6% | 545 days (~18 months)<br>60% |
| Total mission time | 661 days (~22 months) | 905 days (~30 months) |
| Closest solar approach –<br>Without Venus swing-by<br>With Venus swing-by | ~0.5 AU<br>As close as 0.38 AU | 1 AU N/A |

AU: astronomical unit (mean distance from the Earth to the sun).

The Earth-to-Mars transit time for the short stay is at the limits of the human space flight experience base, but the transit time for the long stay is within the experience base. The experience base for surface time is now very limited, but should be increased by Lunar Outpost experience; this represents a future reduction in uncertainty and possible mitigation of a risk.

The disciplines that are represented by the CHP Team assessed the scenarios within their area of expertise, thus resulting in different assessments of the two mission scenarios. If AG is used as a countermeasure, the Zero-g Countermeasures analysis favored the short-stay option; but, in the absence of AG, it demonstrated no preference. The Human Factors and Medical Care analyses also showed a slight advantage for the short-stay option. The Environmental Health analysis favored the long-stay option if an SHAB is only available for the long stay. The Radiation Protection analysis slightly favored the long-stay option, and the Medical assessment found an SPE-induced radiation contingency to be one of the most difficult mission environmental risks to manage, unless considerable shielding improvements are implemented. Based on past assessments, it is believed that a heavily shielded location (configuration and subsystem placement) can be achieved without additional parasitic shield mass. However, an integrated assessment across all of the disciplines did not ascribe a clear advantage to either option.

Both options would pose significant health risks. The short-stay option has about 27% less mission-duration risk than the long-stay option because it requires less time away from Earth (22 months vs. 30 months), but this is offset by the fact that many (but not all) of the health risks increase most rapidly early in flight and then decrease more slowly, if at all, after the first few months. The radiation exposure risk, however, is not less for the short-stay mission. The advantages and disadvantages found by each discipline are described below.

#### *3.3.12.1 Radiation risk: comparison of short- and long-surface-stay missions*

The Mars radiation assessment that is presented here compares the risk due to radiation exposure to crew members between the short-stay (621 days in free space, 40 days on surface) and long-stay (360 days free space, 545 days on surface) Mars missions. The short-stay mission class includes a Venus swing-by or deep-space maneuvers with a large portion of the Earth return trajectory at less than 1.0 AU. The closest solar approach for the Earth return trajectory for a nominal mission is between 0.5 and 0.8 AU. The long-stay mission class maintains transit trajectories are 1 AU or greater. The human Mars mission is assumed to occur at any time during the solar cycle; that is, preference will not be given to solar maximum conditions when GCR flux is at its lowest although the probability of SPEs is greatest. The assessment is based on past analyses, current knowledge, and embedded assumptions. Where possible, the health risks are quantitatively assessed. However, due to lack of current research results, other health risks can only be qualitatively assessed at this time. Recommendations are provided where past analyses should be updated based on new research results and mission definition.

Both mission types are well above permissible exposure levels for crew with large uncertainties. The estimated risk of radiation-exposure-induced death (%REID) is estimated to be 7.8% for short stay vs. 8% for long stay. The 95th confidence interval (CI) for this estimate is well above 16%. Current permissible exposure limits (PELs) restrict exposure to 3% at the 95th confidence level (CL). Risk mitigation strategies as well as uncertainty reduction are required prior to a human Mars mission.

Based on current knowledge about space radiation risks to humans, the scientific basis to pick a short-stay over a long-stay mission, or vice versa, has not been fully established. Lack of knowledge that contributes to the difficulty of selecting one mission class over the other includes:

1. The role of non-targeted effects for cancer induction during both the short- and long-stay missions. If non-targeted effects are found to make up a significant fraction of the overall GCR cancer risk, there may be little dependence of the risk of carcinogenesis on mission duration or shielding amount.

2. Insufficient knowledge about the amount of protection provided by the Mars surface and atmosphere, especially during the long-stay mission. The Mars radiation environment may be more severe than previously estimated due to the production and transport of neutrons, mesons, muons, and electromagnetic cascades. Effects of a mixed field environment (neutrons and charged particles) on radiobiological risks are unknown. A larger risk contribution during a long stay on the surface may be a future discriminator. In-situ precursor validation data are required.

The issues that are involved in short-stay missions may be larger to overcome due to longer times in free space and trajectories with close passage to the sun. Major factors include:

1. The probability of large SPE exposure at close proximity to the sun contributing to cancer risk and some acute radiation syndromes (non-mission-threatening). Understanding dose rate effects on cancer morbidity and the radial gradient (including energy and rate) of SPEs at distances of less than 1 AU must be pursued. (*A major assumption for this assessment is that the MTV provides a heavily shielded location at 20 g/cm$^2$ without additional parasitic shield mass.*)

2. The poorly understood risk of central nervous system (CNS) and degenerative tissue damage due to increased exposure from heavy ions. Research is needed to quantify heavy ion effects on CNS, cardiac, circulatory, and digestive diseases.

Investments in risk mitigation strategies should include advanced shielding technologies, countermeasures (radioprotectants and pharmaceuticals), and individual-based risk assessments as well as significant uncertainty reduction.

**Radiation exposure limits for missions beyond low-Earth orbit**

The National Council on Radiation Protection (NCRP) provides guidelines to NASA on crew-permissible exposure limits. Previous NCRP Reports in 1989 and 2000 (Nos. 98 and132[53]) specifically state that the methods that are used to project risk for LEO missions are severely limited for exploration missions because of large uncertainties in the biological effects of high linear-energy transfer (LET) radiation, especially the high-charge and -energy nuclei of the GCR environment. Similar concerns are noted in reports by the National Academy of Sciences (NAS) in 1970, 1973, and 1996 (NAS, 1970[54]). NASA uses probabilistic risk assessment (PRA) approaches (NCRP Report 126, 1997) to evaluate the impacts of uncertainties on dose limits and the evaluation of risk mitigation approaches (Cucinotta et al., 2001[55]), (Cucinotta et al., 2004[56]), (Cucinotta et al., 2006a[57]). Major uncertainties are the evaluation of dose-rate effects and radiation quality effects. Other uncertainties include the evaluation of radiation transmission factors and space environments as well as differences in biological responses during space flight. Quantitative uncertainty estimates can be made for cancer and acute risks. However, for cancer risks the uncertainties due to possible nonlinear responses and a radiation quality dependence on tumor latency have not been made at this time. There are insufficient data to make quantitative estimates for CNS and degenerative risks to tissues (e.g., heart, digestive, etc.) using the available human data for gamma rays alone.

PELs are baselined in the “NASA Space Flight Human System Standard 3001 (NASA STD, 3001[58]).” Career PELs are imposed not to exceed a probability of 3% excess risk of lifetime fatal cancer within a 95% confidence interval. Mission and vehicle requirements are allocated to human systems (transit vehicles, pressurized rovers, habitats, etc.) with consideration given to the crew short-term and career PELs. In addition, NASA programs must follow the as low as reasonably achievable (ALARA) principle, which is a legal requirement intended to ensure astronaut safety.

In future, NASA will have to determine an acceptable level of risk due to radiation exposure for a human Mars mission. Efforts to increase the acceptable level of risk beyond the lunar sortie mission value of 3% probability of cancer fatality will have to address the possibility that as the acceptable levels of cancer risk are increased, concomitant non-cancer mortality and significant morbidity risks will likely occur. Non-cancer risks are expected to be deterministic in nature, occurring above a dose threshold with a severity that increases with dose. In contrast, cancer risks are stochastic in nature with only the probability of the risk, not the severity, increasing with dose. The

[53]NCRP. National Council on Radiation Protection reports: 1989, 1997, 2000.
[54]NAS. National Academy of Science reports: 1970, 1973, 1996.
[55]Cucinotta, F.A., Schimmerling, W., Wilson, J.W., Peterson, L.E., Saganti, P., Badhwar, G.D., and Dicello, J.F.: Space Radiation Cancer Risks and Uncertainties for Mars Missions. Radiation Research 156, 682-688, 2001.
[56]Cucinotta, F.A., Schimmerling, W., Wilson, J.W., Peterson, L.E., and Saganti, P.B.: Uncertainties In Estimates Of The Risks Of Late Effects From Space Radiation. Advances in Space Research, 34(6), 1383-1389, 2004.
[57]Cucinotta, F.A., and Durante, M.: Cancer risk from exposure to galactic cosmic rays: implications for space exploration by human beings. The Lancet Oncology 7, 431-435, 2006a.
[58]NASA Standard 3001, NASA Space Flight Human System Standard Volume 1: Crew Health, National Aeronautics and Space Administration Washington, D.C., 2007.

likelihood of in-flight health risks for 2.5- to 3-year missions may also occur under these conditions. Furthermore, the basis for radiation protection requirements will be weakened if acceptable levels of risks are set too high.

**Mars mission radiation exposure**

Radiation exposure on a human Mars mission will come from the continuous bombardment of GCR, the possibility of SPEs, and, potentially, from nuclear propulsion (if selected). On the Mars surface, the planet protects from half of the free space radiation environment from below while the $CO_2$ atmosphere provides additional protection from above. *In terms of radiation risk to crew, the most distinct difference between the short-surface-stay time and the long-surface-stay time missions is the increased risk due to longer time spent in free space and the close trajectory proximity to the sun for the short-stay-time mission.*

**Mars surface radiation exposure estimates**

The Mars atmosphere (low-density COSPAR model; Smith and West, 1983[59]) provides 16 g/cm$^2$ of $CO_2$ protection in the straight-up direction with protection increasing to over 50 g/cm$^2$ at large zenith angles (toward the horizon) at 0-km altitude (Simonsen 1990[60]). Previous estimates of Mars surface exposures concluded that the atmosphere significantly reduces the exposure from SPEs and GCR (Simonsen et al., 1990; Simonsen and Nealy, 1993[61]; Simonsen et al., 2000[62]; Transport models to estimate SPE exposures on the surface of Mars are fairly mature however, further considerations for GCR surface exposure estimates and validation are warranted due to the uncertainty of secondary neutron production, and the production and transport of mesons, muons, and electromagnetic cascades.

The uncertainties in the surface environment will not greatly impact the short stay of 40 days but may significantly change results for the long stay times of 545 days. Additional pre-cursor measurements are needed for validation prior to a long stay mission. In-situ precursor measurements supporting validation of the calculated Mars surface environment include: charged particle spectral measurements including electrons if possible, and low energy spectrum neutron measurements. Current orbital neutron measurement data from Mars Odyssey’s High Energy Neutron Detector data can support validation. The future data from the Mars Science Laboratory mission Radiation Assessment Detector will provide surface measurements during solar maximum. Plans currently include additional surface measurements in the 2018 timeframe during solar minimum conditions. Earth high altitude balloon data can be utilized to support model development for the production and transport of mesons, muons, and electromagnetic cascades.

**Galactic cosmic radiation**

In comparing the short stay time with the long stay time mission, the total exposure (mSv) from GCR is approximately the same. That is, the greater exposure during the short stay transit (621 days) is approximately the same as the exposure during the long stay transit of 360 days plus the 545 days on the surface. With similar exposure levels (mSv), the risk of cancer mortality is nearly the same or unable to be differentiated based on current knowledge. However, a larger fraction of the exposure received from GCR on the short stay mission, with its long transit time in free space, is from the heavy ion component of GCR ($Z>10$). Thus, changes in cancer risk projection models may significantly change this assessment in the future if heavy ion effects are estimated to be higher or lower.

For example, current research indicates that radiation carcinogenesis can occur through non-targeted effects where radiation carcinogenesis originates in cells adjacent to a HZE nuclei path. Determining the role of DNA damage vs. non-targeted effects has large implications for radiation shielding, mission duration, and approaches to the design of biological countermeasures. If only the cell nucleus and resulting DNA damage is the target for carcinogenesis, HZE nuclei will interact with the target only every few weeks to months in space. In this case, shielding below one track per target is plausible. However, for a target size of several cell layers across (non-targeted effects model),

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[59]Smith,R.E. and West, G.S., compilers 1983: Space and Planetary Environment Criteria Guidelines for use in Space Vehicle Development, 1982 Revision (Volume1). NASA TM-82478.

[60]Simonsen, L.C.; Nealy, J.E.; Townsend, L.W.; and Wilson, J.W.: Space Radiation Dose Estimates on the Surface of Mars. *Journal of Spacecraft and Rockets*, Vol. 27, No. 4, July-August 1990, pp. 353-354.

[61]Simonsen, L.C.; and Nealy, J.E.: Mars Surface Radiation Exposure for Solar Maximum Conditions and the 1989 Solar Proton Events. NASA TP-3300, 1993.

[62]Simonsen, L.C.; Wilson, J.W.; Kim, M.H.; and Cucinotta, F.A., “Radiation Exposure for Human Mars Exploration,” Health Physics, Vol. 79, No. 5, pp. 515-525, November 2000.

shielding below one HZE track per target is not possible, and there may be little dependence of the risk of carcinogenesis on mission duration or shielding amount.

In addition, evidence indicates the risk of non-cancer fatalities (heart, circulatory, and digestive) from GCR will be greater on the short-stay mission due to the greater higher heavy-ion component. There are currently enough data to make a preliminary estimate of fatal heart disease. Similarly, the risk of acute and latent CNS affects (motor function, behavior, or neurological disorders) is also hypothesized as being greater for the short-stay mission because heavy ions are hypothesized as being the greatest contributor to CNS risk. A threshold value for modified behavior may exist for humans (as evidenced by rats) and would most likely depend on age, previous CNS injury (concussion), and genetic makeup. There are not enough data currently available to quantitatively assess.

The Human Research Program (HRP) is performing research on radiation carcinogenesis from non-targeted effects, degenerative tissue disease (risk of latent non-cancer fatalities), as well as acute and latent CNS risk with anticipated results in the 2015–2020 timeframe based on current budget projections.

**Solar proton events**

Acute radiation syndrome and radiation carcinogenesis are potential risks due to SPE exposures. The magnitude of these risks will depend on the probability of flare occurrence, shielding provided, and distance from the sun at time of occurrence. *Radiation carcinogenesis* is a risk for both short- and long-stay Mars missions; however, sufficient vehicle shielding and the protection that is afforded by the Mars surface and atmosphere can significantly reduce this risk. The risk of *acute radiation syndrome* will be an additional risk during the transit phase of the short-stay mission because of close passage to the sun.

Since the surface provides significant protection, the possibility of large SPE exposures will be limited to free space. The probability of occurrence of a single large event will be proportional to the length of time in free space. Therefore, a large SPE exposure from a single event will be 1.7 times larger for the short-stay mission (621days/360days) compared with the long-stay mission during the same period of the solar cycle. The probability of a second SPE ($F>10^9$ F>30MeV) is small; thus, the difference between the two missions for the occurrence of a second event will be much less than 1.7 (Kim et al., 2007). It is recommended that the likelihood of occurrence of a large SPE or multiple SPE events as a function of energy be assessed for both mission types. For softer (less energetic) SPEs, the shielding that will be provided by the spacecraft should be sufficient; however, a PRA for larger, more energetic SPEs is required to prepare for distances that are close to the sun.

The magnitude of the SPE fluence increases at closer radial distances to the sun. The working group consensus recommendation for radial fluence extrapolation from 1.0 AU to other distances (Jet Propulsion Laboratory report edited by Feynman and Gabriel,1988) is to use a functional form of $1/R^{2.5}$ and expect variations ranging from$1/R^3$ to $1/R^2$. This generalization only applies to well-connected solar-flare-associated events (i.e, near-sun injection events). They do not always apply to the extended interplanetary shock source events. *Following this recommendation, an exposure due to a large SPE can be between four ($1/R^2$) to eight ($1/R^3$) times greater at a distance of 0.5AU from the sun with a functional extrapolation of 5.6 times greater ($1/R^{2.5}$) for the short-stay mission trajectory.* However, little data exist to estimate the energy dependence of the radial gradient of an SPE as a function of distance from the sun. Much of the above extrapolations are based on protons in the energy range of a few MeV to tens of MeV, which can be easily stopped by sufficient shielding such as that provided by a heavily shielded vehicle location and the self shielding of the body. More data are required for extrapolations for energies greater than approximately 150 MeV, where the contribution to crew exposure is the greatest.

Miewaldt (2006[63]) states that “Studies and models of the dependence of SPE intensities on radial distance from the [sun] do not all agree (e.g., Reames and Ng, 1998[64]; Ruzmaikan et al., 2005[65]; Lario et al., 2006[66]) suggesting the need for new measurements by a multi-spacecraft mission such as Inner Heliosphere Sentinels during the next solar maximum.” The Heliophysics division in the NASA Science Mission Directorate, both alone and in collaboration

[63]Miewaldt,R.A.; Solar Energetic Particle Composition, Energy Spectra, and Space Weather. Space Science Reviews (2006) 124: 303–316.
[64]Reames, D. V., and Ng, C. K.:, " Streaming-limited Intensities of Solar Energetic Particles" 1998, *Astrophys. J.* 504, pg. 1002.
[65]Ruzmaikan, A., Li, G., Feynman, J., and Jun, I.: 2005, in: *ESA SP-592: Solar Wind 11/SOHO 16, Connecting Sun and Heliosphere*, pp. 441–444.
[66]Lario, D. et al, " Radial and Longitudinal Dependence of Solar 4-13 MeV and 27-37 MeV Proton Peak Intensities and Fluences: Helios and IMP 8 Observations" The Astrophysical Journal, Volume 653, Issue 2, pp. 1531-1544 2006.

with the National Science Foundation (NSF), the NRC, and the National Oceanic and Atmospheric Administration (NOAA) have implemented integrated theoretical and experimental space weather programs to address the issue of characterizing the space weather environment throughout the Heliosphere.

**Mission-threatening acute risks** (radiation sickness, mortality) can be mitigated with shielding mass. Mission-threatening acute risk for the two missions can most likely be leveled with proper design. In past studies, Mars configurations have provided significant shielding inherent to the vehicle due to the large amount of equipment/consumables (subsystems, foodstuffs, water, etc.) that are required for these extended-mission durations (Nealy et al., 1991[67]; Simonsen et al., 2000). There is "no first order discriminator" for a mission-threatening acute risk (radiation sickness or death) between the long-stay mission and the short-stay mission that is *based* on the assumption that a more heavily shielded location in the vehicle achieving approximately 20 g/cm$^2$ of shielding (by design, not parasitic shield mass) can be designed. However, large SPE exposures are still possible during close passage to the sun, leading to other non-mission-threatening acute radiation syndromes such as blood count changes, nausea, and sterility in individuals. This larger SPE dose will also contribute significantly to the cancer risk as discussed below. Dose rate effects are a current area of research.

Likewise radiation carcinogenesis from SPEs can also be mitigated with shielding mass. Although a heavily shielded location in the vehicle achieving 20 g/cm$^2$ (as discussed above) will significantly mitigate this risk, the risk will remain greater for the short-stay mission because of the longer transit times (1.7 times higher likelihood of occurrence) and close passage to the sun (5.6 times larger exposure if $1/R^{2.5}$ is valid for entire SPE fluence energy spectrum), as discussed above. An absolute risk – the event likelihood multiplied by %REID—has not been evaluated.

It is recommended that the DRM transit vehicle habitat module interior layout be analyzed to determine the amount of shielding that can be achieved by design (i.e., no parasitic shielding), including food and other consumables that are brought with the crew. Analysis of shielding as a function of mission phase is important since the shielding from consumables may decrease as the mission progresses; that is, more shielding from consumables is available on the outbound portion of the trip.

**Mars transit vehicles using nuclear propulsion**

To minimize crew radiation exposure on transit vehicles with nuclear propulsion, past and current vehicle designs have maximized the distance between the reactor and the crew compartment (e.g., on long trusses), provided shielding using system/subsystem placement (including large liquid hydrogen ($LH_2$) propellant tanks), and included external biological shadow shields. A previous early study (Willoughby et al., 1990[68]) considered the use of very high thrust/ high thermal power nuclear thermal rockets for transit vehicle primary propulsion for an "all-up" 434-day, round-trip (~30-day stay) Opposition Class Mars mission. The transit vehicle that was analyzed in the 1990 study also assumed that two different high-thrust engines were used: an approximately 250 klb$_f$ thrust/5,000 MW$_t$ engine for TMI and a second approximately 75 klb$_f$ thrust/1,575 MW$_t$ engine for Mars orbital capture and trans-Earth injection (TEI) engines. The estimated crew exposures due to NTR firings were slightly greater than 100 mSv (~10 Rem) with over 90% of the estimated exposure occurring during the final TEI burn as the core propellant tank was drained. The total dose incurred from a nuclear-powered vehicle was approximately 10% of the total dose that was incurred for both the long- and short-stay-time missions (Nealy et al., 1991). The difference in exposure levels between the short- and long-stay mission will be proportional to the mission delta-Vs and the duration of the NTR firing. For the current NTR crewed transit vehicle designs that were considered in the Mars DRA 5.0 Phase 1 and 2 analysis cycles, there no large 250 klb$_f$ thrust/5,000 MW$_t$ engines are used. The single core propulsion module on the crewed transit vehicle uses three 25 klb$_f$ thrust/~350 MW$_t$ engines. For the baseline long-surface-stay Mars mission option and current payloads, the total burn time on the engines is approximately 80 minutes for the TMI (~55 minutes), Mars orbit capture (~15 minutes), and TEI (~10 minutes) maneuvers. The Mars departure delta-V for the "all-up" 434-day, round-trip Opposition Class mission that was analyzed by Willoughby and Nealy was also approximately 2.5 times larger than that used in the current DRA 5.0 study (~3.96 km/s vs. ~1.56 km/s), further reducing the crew exposure dose from approximately 100 mSv to less than 40 mSv. Further refinement of the

[67]Nealy, J.E.; Simonsen, L.C.; Wilson, J.W.; Townsend, L.W.; Qualls, G.D.; Schnitzler, B.G.; and Gates, M.M.: Radiation Exposure and Dose Estimates for a Nuclear-Powered Manned Mars Sprint Mission. 8th Symposium on Space Nuclear Power Systems, Albuquerque, NM, January 1991, pp. 531-536.

[68]Willoughby, A.J.; Stevenson, S.M..; Bolch, W.E.; and Thomas, J.K.: "Astronaut Radiation Safety Evaluated for Combinations of Natural and Man-Made Sources," NASA TM-103138, Lewis Research Center, 1990.

mission payloads, crewed transit vehicle configuration and interior shielding arrangement within the habitat module will help to better define the exposure dose that is attributed to NTR engine operation. Overall, it is expected to be a small percentage (~5%) of the overall dose that is associated with the natural radiation sources.

**Summary of Radiation Exposure and Risk**
**Mission radiation exposures**
A summary of mission exposures (table 3-7) has been estimated based on past analyses of Simonsen et al. (2000), Clowdsley, 2007[69], and Nealy et al., 1991 that compares well with the most recent calculations of Cucinotta et al., 2005[70]. Table 3-7 estimates are for the Mars transit habitat concept (Simonsen et al., 2000). GCR transit exposures assume that crew members spend two-thirds of their time in the TransHab living space, which is lightly shielded, and one-third of their time sleeping in a heavily shielded location providing 19 g/cm$^2$ of protection. An example exposure due to a large SPE event (Aug. 1972) assumes that the crew is in the heavily shielded location. Surface exposures assume protection is provided by the COSPAR low-density atmosphere model at an altitude of 0 km. Surface exposure estimates assume no additional shielding from the habitat or regolith. Organ doses vary only modestly with surface habitat shielding on the Mars surface (Simonsen et al., 1991[71]; Saganti et al., 2002[72], Clowdsley, 2007). Many embedded assumptions are implicit in these values and are noted here as a first-order discriminator only.

**Table 3-7.** Estimates of Radiation Exposure

| Radiation Source | Short-stay Mission Exposure BFO* | | Long-stay Mission Exposure BFO | | Total BFO Short Stay (mSv) | Total BFO Long Stay (mSv) |
|---|---|---|---|---|---|---|
| | Transit Exposure (mSv) | Surface Exposure (mSv) | Transit Exposure (mSv) | Surface Exposure (mSv) | | |
| GCR at Solar Minimum | 1,030–1,240 | 25–30 | 720 | 335–405 | 1,055–1,270 | 1,055–1,125 |
| GCR at Solar Maximum | 475 | 10 to 15 | 275 | 120–175 | 485–490 | 395–450 |
| August 1972 SPE at 1 AU | 60–90 | 25 | 60–90 | 25 | | |
| August 1972 SPE at 1.5 AU | N/A | 9 | N/A | 9 | | |
| Nuclear Propulsion | 100 | N/A | 100 | N/A | | |

BFO = blood-forming organ

For total GCR exposure, the difference between the two missions is indistinguishable based on current information if the dose equivalent (mSv) is compared. In terms of SPE exposure, the most notably difference between the two missions is the large exposure that can be incurred at 0.5 AU for the short-stay mission. The SPE exposure estimates for distances at other than 1 AU assume that the SPE fluence extrapolation is independent of energy; i.e., that the entire spectrum is multiplied by $1/R^{2.5}$, thus the exposure (mSv) is 5.6 times greater at 0.5 AU and 2.7 times less at 1.5 AU.

For a mission during solar minimum conditions without a flare event, exposure estimates for the short- and long-stay mission are *both* on the order of 1,055 to 1,270 mSv compared with a permissible exposure of 250 mSv for a 45-year-old male at a 95% confidence level. For missions during solar maximum conditions with the August 1972 event occurring in transit with 19 g/cm$^2$, the short-stay mission exposure estimate is 825–1,000 mSv (flare at 0.5 AU) while the long-stay mission estimate is 455–540 mSv (flare at 1 AU). The long-stay mission during solar

[69]M.S. Clowdsley. "Radiation on Mars – Exposure Due to Charged Ions and Albedo Neutrons." *presented for National Academies Project* "Evaluation of Radiation Shielding for Lunar Exploration," Meeting #3 in Washington DC. May 10, 2007.
[70]Cucinotta, F.A.; Kim, M.H.; and Ren, L.; "Managing Lunar and Mars Mission Radiation Risks: Part I: Cancer Risks, Uncertainties, and Shielding Effectiveness. July 2005.
[71]Simonsen, L.C.; Nealy, J.E.; Townsend, L.W.; and Wilson, J.W.: Martian Regolith as Space Radiation Shielding. *Journal of Spacecraft and Rockets*, Vol. 28, No. 1, January-February, 1991, pp.7-8.
[72]Saganti, P.B., Cucinotta, F.A., Wilson, J.W. and Schimmerling, W.: Visualization of Particle Flux in the Human Body on the Surface of Mars. *Journal of Radiation Research* 43, 119-124, 2002.

minimum with an estimated exposure of 540 mSv compares more favorably with the 50-year-old male PEL of 303 mSv at the 95$^{th}$ CL. Although current research indicates that the age dependence for career limits is not as great, career PELs for older individuals is expected to decrease (BEIR VII[73]).

**Mission radiation risks**

Previous studies have also quantified mission risk in terms of %REID. Risk estimates for Mars missions representative of the short- and long-stay missions are shown in table 3-8. However, the Mars swing-by mission in table 3-8 assumes no surface stay. Results of this previous study assume the vehicle provided a total of 5 g/cm$^2$ or a total of 20 g/cm$^2$ of protection. For missions during solar minimum, only exposure due to GCR was assumed. For the missions during solar maximum, it was assumed that an SPE (August 1972) occurred at $R$ = 1.0 AU in free space behind either 5 g/cm$^2$ or 20 g/cm$^2$ of aluminum (Al) shielding. For GCR surface exposures, the low-density model of the Mars atmosphere was used with no additional shielding assumed from a habitat structure.

**Table 3-8.** Estimates of Radiation Risk for a 40-year-old Female (Cucinotta et al., 2005)

| Mission | Total Days | Deep Space Days | Surface Days | Assumed Environment | Assumed Transit Shielding | % Fatal Risk (95% CI) | Exposure (mSv) |
|---|---|---|---|---|---|---|---|
| MAT short-stay DRM<br>MAT long-stay DRM | 661<br>905 | 621<br>360 | 40<br>545 | GCR min or max<br>GCR min or max | TBD<br>TBD | | |
| Mars Swing-by<br>Mars Surface | 600<br>1,000 | 600<br>400 | 0<br>600 | Solar min<br>Solar min | 5 g/cm$^2$ Al<br>5 g/cm$^2$ Al | 4.9% [1.4, 16.2]<br>5.1% [1.6, 16.4] | 1,030<br>1,070 |
| Mars Swing-by<br>Mars Surface | 600<br>1,000 | 600<br>400 | 0<br>600 | Solar min<br>Solar min | 20 g/cm$^2$Al<br>20 g/cm$^2$ Al | 3.9% [1.2, 12.7]<br>4.1% [1.3, 13.3] | 870<br>960 |
| Mars Swing-by<br>Mars Surface | 600<br>1,000 | 600<br>400 | 0<br>600 | Solar max, Aug 72<br>Solar max, Aug 72 | 5 g/cm$^2$ Al<br>5 g/cm$^2$ Al | 5.7% [1.8, 17.1]<br>5.8% [2.0, 17.3] | 1,210<br>1,240 |
| Mars Swing-by<br>Mars Surface | 600<br>1,000 | 600<br>400 | 0<br>600 | Solar max, Aug 72<br>Solar max, Aug 72 | 20 g/cm$^2$ Al<br>20 g/cm$^2$ Al | 2.5% [0.76, 8.3]<br>2.9% [0.89, 9.5] | 540<br>600 |

For a mission during solar minimum, the REID from cancer is calculated to be between 4.9%–5.1% for a 5-g/cm$^2$ vehicle with no SPE event. It should be noted that for mission lengths of 2 to 3 years, even during solar minimum, sizeable SPEs can and do occur. It is recommended that the likelihood of SPE occurrence and resulting risk during each mission phase (outbound transit, surface, inbound transit) be evaluated. Fatal heart disease will contribute to %REID by an additional approximately 60%, increasing the estimate to *7.8% to 8% for short vs. long stay*. The 95$^{th}$ CI is well above 16%. Current PELs limit exposure at the 95$^{th}$ CI to 3%. Risk mitigation strategies as well as uncertainty reduction are needed to meet current requirements for missions beyond LEO at 95$^{th}$ CI.

The risk models that were used for this assessment, as recommended by the NCRP, assume a linear response at low fluence or dose resulting in a constant risk per unit dose. However, mission risk may not necessarily be proportional to dose equivalent (mSv) and is dependent on the shape of the dose-response curve. Evidence exists that suggests that the risks are not linear in all cases and that is demonstrable risk from non-targeted and cancer promotion effects. It is well known that particle hits per cell are less than unity for Z>2 ions until mission lengths greater than 1 year occur. For non-targeted effects, neoplastic transformation occurs through aberrant signals in adjacent cells with signals as far as 1 mm observed. Under these mechanisms, risk is not linear with dose and varies as Dose Eq$^{P}$, where P is less than 1. Similarly, if radiation acts to promote the existing pre-neoplastic lesions that are present in adults, the dependence of risk will vary in less than a linear fashion (Cucinotta and Durante, 2006a; Sachs et al., 2005). Because these possibilities would reduce the importance of mission length or shielding, it is an important research consideration for Mars missions and may preclude many conclusions on crew risks.

**Radiation-exposure-related operations**

Radiation operations will focus on crew dosimetry and monitoring, space weather observations, and supporting decisions to minimize crew exposures. General dosimetric measurements that are ambient with the crew with data

[73]BEIR VII. National Research Council of the National Academies, "Health Risks from Exposure to Low Levels of Ionizing Radiation: BEIR VII Phase 2." The National Academies Press Washington, D.C. 2006.

transmission back to Earth will be similar for both the short- and long-stay missions while in transit and on the surface. Although background levels of GCR exposure will be measured and reported, most operational needs will focus on SPE events.

*Transit*. Current SPE forecasting (no occurrence, occurrence, and intensity) is limited. Although there will be a communication time lag between Earth and the transit vehicle, this time difference is on the order of the amount of time that is required to interpret solar observations and formulate a course of action. For the short-stay mission with close passage to the sun and trajectories that are possibly on a different magnetic field line than Earth, there may be advanced exposure on the spacecraft. However, with a heavily shielded location within the vehicle, on-board active monitoring, and crew training, crew members could take the first steps in protecting themselves while the situation is assessed at Earth. Likewise, during the long-stay time transit, advanced exposure is also possible due to location with respect to magnetic field lines, and the same actions could be taken by the crew. The main difference between the two mission types will be the intensity of the event due to the close passage to the sun as, described above.

*Surface Stays*. SPE exposures on the Mars surface are expected to be small (whether on EVA or within a habitat); however, they still contribute to a crew member's total exposure and must be monitored to minimize risk. Therefore situational observations at Mars are required since the communications time lag is on the order of the decisional time (hours) that is required to take action. This could be accomplished with a radiation dosimetry suite on the surface of Mars, as well as a complementary set of instrumentation that is located at Mars L1 with spacecraft telemetry to both Earth and Mars. A trained crew member at Mars could characterize the situation and take appropriate action. With these assumed capabilities provided, there is no "first-order discriminator" in risk for SPEs during surface stays between the two missions.

**Radiation exposure mitigation strategies**

Current exposure estimates are well in excess of baselined permissible exposure limits. Risk mitigation strategies include advanced shielding technologies, countermeasures (radioprotectants and pharmaceuticals), and individual based risk assessments as well as uncertainty reduction. The best solution may be a combination of these mitigation strategies. The NASA HRP and Exploration Technology Development Program (ETDP) are making investments in these areas.

*Advanced shielding technologies*. Excessive shield mass will be prohibitive. However, material selection and multifunctional shield technology can reduce crew exposure. Hydrogen-rich shielding materials and storage technologies as well as multifunctional structural concepts providing structural, thermal, micrometeoroid and orbital debris protection, and radiation shielding should be pursued and evaluated. ETDP is making small investments in this area.

*Countermeasures*. HRP is investing in research to understand and quantify the biological risks of space radiation exposure. This research will support the identification of risks that will require countermeasures and the likely approaches to either select or develop biological countermeasures. A major goal is to develop a quantitative approach to countermeasures since their use operationally will be enhanced if radiation PELs can be adjusted based on the countermeasure. An obstacle is the large number of tissues that contribute to the overall risk to astronauts. There are differences in genetic pathways across tissue types and risks.

*Individual-based Risk Assessment*. Presently, there are many insights into radiation resistance genetic characteristics that could be used to select astronauts. Information in this area is growing exponentially. However, ethical and legal constraints to crew selection for radiation sensitivity are significant. The NASA HRP has contracted the NCRP to write a report on these issues, which should be published in 2009.

*Uncertainty Reduction*. Current PELs are written such that an individual's probability of REID is not to exceed 3% at the 95$^{th}$ CL. Large uncertainties exist for a human Mars mission, and uncertainty reduction must be pursued. A reduction in uncertainty from four fold down to two fold by 2014 is the near-term goal of the HRP to support lunar and Mars mission planning. Further uncertainty reduction to 50% prior to the Mars mission in the 2020 timeframe is the long-term goal of the HRP. Uncertainty reduction in the areas of radiation quality effects on biological damage, dependence of risk on dose rates in space, predictions of SPEs, extrapolation from experimental data to humans, and individual radiation sensitivity are all considered major uncertainties. Other areas of uncertainty reduction include better space radiation environmental models, physics of shielding assessments, microgravity effects, and statistical and/or recording errors in human data.

#### *3.3.12.2 Zero-g countermeasures*

Non-radiation flight phase countermeasures analysis was performed by members of the HRP Human Health and Countermeasures discipline. They made the following assumptions:

- Hypogravity exposure will be the primary driver for untoward effects on humans.
  - Any remaining relevant unknowns with respect to these mission durations will be characterized during the remaining years of the Space Shuttle and ISS Programs.
  - The human responses to long-duration (> 6 months) microgravity exposures (and subsequent transitions back to 1 g) will be investigated, probably by having the appropriate number of crew members stay on the ISS longer than the standard 6-month increment.
  - Biomedical research during and after planned lunar and Mars missions must be relied on to fully characterize human responses to gravitational loading between 0 g and 1 g. Data from lunar sortie missions will inform expectations (through refined risk assessments) for Lunar Outpost missions; data from lunar outpost missions will inform expectations for Mars missions, and data from early Mars missions will inform expectations for later Mars missions.
  - The accuracy of interpolating biological and physiological responses to Mars gravity will be improved substantially after responses to prolonged lunar gravity have been well characterized.
- The potentially detrimental effects of hypogravity and space radiation exposure of humans, food supplies, and pharmaceuticals are not included in this analysis.
- Adequate countermeasures will be in place in the Mars transit habitats and SHABs to offset the primary effects of environmental factors such as isolation, confinement, and altered light/dark cycles.
- Environmental control systems that are built into Mars transit habitats and SHABs can maintain the cabin air (temperature, pressure, composition, microbial content, etc.), water (composition, microbial content, etc.), lighting, acoustic noise, and other factors within acceptable limits.
- The effects of EVA suit design on the performance of crew members will be well characterized in ground-based simulations and during lunar mission operations.
- Sufficient mission resources (up-mass, power, volume, etc.) will be devoted to crew health requirements for crew members to maintain acceptable fitness levels through exercise, to provide access to adequate nutrients and pharmaceuticals, and to accommodate other crew monitoring, countermeasure, and treatment equipment.
- Countermeasure for bone demineralization during weightless transit are assumed to be no more effective than those in use aboard the ISS at the time of this analysis.
- Currently available validated countermeasures for renal stone formation will be used, and crew members will remain sufficiently hydrated to avoid saturation of urine.
- Optimal nutrition will be provided for all aspects of human health.
- Risks of falling in the Mars gravitational field will be effectively mitigated through corrective actions that will be applied to impaired balance, orthostatic intolerance, visual dynamics, neuromuscular coordination, vitamin D deficiency, and other identifiable causes.
- Optimal crew selection will mitigate risk by selecting out predisposing clinical factors for identifiable disease processes.
- Loads that are applied to bone will be sufficiently reduced or avoided by engineering (EVA suit design, robotics, etc.) to minimize bone fracture risk.
- Crew members will comply fully with all countermeasure prescriptions.
- Risks after return to Earth are acceptable due to the availability of medical treatment.

The short- and long-stay options were analyzed both without and with the artificial gravity that will be available during the transit phases, with the characteristics listed in table 3-9.

**Table 3-9.** Assumed Characteristics of Artificial Gravity

| | |
|---|---|
| **Infrastructure** | • Whole vehicle is rotated for majority of time in transit<br>– Provides chronic crew exposure<br>– 1-g transit will be as effective as 1 g on Earth in preventing space flight effects<br>• Improves habitability and allows simplification of crew equipment<br>• Crew is tolerant of 0 g for reasonable intervals |
| **Radius of Rotation** | • 25–56 meters (depending on rotation rate) |
| **Rotation Rate** | • 4–6 rpm (for 1 g) |
| **G-level** | • 1 g during majority of transit periods<br>• Gradual decrease from 1 g to 3/8 g before Mars arrival<br>• Gradual increase from 3/8 g to 1 g after Mars departure |
| **Impact of Transition** | • Inevitable<br>• Acceptable<br>– Within experience base (space shuttle, ISS) |

Without artificial gravity, neither option has a clear advantage (table 3-10). With artificial gravity, both short and long stays carry less risk, but the short-stay option has an advantage because the overall mission duration is shorter (table 3-11) and the time that the astronauts are exposed to the assumed deleterious effects of hypogravity (3/8 g) on Mars is only 6% of that in the long-stay option.

**Table 3-10.** Physiological Countermeasures without Artificial Gravity

| | Short Stay | Long Stay |
|---|---|---|
| Advantages | • Shorter overall exposure to g < 1<br>– No data on 3/8-g physiology | • Shorter overall exposure to 0 g (12 months)<br>• Functional recovery time on Mars (probably ~6–10 days after 6-month transit) is small fraction of surface stay time |
| Disadvantages | • Longer overall exposure to 0 g (20 months)<br>• Functional recovery time on Mars (possibly up to ~10–20 days after 10-month transit) is unacceptable fraction of surface stay time | • Longer overall exposure to g < 1<br>• No data on 3/8-g physiology |

**Table 3-11.** Physiological Countermeasures with Artificial Gravity

| | Short Stay | Long Stay |
|---|---|---|
| Advantages | • Shorter overall mission duration<br>• Low to no deconditioning outbound<br>• Recovery (nearly) complete before return to Earth | • Low to no deconditioning outbound<br>• Recovery (nearly) complete before return to Earth |
| Disadvantages | • Non identified | • Longer overall exposure to g < 1<br>– No data on 3/8-g physiology |

The following knowledge gaps were identified, to be resolved through goal-directed research (**Note:** these are consistent with the risks and gaps that were identified in the HRP Program Requirements Documen (PRD): HRP-47052 Rev. A).

- Use computer modeling to evaluate the ability of loads that are applied by performing mission tasks to affect fracture risk.
- Accurately assess stress fractures and vertebral compression fractures of the spine after space flight.
- Investigate the impact of low radiation doses on bone and osteogenic cells in bone marrow.
- Define the nutrient requirements for space travelers, and determine how these requirements relate to physiologic systems.

- Determine the effects of non-nutritional countermeasures on nutrition.
- Compile a reference database on drug effectiveness in space.
- Determine changes in bioavailability, pharmacokinetics (PK), and pharmacodynamics that occur in critical-care medications in space flight, and establish an optimal steady-state PK for chronic care medications.
- Establish stability and shelf life of acute and chronic treatment medications.

#### *3.3.12.3 Medical care and environmental health*

The Medical Care and Environmental Health analysis was performed by members of the HRP Medical Capabilities discipline. They assessed the short-stay option to have a slight advantage because the total mission length would be less and crew members would have less exposure to mission risks (table 3-12). However, the team identified a potential risk of over-subscription of the 1-month surface stay mission tasks, to more fully justify the resources that would be expended on any astronaut mission to Mars that would not be present in an 18-month Mars surface stay. Any such over-subscription might increase the probability of crew injury or illness through over-extension of the highly motivated and highly tasked astronaut team.

**Table 3-12.** Assessed Advantages and Disadvantages of Medical Care Capabilities

| | Short Stay | Long Stay |
|---|---|---|
| Advantages | • Less overall mission exposure | • More likely to have robust, autonomous surface medical capability (Level V)<br>– Goal: Make Mars the second safest place in the solar system |
| Disadvantages | • More demand on transit medical capability<br>• Lack of medical capability to deal with increased risk of acute SPE | • Longer total mission exposure<br>– Higher likelihood of medical contingency<br>• More EVA, surface construction, maintenance<br>– Higher likelihood of injury due to more mission exposure |

The Medical Care analysis did not identify specific knowledge gaps for resolution through goal-directed research beyond those that were identified by the other disciplines; however, there are technology development needs to enable the capability that is required by the Spaceflight Health Standards Document. The medical capability to manage a large, acute radiation exposure is currently beyond the scope of space flight medical care, especially if extreme measures such as bone marrow transplantation would be required. However technology development in this arena during the next 20 years may allow this treatment capability, or even development, of effective countermeasure performance to progress considerably. The acute solar exposure risk that is posed by the short-stay mission may be the highest unmitigated risk from a CHP Team perspective, aside from the accepted risk of catastrophic vehicular launch and entry/landing failures. However, the radiation risk from an SPE can be greatly reduced within a heavily shielded location of the vehicle (on the order of 20 g/cm$^2$) and by avoiding close passage to the sun.

The HRP PRD (HRP-47052, Rev. A) documents the need for adequate medical resources to treat as many spontaneously occurring medical conditions as may reasonably be expected to occur in highly fit, carefully screened astronauts during up to 30 months away from Earth for both transit and surface mission phases. Also, the need to have Mars surface capability to manage a host of (1) occupational, (2) environmental, and (3) exploration-associated traumatic injurys will be essential to limiting the mission impact of those expected contingencies. The likelihood of environmental conditions such as decompression illness will not be substantially higher in a long-stay mission, aside from the increased number of EVA exposures, due to a likely more relaxed pace of operations and strict adherence to procedures, and will therefore trigger a lower likelihood of a task-rich, schedule-induced events. One exception may be micrometeroid impacts, the probability of which increases directly with surface dwell time, especially the amount of time the crew is outside the habitat on EVA, which will increase the risk of crew injury from micrometeoroid collision with the suited astronaut. Toxic exposures and life support system failures leading to hypercarbia, hypoxia, etc. are addressed in the environmental section, although these clearly can be influenced by having the robust and redundant hardware systems that are more likely to be present on long-stay missions. Repetitive use injuries, especially during EVA, and dust exposure effects are obviously more likely as the cumulative exposure duration is prolonged; thereby driving the need for better countermeasures and mitigation strategies on the long-stay mission for these conditions.

As shown in table 3-13, the likelihood of having fully capable Level V medical care is much higher on a long-stay mission due to the relative resource allocation to the medical system that would be justified based on probability of occurrence. So, the medical contingency response and medical autonomy during a long stay may be relatively superior, thus reducing mission impact when the contingency event occurs. Therefore, technology development in medical diagnostics and care provision to make it lighter and smaller without sacrificing capability is a clear-cut need in the medical system, regardless of which mission scenario is chosen.

#### *3.3.12.4 Human factors*

The Human Factors analysis was performed by members of the HRP Behavioral Health and Performance (BHP) discipline. This analysis made the following assumptions:

- The SHAB will not be the same for long- and short-stay strategies.
- The transit habitat will be smaller than the SHAB.
- Crew size will be six members.
- At least one of the crew members will be a physician, and other crew members will be cross-trained as emergency medical technicians.
- Assessment, quantification, and characterization of all BHP risks (as documented in HRP-47052, Rev. A) will have been determined, and acceptable limits will have been established for the Mars mission.
- BHP countermeasures, monitoring tools, guidelines, and health standards will have been validated for the Mars mission on the ISS and the Lunar Outpost, and in other appropriate analogs on Earth.
- Selection of the Mars crew is based on validated criteria.
- Mars crew training and selection includes protocols involving extreme environments and lengthy durations.
- Mobile robots are used on the surface to reduce astronaut workload and risk and to enable exploration objectives.
- Effective microgravity countermeasures will be used in transit to resolve deconditioning issues.
- Appropriate supplies are pre-deployed or otherwise available in space and/or Mars.

The short-stay option has the advantage over the long-stay option because the total mission length is less (table 3-13).

**Table 3-13.** Advantages and Disadvantages for Behavioral Health and Performance Discipline

| | Short Stay | Long Stay |
|---|---|---|
| Advantages | • 22-month total mission duration is closer to boundary of human experience base (1 Russian cosmonaut flight: 14 months; 6 cosmonauts: 6–14-month flights)<br>• Shorter mission presents far less risk of psychiatric or behavioral condition emerging<br>– Based on Antarctic experience, mission stress curve increases linearly with time<br>– Shorter exposure to and less entrainment required for martian solar day | • Less time in confined transit vehicle<br>• More EVA opportunities<br>• Less schedule stress during surface period (based on historical considerations) |
| Disadvantages | • Poorly understood risk of CNS damage possibly leading to cognitive, behavior, learning, and memory changes due to increase exposure to free space, heavy-ion environment | • 30-month total mission duration is outside of human experience base (1 Russian cosmonaut flight: 14 months) |

In addition to the risks and gaps that were identified in HRP-47052, Rev. A, the Human Factors analysis identified a specific knowledge gap for resolution through goal-directed BHP research:

- Determine the risk of neurobehavioral effects of greater GCR exposure during longer transits and greater SPE exposure during close perihelion passage. (Note that this is analogous to the radiation CNS gap on the effects of heavy ions on the CNS.)

Neither the short- nor the long-stay option would likely pose lower health, safety, and performance risks because both scenarios pose significant health risks. The short-stay option has about 73% of the mission-duration hazard potential of the long-stay option because it requires less time away from Earth (22 months vs. 30 months), although many – but not all – of the health risks increase most rapidly early in flight.

The Earth-to-Mars transit time for the short-stay mission is at the limits of the human space flight experience base, but the Earth-to-Mars transit time for the long-stay mission is well within the experience base. For some disciplines, assumptions determine which option has lower risk: the short stay is preferred by Physiological Countermeasures only if artificial gravity is used as a countermeasure, and the short stay is preferred by Human Factors and Habitability only if an SHAB is not available for the long stay. The experience base for surface time is now very limited, but should be increased by Lunar Outpost experience; this represents future mitigation of a risk.

The short-stay option would likely pose slightly less technical risk because, in the specific absence of a surface habitat, less new habitation technology would be needed and less mass and volume delivery would be required.

### 3.3.13 Long-/short-mission mass comparison

The comparison of total mission mass, which is commonly referred to as initial mass in low-Earth orbit (IMLEO) for both the long-stay (Conjunction Class) and short-stay (Opposition Class) mission is shown in figure 3-19. These estimates were derived from the integrated architecture assessments for both the NTP and chemical propulsion transportation systems options. It must be noted that the short-stay estimates excluded the mission opportunity of 2041 due to the excessive interplanetary translational propulsive requirements (delta-V) for that specific opportunity. Although this violated one of the governing ground rules of the study (i.e., *GR-105: The architecture will support any mission opportunity to Mars*), the analysis team felt that elimination of this specific opportunity was warranted due to the unreasonableness of the resulting vehicle size. As can be seen from figure 3-19, the variation in total mission mass is essentially the same between the options that were analyzed (<10%). The short-stay missions require fewer elements (no SHAB), but require additional interplanetary propulsion on the order of 3 to 7 km/s depending on the mission opportunity across the synodic cycle. On the other hand, long-stay missions use more energy-efficient trajectories, but require more mission elements: namely, an SHAB and surface exploration systems as well as an additional lander to land those systems. It is also important to understand how these mission approaches impact overall system designs. As can be seen from figure 3-19, the long-stay missions enable the opportunity for development of a common vehicle design between the cargo and crew variants. The difference in cargo and crew vehicle sizes is on the order of 10% to 15% for the long-stay mission options, but there is essentially a 50% difference in vehicle size for the short-stay missions. The ability to develop a single vehicle design to support both crew and cargo missions provides a clear advantage to the long-stay mission approach.

When comparing total mission mass it is also important to address the relative sensitivity of the architecture to variations in overall implementation approach. This architectural sensitivity is generally described in terms of a "gear ratio" where the total architectural mass is measured in terms of the change in mass at specific points in the mission profile. For instance, an additional kilogram delivered to Mars orbit will result in more than 1 kg in Earth orbit at the beginning of the mission. Understanding the sensitivity, or gear ratios, is important since it provides a measure of stability of the architecture to change in system design, mass, payload mass, or technology effectiveness. Architectures with smaller gear ratios are relatively less sensitive to change and, thus, result in less overall implementation risk. The gear ratios for both the long- and the short-stay architectures, including the assessment of nuclear and chemical propulsion, are provided in figure 3-20. The propulsive efficiency of NTR propulsion (900 seconds vs. 473 seconds specific impulse ($I_{sp}$)) results in lower overall architecture sensitivity. Likewise, the long-stay architecture results in lower gear ratios, and lower architectural sensitivity, due to the lower overall interplanetary propulsive requirements.

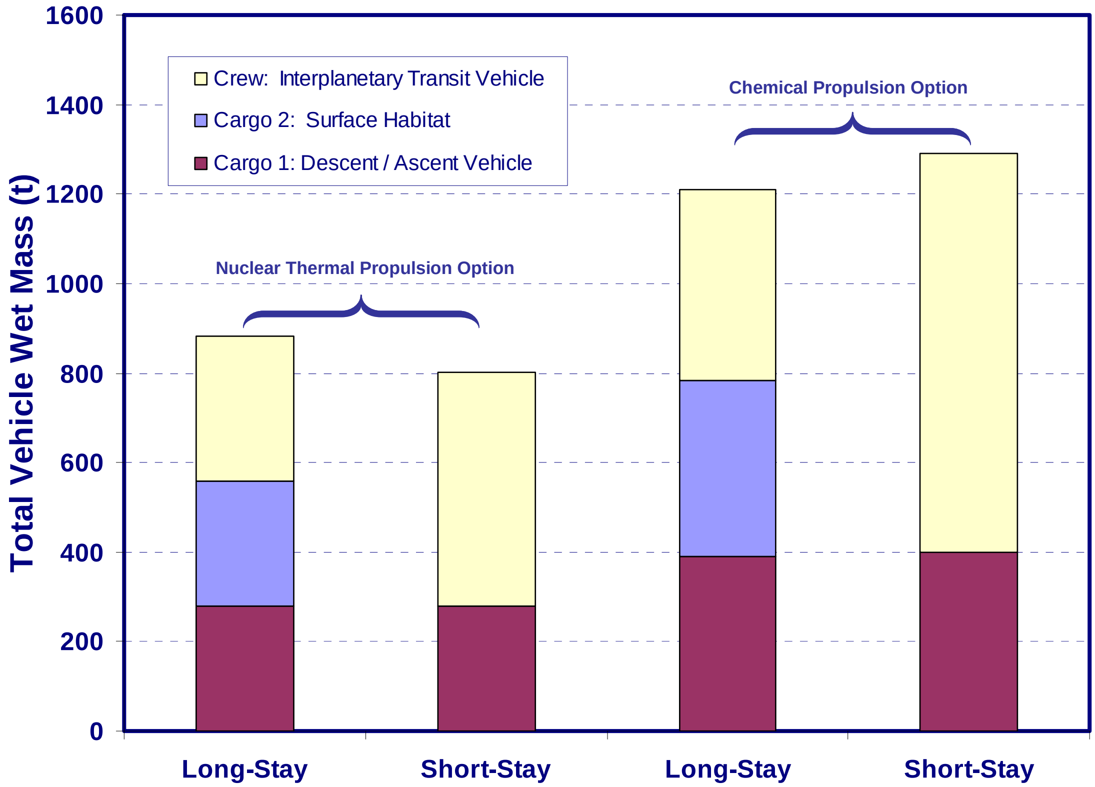


**Figure 3-19.** Long/short total mass comparison.

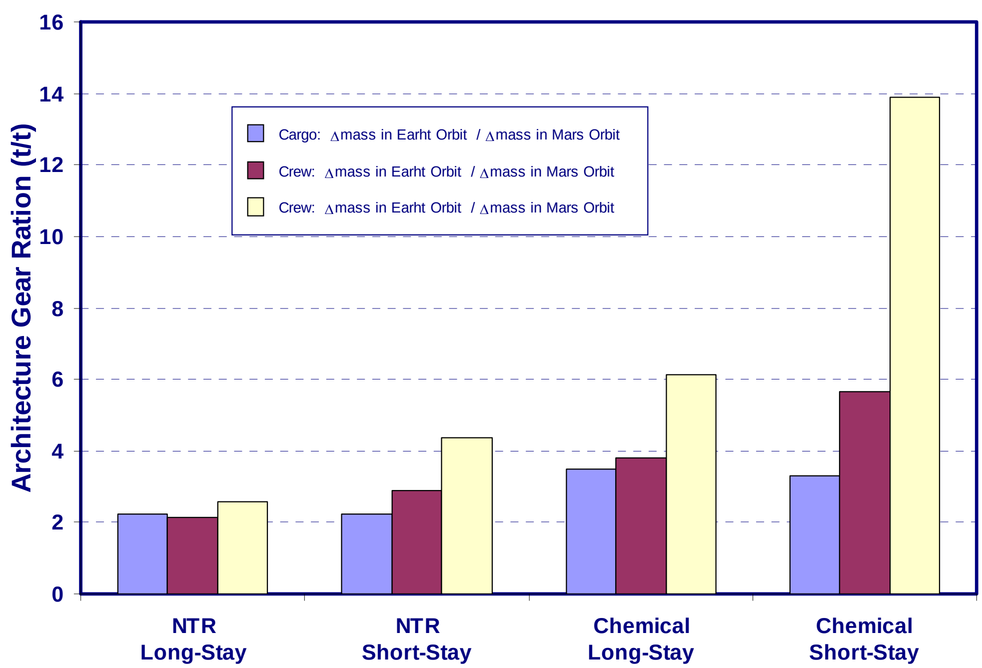


**Figure 3-20.** Long/short architecture sensitivity comparison.

### 3.3.14 Long-/short risk comparison

Assessments of the architectural crew safety (probability of loss of crew) as well as the mission success (probability of loss of mission) were conducted for both the long- and short-stay architectures. These comparisons must be considered first-order assessments due to the relative uncertainty resulting from the immaturity of the system concepts that are under consideration. End-to-end mission models were developed using “best” known data to date including space shuttle and ISS histories. These models were also developed “as is,” with no credits taken for flight demonstrations (e.g., large-scale EDL) or other architectural activities (e.g., lunar). This process thus gives an adequate apples-to-apples comparison of the two mission classes that are under consideration.

Although the short-stay missions appear to provide less overall loss of mission, there is no clear advantage given the maturity of the understanding of the systems to date (figure 3-21). Due to the longer mission duration of the long-stay mission approach, overall system reliability is a driver of mission success. Gaining better understanding of the system performance for long periods is necessary to reduce loss of mission. Technology and system demonstrations on the ISS and lunar programs provide a vital link to reducing this risk.

Since the initial comparative risk models did not include flight demonstrations or the lunar program as risk mitigation steps, first use of the EDL system as well as overall system reliability are key contributors to crew safety. In addition, close perihelion passage, which is necessary for the short-stay mission approach, becomes a crew risk driver. The initial risk results indicate that the short-stay missions decrease the duration of equipment reliability, but increase the number of Ares V launches. Certain elements are reduced with no SHAB, but cause a lack in maturity leading to greater risk for crewed missions (i.e., EDL). Equipment reliability can be enhanced by scavenging techniques when a crew is present. These techniques can be learned during lunar missions.

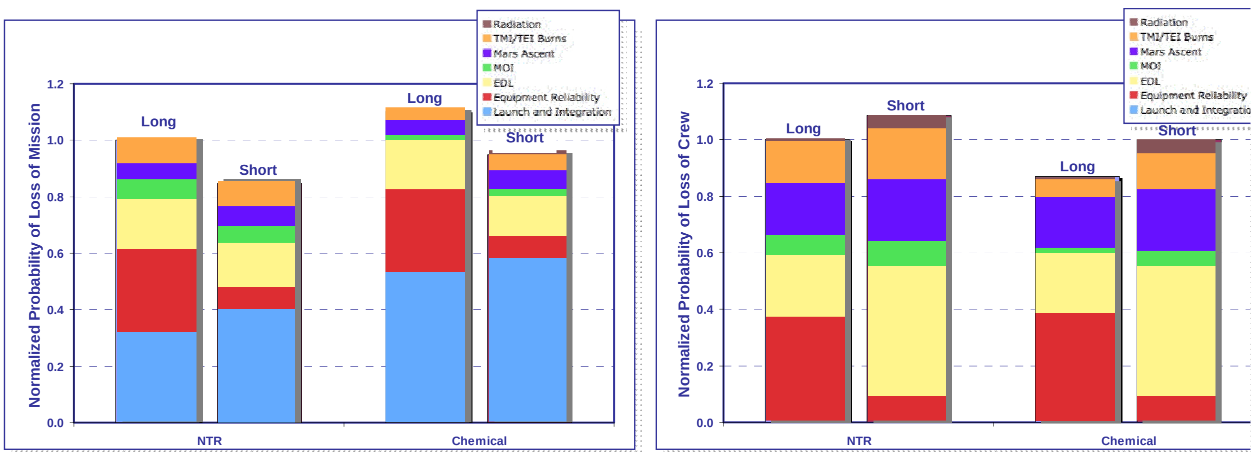


**Figure 3-21.** Long/short risk comparison.

### 3.3.15 Long/short cost comparison

For the short vs. long stay, the difference in cost (figure 3-22) is due predominately to the surface systems including the development and recurring cost of the extra SHAB, the recurring cost of an extra descent stage, the long-duration rover, the additional scientific equipment, etc. There is some uncertainty in the magnitude of the difference as some of these systems are not well-defined yet.

The cost difference in the flight systems is swamped by the cost difference in the surface systems. This is due to the modular nature of the MTVs and the similar number of total launches and flight elements. Even so, there is a slight cost savings for the short-stay flight systems and launch costs. Cost of the surface systems for the long-stay missions may be further reduced depending on the commonality with lunar systems and lunar technology development activities.

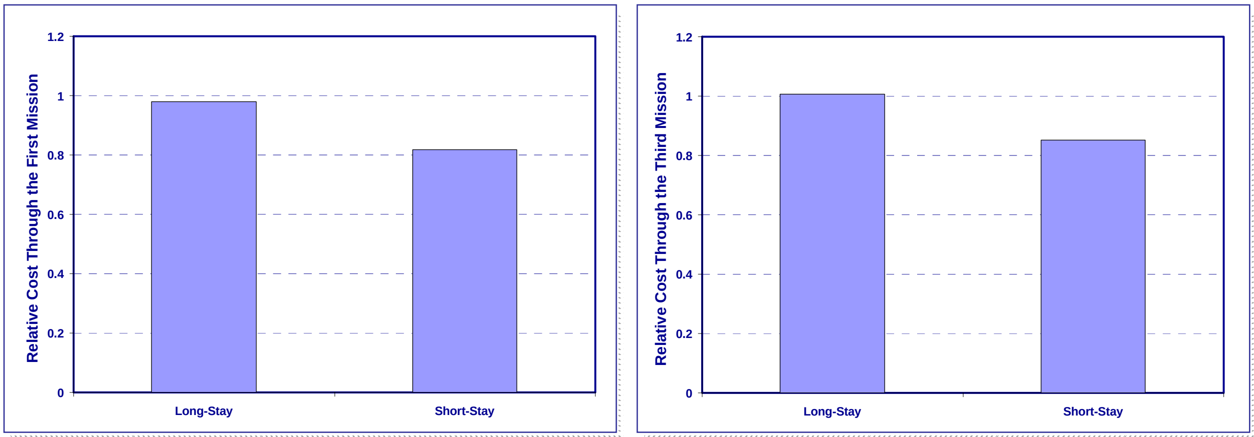


**Figure 3-22.** Long/short cost comparison.

### 3.3.16 Long/short mission recommendation

A summary of the overall FOMs that were considered for the long-/short-mission mode decision are shown in figure 3-23. These results were discussed with the Agency Steering Group on July 23, 2007. After deliberating the results, the Steering Group concurred with the recommendation of proceeding with the long-stay (Conjunction Class) mission approach. As can be seen from this figure, most of the FOMs favor the long-stay approach, with the exception of overall mission duration and slight cost advantage. This recommendation is based entirely on our collective current understanding of system and concept performance at this time. As data are obtained and additional missions are conducted, this decision could be readdressed if warranted.

**Human Exploration Of Mars**

| Long Surface Stay (Conjunction Class) | Figure of Merit | Short Surface Stay * (Opposition Class) |
|---|---|---|
| Similar | Total mass in Low-Earth Orbit (mt) | Similar * |
| **45% Smaller** | LEO Complexity / Size of Crew Vehicle | Larger |
| **~3100 crew-sols** | Expected Useful Crew Sols on Surface (mission return) | ~80-500 crew-sols |
| **Best** | Exploration Goal Satisfaction (range, depth, frequency) | Lower |
| **3 / 6 kg/kg** | Architecture Sensitivity (gear ratios: NTR/Chem) | 4 / 13 kg/kg |
| No Clear Advantage | Probability of Loss of Crew | Somewhat Less |
| Somewhat Less | Probability of Loss of Mission | No Clear Advantage |
| 950 | Total Mission Duration | **650 days** |
| **500 sols** | Mission Flexibility (contingency replanning) | Few sols |
| **Less** | Crew Health Risks from Radiation Exposure | More |
| **200** / 500 / **200** | Crew Exposure to Zero-G (days out / surface / back) | 180 / 30 / 360 |
| Available | Backup Lander and Surface Habitat | None |
| Somewhat More | Cost Through First Mission | Slight Advantage |
| Somewhat More | Cost Through Third Mission | Slight Advantage |

**Figure 3-23.** Long/short figure of merit summary.

## *3.4 Mars Cargo Deployment (All-up vs. Pre-deploy)*

A nominal human Mars mission will require assets to accomplish three major mission phases – outbound transit from Earth to Mars, mission activities in the vicinity of Mars and on its surface, and inbound transit from Mars to Earth. Under these nominal conditions, not all of the mission assets are needed or used by the crew during the outbound phase of a mission. Examples of these assets include all of the systems that are used on the surface (including habitation), the vehicle that is used for entry and landing as well as launch from the surface, and any ISRU equipment (if used in a particular scenario).

This deferred need opens the option of sending some of these assets on an earlier, typically more energy-efficient trajectory and, thus, delivering more of these assets (measured in terms of mass) for the same amount of propellant (as compared to that used by the crew) or delivering the nominal assets (mass) for less propellant and associate launch vehicles. This approach has become known as the “split” or “pre-deploy” mission approach.

There are pros and cons that are associated with the pre-deployed and the all-up options, not all of which are performance related. The significant pros and cons have been identified and are examined in this section.

### 3.4.1 Pre-deployed cargo mission option

The trajectories and flight times that were associated with the so-called “short-stay” and “long-stay” mission options were described previously. Figure 3-24 shows the implications of a pre-deployment strategy for both the short- and long-stay missions. A first human mission in the 2030 timeframe along with the subsequent human missions is used to illustrate campaign-level implications of this strategy.

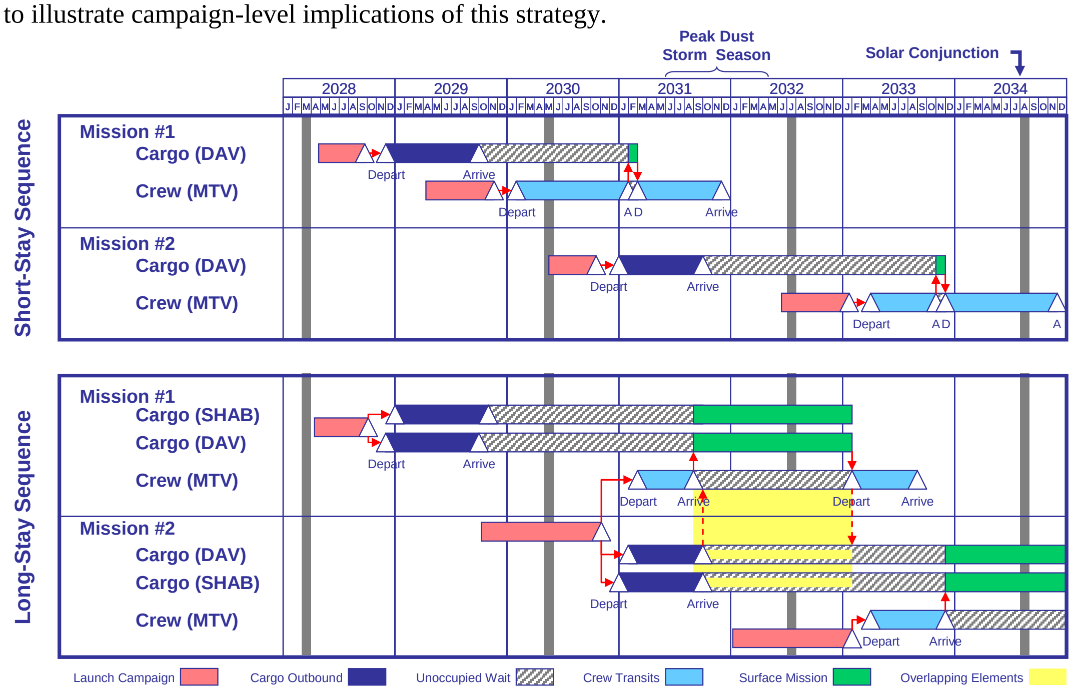


**Figure 3-24.** Mission and trajectory comparison chart – pre-deploy case.

For all of these illustrative cases, it has been assumed that launches from KSC will be spaced at 30-day intervals. Assessment by KSC personnel indicates that this launch interval could be sustained using currently planned facilities without augmentation. In addition, 60 days of schedule margin have been inserted prior to the actual launch window.

For the short-stay mission sequence, the only asset to be pre-deployed is the DAV. This is sent to Mars on the first minimum-energy trajectory prior to the crew launch on an opposition trajectory. The DAV arrives at Mars before the crew launches from Earth, allowing time to confirm that the it is in its proper orbit and functioning normally (implying that there is time to “abort” the mission by simply not launching the crew should there be an issue with the DAV at this phase). The DAV is now placed into a minimal operating configuration and remains in this state for more than 1 year before the arrival of the crew. While the first crew is in transit to Mars, the launch campaign for the second crew's DAV begins. This DAV is in transit to Mars while the first crew carries out its Mars surface mission and begins the return to Earth. This second DAV arrives at Mars and is similarly positioned and checked prior to the departure of the second crew. This DAV waits in its orbit for approximately 2 years prior to the arrival of the second crew. This is a significantly longer wait than that which was experienced by the first DAV, but the variability from mission to mission is typical of the short-stay mission opportunities. Each crew relies on its own DAV for completion of its mission. In addition, launch windows for this combined use of Conjunction Class and Opposition Class trajectories is such that there is no overlap in the launch campaign at KSC.

For the long-stay mission sequence, two cargo elements are pre-positioned to support the crew's surface mission: the DAV and an SHAB with other surface equipment. Both of these elements are launched in the same minimum-energy opportunity just over 2 years prior to the launch of the crew. The launch campaign for the first two cargo elements begins approximately 8 months prior to the opening of the launch window. The cargo elements arrive at Mars approximately 8 months later and are placed into the appropriate orbit (for the DAV) or at the surface location (for the SHAB). They are checked for proper function and then placed into a minimal operating configuration to remain in this state for more than 2 years before the arrival of the crew. The next minimum-energy window (for the next cargo elements) opens shortly before the fast-transit trajectory window for the first crew, but these launch windows are still close enough that a combined launch campaign at KSC is required. This launch campaign for the second crew's cargo and for the first crew begins as much as 1 year before either windows open so that all of these elements are ready for their respective departures. The first crew arrives before the cargo elements for the second mission and nominally uses the assets that were launched over 2 years previously. However, should either the DAV or SHAB suffer a failure between the time the first crew launches from Earth and when it leaves Mars to return to Earth, the second set of cargo elements can be used, thus potentially preventing loss of mission or of the crew. This is a unique feature of the pre-deployment strategy when applied to the long-stay mission; this overlap of assets is not available for any of the short-stay options or for the all-up strategy.

### 3.4.2 All-up mission option

Figure 3-25 shows the implications of an all-up strategy for both the short stay and long stay missions. A first human mission in the 2030 timeframe along with the subsequent human missions illustrates campaign level implications of this strategy as well.

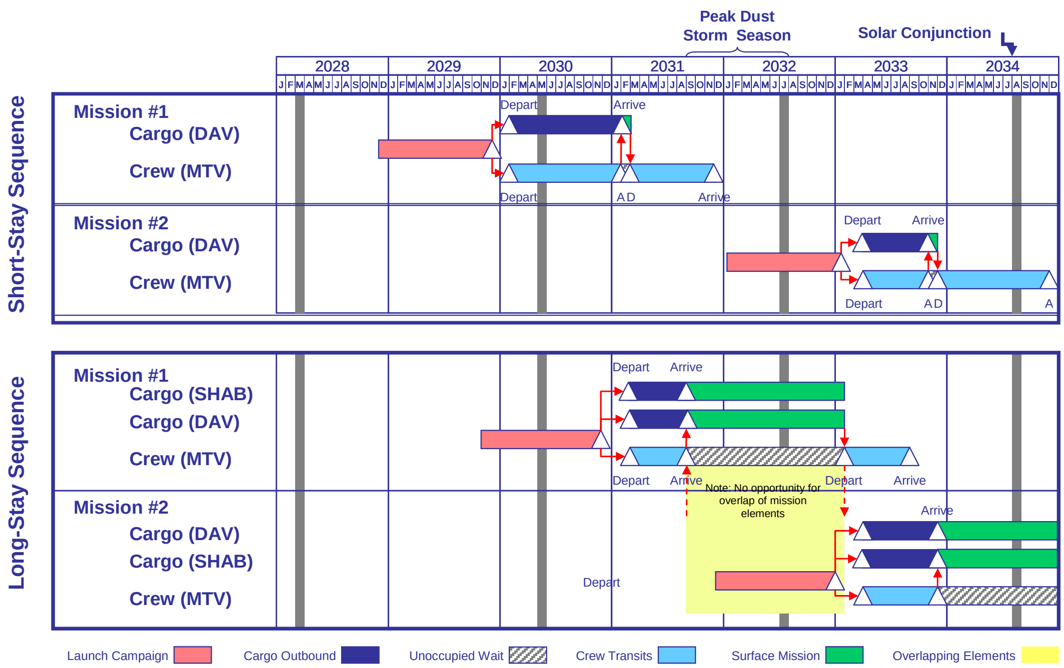


**Figure 3-25.** Mission and trajectory comparison chart – all-up case.

As discussed previously, it has been assumed that launches from KSC will be spaced at 30-day intervals. Assessment by KSC personnel indicates that this launch interval could be sustained using currently planned facilities without augmentation. In addition, 60 days of schedule margin have been inserted prior to the actual launch window.

For the short-stay mission sequence, the only cargo element that is required is the DAV. This element is launched on the same trajectory as the crew in their MTV. The combined launch campaign at KSC for both of these elements begins approximately 1 year prior to the launch window, based on the launch rate that is mentioned above. Both elements arrive at the same time, and the nominal surface mission is carried out. KSC initiates the launch campaign for the second mission approximately 2 years after it completes the first campaign, although there is some variability in this interval due to the natural spacing between the short-stay trajectory opportunities.

For the long-stay mission sequence, two cargo elements are required to support the crew's surface mission: the DAV and an SHAB with other surface equipment. All of these elements are launched on a fast-transit trajectory so that they all arrive at Mars at the same time. While it is conceivable that all of these elements could be integrated into a single stack while in LEO, the total mass of such a stack would be quite significant (i.e., in some cases equivalent of several ISSs) and likely difficult to control. The total thrust that would required to avoid significant gravity losses during departure also makes this approach less desirable. The alternative – three closely spaced departures from LEO during the same launch window followed by a rendezvous (but not necessarily docking) in interplanetary space – is also not trivial but is considered manageable, and thus would be the preferred approach for this option. The KSC launch campaign begins approximately 1 year before these elements depart for Mars; this is similar to the situation that was described for the pre-deploy strategy. The launch campaign for the next mission begins approximately 1 year after completion of the first campaign. There is no overlap at Mars of the two crews or their equipment

### 3.4.3 Cargo deployment advantages and disadvantages

#### *3.4.3.1 Advantages of the pre-deploy mission option*

Pre-deploying cargo elements means that a minimum-energy trajectory can be used. This minimizes the in-space propulsion requirement, but also results in a longer period of time for the transfer to Mars. Given that there is no crew on board, this is not a significant penalty for these systems. By pre-deploying these elements, it is possible to verify that they have been placed in their proper orbit or surface location and that they are operating nominally prior to the crew leaving Earth. Should anomalies occur during this phase of the mission, time is available to attempt to correct them without losing valuable crew time. For the surface elements, it is also possible to begin using these assets for useful tasks at the surface site. Examples include autonomous setup of surface infrastructure (e.g., power plants or ISRU facilities) and verification of sites that are chosen for crew activities (e.g., the DAV landing site, early traverse sites, special access [planetary protection significant] sites prior to crew arrival, etc.). The time that is available before the crew arrives can also be used to create large quantities of ISRU commodities in an efficient manner as well as to verify their successful production prior to crew departure. Use of multiple launch windows to support a single mission provides the opportunity to spread out the launch campaigns at KSC and, thus, avoid building additional facilities to handle periodic but significant processing and launch surges. Pre-deploying these cargo elements for the long-stay mission option also provides the unique opportunity for crew members to use the assets that were pre-deployed for the following crew in a contingency situation on their own mission.

#### *3.4.3.2 Disadvantages of the pre-deploy mission option*

Pre-deploying assets, by definition, requires that the assets remain fully functional for longer periods of time (in some cases, double the total lifetime compared to the all-up option) to complete the entire mission. This introduces reliability issues and associated risks to both crew safety and mission success. These issues could, in some cases, be mitigated by repair or periodic maintenance by the crew. However, the crew will not have access to these assets for approximately the first half of the entire mission. In addition, were these assets taken with the crew, they could provide an "Apollo 13"-style safe haven for the outbound portion of the mission.

#### *3.4.3.3 Advantages of the all-up mission option*

For this option, all assets that are used by the crew are developed, checked out, and launched during the same launch window. This means that all assets will have a similar operating lifetime requirement, alleviating some of the total operating life reliability requirements on these systems. Although it is unlikely that all of these assets will be integrated into a single stack for the outbound flight, the potential exists for these elements to rendezvous during the outbound trajectory, thereby giving the crew access to all mission assets for this phase of the mission. This includes the potential for an “Apollo 13”-style use of these assets should a significant contingency occur.

#### *3.4.3.4 Disadvantages of the all-up mission option*

The all-up option requires that all mission elements – crew and cargo – fly on the same outbound trajectory to Mars. This trajectory is typically a shorter-duration and, thus, more energetic trajectory, resulting in larger in-space transportation systems and an overall increase in mass launched to LEO. Once deployed, all mission activities, including the production of ISRU commodities, must be accomplished during the time period when the crew is deployed; there is no opportunity to conduct preliminary investigations (with robotic assets) or verify successful completion of any ISRU production before the crew has been committed to flight. Similarly, when the crew members depart Earth, they are taking all mission assets with them; there is no opportunity to use assets that are nominally deployed for the next mission to support the current mission. (**Note:** This option exists only for the long-stay-type mission and does not improve the crew safety, just the mission success, because any crew has the option to leave the surface and return to its orbiting transfer vehicle at any point in the surface mission. In addition, exercising such an option would automatically result in deferring the next mission due to lack of assets at Mars, thus diminishing the overall productivity of the campaign.) Finally, launch of all assets places additional pressure on processing and launch facilities at KSC, although this may be no more difficult than that experienced by the demands of some long-stay options.

### 3.4.4 Cargo deployment options assessments

Two branches of the study trade tree were examined in more detail to address the key issues associated with this trade off (see figure 3-26). Both of these options are on the “long stay” branch of the trade tree. This choice was made because experience has shown that the trends will be similar for the “short stay” branch of the trade tree. The specific cases examined were those numbered 10, 12, 22, and 24 at the bottom of the trade tree.

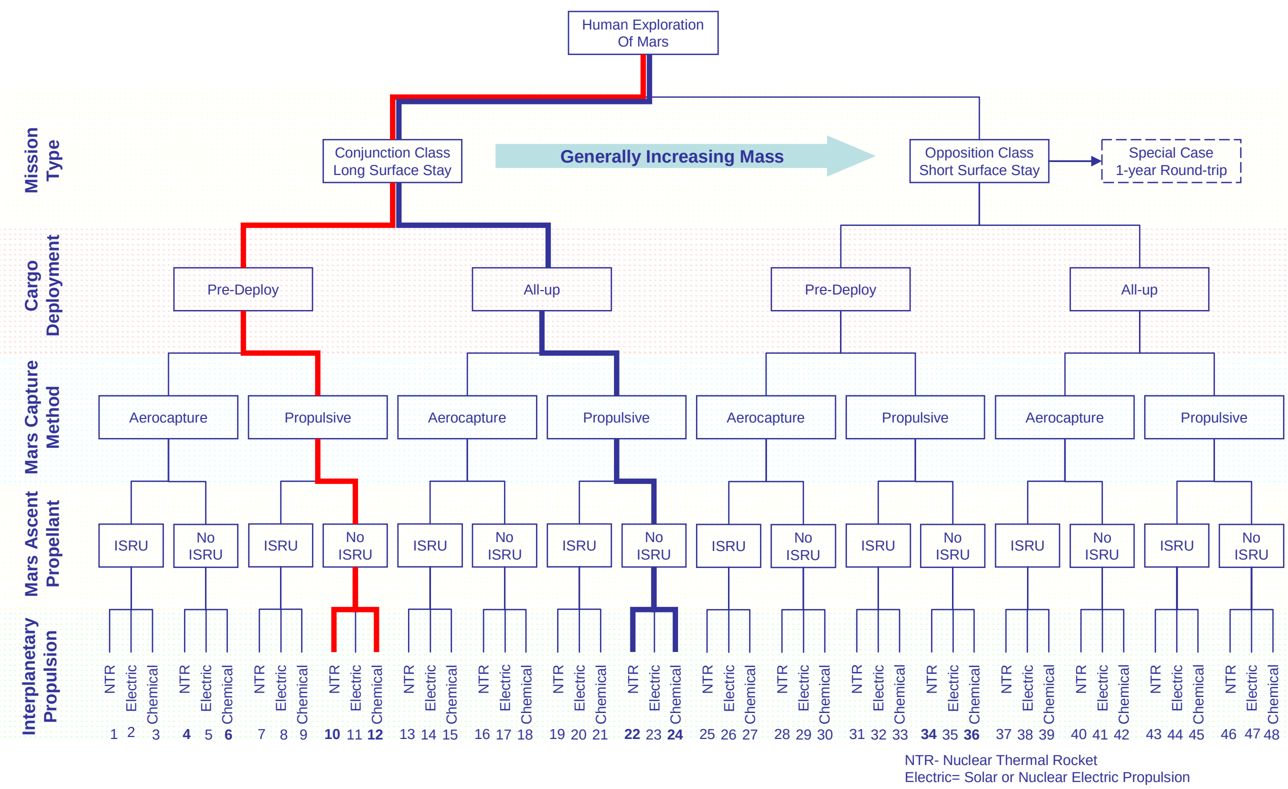


**Figure 3-26.** Pre-deploy trade tree.

For both the NTR and chemical transportation system options, the pre-deploy option consistently shows better overall mass performance (in terms of total IMLEO) by factors of 5% to 10% (see figure 3-27).

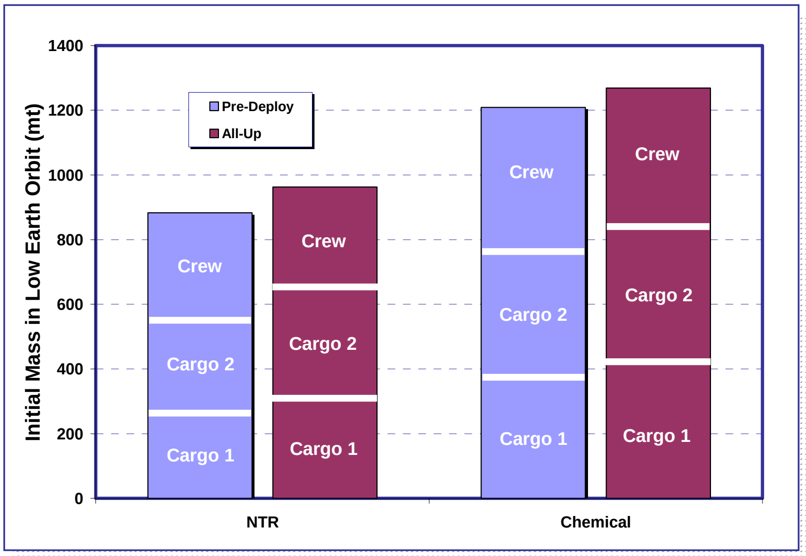


**Figure 3-27**. Pre-deploy total mass comparison.

There were areas in which the all-up option shows some advantages, one example being better use of the Ares V lift capability. (The chemical transportation option assumed development of a single upper stage and use of multiple copies of this common stage to meet a particular delta-V requirement. In this case, that common stage was able to use more of the payload lift capability of a single Ares V than a comparable common stage that was optimized for the pre-deploy option.)

However, the total IMLEO trend held for all of the options that were considered in this analysis and is expected to hold for other trade-offs between these two options. Thus, the mass performance comparison favors (although slightly) the pre-deploy option. Further analysis and refinement is expected to address many of those areas where the all- up option currently shows an advantage.

Long-stay missions have a relative small difference between minimum energy trajectories and fast-transit trajectories. This is one area where the choice of the long-stay mission for comparison purposes is not reflective of the short-stay option.

Given that the long-stay option was used for this comparison, results indicate that these options show similar sensitivity to architecture or system changes. This means that unlike the long-/short-stay comparison where the impact of (inevitable) changes can be dramatically different depending on the option selected, whichever option is selected here will be impacted to the same degree. This also means that the gear ratio is not a discriminator for this particular assessment (see table 3-14 for this comparison).

**Note:** As previously stated, the chemical transportation option assumed development of a single “common” upper stage and used multiple copies of this stage to meet a particular delta-V requirement. In this case, that common stage was able to use more of the payload lift capability of a single Ares V than a comparable common stage that was optimized for the pre-deploy option. This contributed to a slightly better gear ratio for the all-up option despite the fact that the mass performance tended to favor the pre-deploy option.

**Table 3-14.** Pre-Deploy Gear Ratio Table

| | Pre-deploy | | All-up | |
|---|---|---|---|---|
| | NTR | Chemical | NTR | Chemical |
| Cargo: $\Delta M_{IMLEO}/\Delta M_{MO}$ | 2.24 | 3.49 | 2.30 | 4.01 |
| Crew: $\Delta M_{IMLEO}/\Delta M_{MO}$ | 2.14 | 3.80 | 2.14 | 3.70 |
| Crew: $\Delta M_{IMLEO}/\Delta M_{TE}$ | 2.58 | 6.12 | 2.58 | 5.97 |

For the level of analysis that was conducted in this comparison, the primary conclusion is that there is no clear advantage from a risk perspective for either the pre-deploy or all-up option – the cumulative probability of loss of mission (PLOM) and probability of loss of crew (PLOC) are essentially the same. Examining the details that went into this conclusion, the number of launches is a significant contributor both for the inherent risk of a launch and for the probability of completing the launch campaign in the allotted period of time (to which has been added a 120-day contingency for this analysis).The number of launches is driven not only by this choice of options, however, but by the propulsion technology choice. This latter choice drives the number of launches for each of these deployment options to differing degrees and at times in different directions (for the current analyses, the all-up NTR mission takes an extra Ares V launch while the pre-deploy chemical mission takes two extra launches), resulting in no clear trend for risk. An additional risk contributor is the cumulative operating time on these systems; i.e., the longer they must operate, the greater the chance of failure due to “old age” factors. Figure 3-28 shows results assuming that all systems operate at a nominal level for the entire deployment time, which is a conservative assumption for these systems. The all-up mission benefits significantly from a reduction in the duration of equipment reliability. However, if all components are dormant prior to crew arrival, the long-stay pre-deploy mission would see a risk reduction of 5% from the value that is indicated. Equipment reliability can be enhanced by scavenging techniques when a crew is present. These techniques can be learned during lunar missions. However, no credit has been taken in these estimates for any precursor activities or demonstrations (a conservative assumption). Thus, some systems are penalized from a risk perspective because this is the theoretical “first use” (a specific example being the DAV for the all-up option).

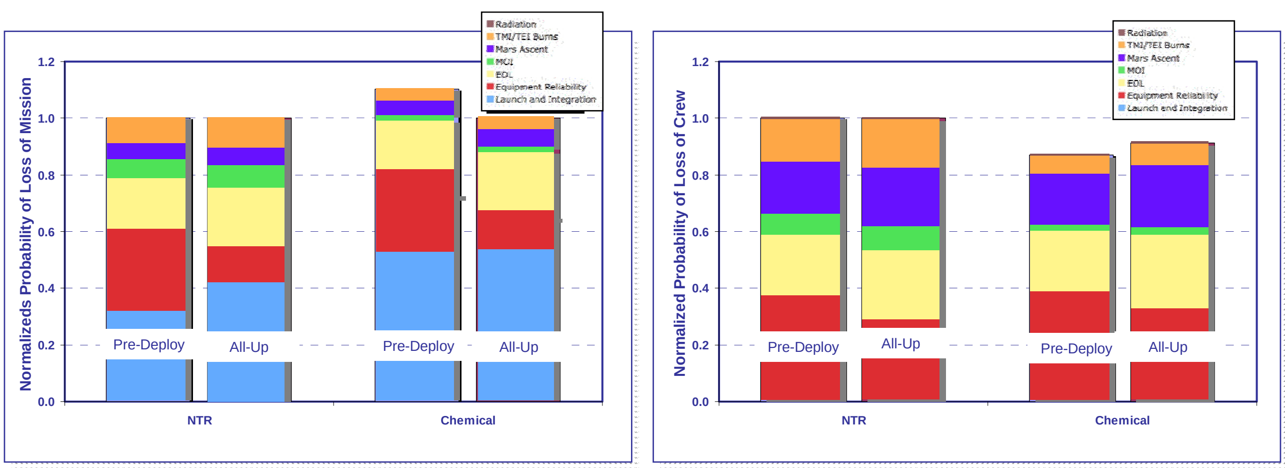


**Figure 3-28.** Pre-deploy risk comparison.

For the pre-deploy option on the long-stay mission, it has been noted that assets for subsequent crews can be used in contingency situation by previous crews. This option does not improve the PLOC, just the PLOM, because (based on the GR&As that were used) any crew has the option to leave the surface and return to its orbiting transfer vehicle at any point in the surface mission, where sufficient consumables are carried to sustain the crew for an entire round-trip flight time. In addition, exercising such an option would automatically result in deferring the next mission due to lack of assets at Mars, thus diminishing the overall productivity of the campaign and effectively causing a “loss of campaign.”

For the all-up vs. pre-deploy comparison, the trade should be driven primarily by the risk assessment, but there remains a distinct difference in cost. That difference basically boils down to the Mars program paying for JSC and KSC operations 2 years earlier than it would have otherwise. However, this is something of a "virtual" savings to NASA as a whole, since it can be assumed that for the all-up missions JSC and KSC costs will have to be paid by someone over those 2 years. There is a similar issue going on now with CxP, space shuttle, and ISS operations costs.

This factor leads to a slight cost advantage, although it may not be real, for the all-up option through the first mission. After a three-mission campaign, this factor tends to be washed out by other effects and the two options become virtually identical (given the level of detail in this analysis). The number and types of systems that must be developed and then acquired for each mission are similar between the two options, and the advantage is dependent on the propulsion system that is chosen (see figure 3-29).

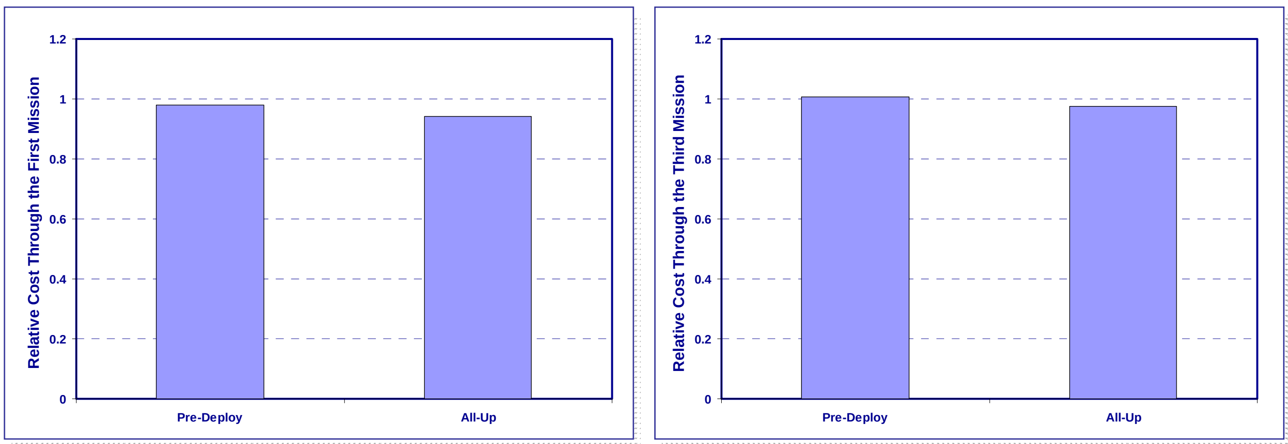


**Figure 3-29.** Pre-deploy cost comparison summary.

### 3.4.5 Cargo deployment recommendation

Based on the analyses and results that are presented here, the study team recommends that the pre-deploy feature be used as the reference approach in DRA 5.0 based on the following results and considerations:

- Slightly lower IMLEO to accomplish mission
- Backup DAV and SHAB available (for long-stay mission only)
- Opportunity for significant surface operations in advance of crew, resulting in greater cumulative mission return
  - Infrastructure setup and verification
  - Open opportunity for significant ISRU operations
  - Reconnaissance of sites targeted for crew to explore
- Potentially simpler LEO operations with fewer elements to be integrated prior to departure

A more extensive comparison of these two options is shown in figure 3-30.

## *3.5 Mars Orbit Capture Method (Propulsive vs. Aerocapture)*

To place a spacecraft in orbit around a planetary body, sufficient velocity must be removed such that the gravitational field of the target body will transform the approach hyperbolic trajectory into a closed elliptical orbit. Traditionally, this has been accomplished using chemical propulsion to provide deceleration forces to slow the spacecraft to the required velocity for orbit capture. For planetary bodies that posses an atmosphere, including Mars, using atmospheric drag to provide aerodynamic deceleration may result in significant mass savings over the more traditional propulsive orbit insertion methods. In the development of the architecture for this study, all elements (crewed and cargo) are assumed to be first inserted into a Mars orbit for operational and safety reasons prior to EDL, as opposed to a direct entry from the Earth-Mars cruise phase. Due to the multiple elements and large volumes that are associated with the crewed elements of the mission architecture (transit habitat, descent stage, and CEV/Orion Earth return vehicle with the TEI stage), the use of aerocapture for Mars orbit insertion was considered impractical and was used only for the cargo elements. This effort was focused on determination of the mission performance, cost and risk sensitivities of using aerocapture vs. the more traditional propulsive capture methods.

**Human Exploration Of Mars**

| Pre-Deploy Cargo | Figure of Merit | All-Up Mission |
|---|---|---|
| | Performance | |
| **Slightly lower** | Total mass in Low-Earth Orbit (mt) | Slightly higher |
| Similar | Total Number of Ares-V Launches | Similar |
| Similar | Total Number of Ares V Launches in Single Window | Similar |
| | Risk | |
| No clear advantage | Probability of Loss of Mission (Plom) | No clear advantage |
| No clear advantage | Probability of Loss of Crew (Ploc) | No clear advantage |
| Higher | Max Cum Time on Systems by End of Mission | **Lower by one year or more** |
| | Cost | |
| Slightly higher | Cost through first mission | Slight advantage |
| Similar | Cost through third mission | Similar |
| | Other | |
| **Available (long stay only)** | Backup DAV and SHAB | None |
| **Longer duration** | Exploration Goal Satisfaction (longer surface operations) | Shorter duration |
| Smaller units | LEO Complexity / Size of Elements launched | Large single unit or convoy |

**Figure 3-30.** Pre-deploy figure of merit summary.

Aerocapture is a method to directly capture into the orbit of a planet from a hyperbolic arrival trajectory using single, atmospheric aerodynamic drag pass, thereby reducing the propellant that is required for orbit insertion. Over the last several decades, multiple aerocapture systems analysis studies have been conducted for multiple planetary destinations (Earth, Mars, Venus, Titan, Neptune), with a variety of aerodynamic shapes and guidance algorithms. Although they have all concluded that aerocapture is a moderate to relatively low-risk technology (Hall, 2005[74]; Cerimele, 1985[75]; Lockwood, 2003[76]; Lockwood, 2004[77]; Wright, 2006[78]), these studies were typically limited to the significantly smaller payloads (1–2 t) that are associated with robotic missions. This effort attempted to address aerocapture performance for the much larger 50–100-t payloads that are required for human-class missions. The aerocapture technique requires an aeroshell with sufficient thermal protection system (TPS) to protect the payload from the aerodynamic heating that is encountered during the atmospheric pass. During the aeropass maneuver, an atmospheric flight guidance and control algorithm is used to target the trajectory to a specified condition following atmospheric exit; then an orbit periapsis raise maneuver is executed to achieve the target orbit conditions, as shown in the aerocapture flight profile schematic in figure 3-31.

[74]Jeffery L. Hall, Muriel A. Noca, and Robert W. Bailey, "Cost–Benefit Analysis of the Aerocapture Mission Set," *AIAA Journal of Spacecraft & Rockets*, Vol. 42, No. 2, (pp. 309-320), 2005.
[75]Cerimele, C. J., and Gamble, J. D., "A Simplified Guidance Algorithm for Lifting Aeroassist Orbital Transfer Vehicles," AIAA-85-0348, AIAA 23rd Aerospace Sciences Meeting, Reno, NV, January 1985.
[76]Lockwood, Mary Kae, Titan Aeorcapture Systems Analysis, AIAA-2003-4799, 39th AIAA/ASME/SAE/ ASEE Joint Propulsion Conference and Exhibit, Huntsville, Alabama, July 20-23, 2003.
[77]Lockwood, Mary Kae, Neptune Aerocapture Systems Analysis, AIAA-2004-4951, AIAA Atmospheric Flight Mechanics Conference and Exhibit, Providence, Road Island, August 16-19, 2004.
[78]Henry Wright, et al. "Mars Aerocapture Systems Study", NASA TM 2006-214522, Nov. 2006.

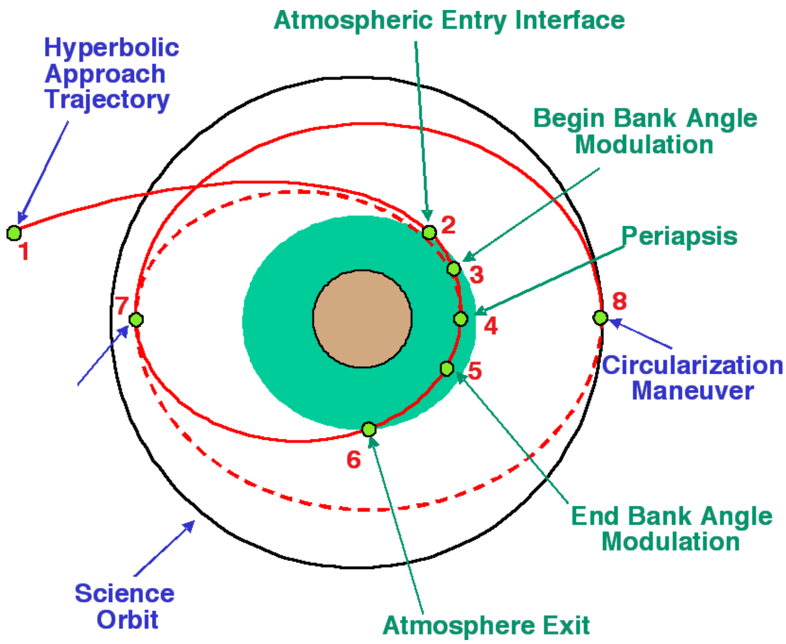


**Figure 3-31.** Aerocapture flight profile.

Although the aerocapture technique has not yet been demonstrated on an operational mission, studies have demonstrated the feasibility of, and identified potential savings in, the propellant and overall system mass that is required for orbit insertion. The primary aerocapture technology challenges are the TPS, sufficient knowledge of the atmospheric density profiles, and aerocapture guidance and control algorithms. TPS challenges are thought to be no more demanding than direct-entry TPS but are configuration-specific to new shapes and heat pulse duration.

Given that the Mars EDL system will already require a hypersonic entry aeroshell, the system can be easily modified to also serve as the aerocapture aeroshell by adding the appropriate TPS to that which already exists for the hypersonic entry phase of the mission. In fact this and a small amount of additional propellant for the post-aerocapture periapsis raise and orbit trim burn are the only major hardware additions to the system to enable an aerocapture maneuver. The question then is: What are these additional mass requirements that are relative to the propellant mass requirements for the propulsive orbit capture?

To determine the potential mass savings, system-level trades were conducted using aerocpature and both chemical propulsion and NTP options for Mars orbit insertions. First, it was important to understand the sensitivities of aerocapture performance to the possible variations in vehicle design for this mission. The key parameters of interest were ballistic number, lift-to-drag ratio (L/D, target orbit (a 500-km circular orbit, or a 1-Mars-sol orbit period), and the atmospheric entry velocity of the arriving vehicle. Early in the study, the specific EDL system definitions were not yet fully defined; so, to investigate the scope of the problem, several initial conservative assumptions were made. The desired useful landed mass at the surface of Mars was assumed to be between 20 and 80 t. These values were used to derive an entry mass using a set range of “gear ratios” that was obtained from historical studies. A lower bound on the useful mass/entry mass gear ratio equal to 0.5 was selected based on early human Mars mission studies, along with an upper bound on this gear ratio term that was equal to 0.64 and was obtained from more recent design studies. The entry mass from orbit thus spanned a range from 31 to 160 t.

The orbit mass was then estimated by assuming an approximate DV = 110 m/s requirement for a deorbit maneuver. Using an $I_{sp}$ range of 330–450 seconds resulted in an orbit mass of 32 to 166 t. The estimated aerocapture mass was then computed by adding the propellant mass that is required to perform the post-aerocapture circularization burn of approximately 150 m/s. Assuming the same $I_{sp}$ range as above, the resultant aerocapture mass range was 33 to 174 t.

Based on the assumption of no EVA on-orbit assembly of components, aeroshell dimensions were assumed to be constrained to the estimated capability of the launch vehicle. The reference Ares V launch vehicle payload shroud provided accommodation for a 7.5-m-diameter and 12-m-long vehicle. Potential growth was estimated to 12 m in diameter and 35 m in length. To limit the scope of the study, initial geometric assumptions were based on an ellipsled configuration for the aeroshell (Wright, 2006); however, given the same dimensions, small modifications to the trim angle-of-attack could be made to achieve similar ballistic numbers and L/D ratios for other shapes including biconics and triconic class of slender body mid-L/D designs. A summary of the estimated range of aerocapture vehicle parameters is provided in table 3-15.

**Table 3-15.** Estimated Range of Aerocapture Vehicles

| | Minimum | Maximum |
|---|---|---|
| Desired useful landed mass | 20,000 kg | 80,000 kg |
| Gear ratio (useful landed to entry mass) | 0.5 | 0.64 |
| Entry mass | 31,000 kg | 160,000 kg |
| Deorbit burn delta-V | 110 m/s | 110 m/s |
| Isp | 330 s | 450 s |
| Orbit mass | 32,000 kg | 166,000 kg |
| Post-aerocapture circ. burn delta-V | 150 m/s | 150 m/s |
| Aerocapture mass | 33,000 kg | 174,000 kg |
| Ellipsled diameter | 7.5 m | 12 m |
| Ellipsled length | 12 m | 35 m |

Given the pre-aerocapture total vehicle mass range of 33 to 174 t and the range of the vehicle sizes from 7.5 m × 12 m to 12 m × 35 m, the packing density of the vehicle was determined to compare with the results of previous configurations. A comparison of the aeroshell range estimates to those of several historical crewed and robotic vehicles is shown in figure 3-32. The three red ×'s on the upper and lower boundaries of the blue region represent three aeroshells that were examined in further detail, the 7.5 m × 12 m, 10 m × 25 m, and 12 m × 35 m ellipsled geometries. The upper limit (blue line) is determined by the packaging density of the aeroshell of 174 t mass, while the lower limit is determined by the packaging density of the aeroshell of 33 t.

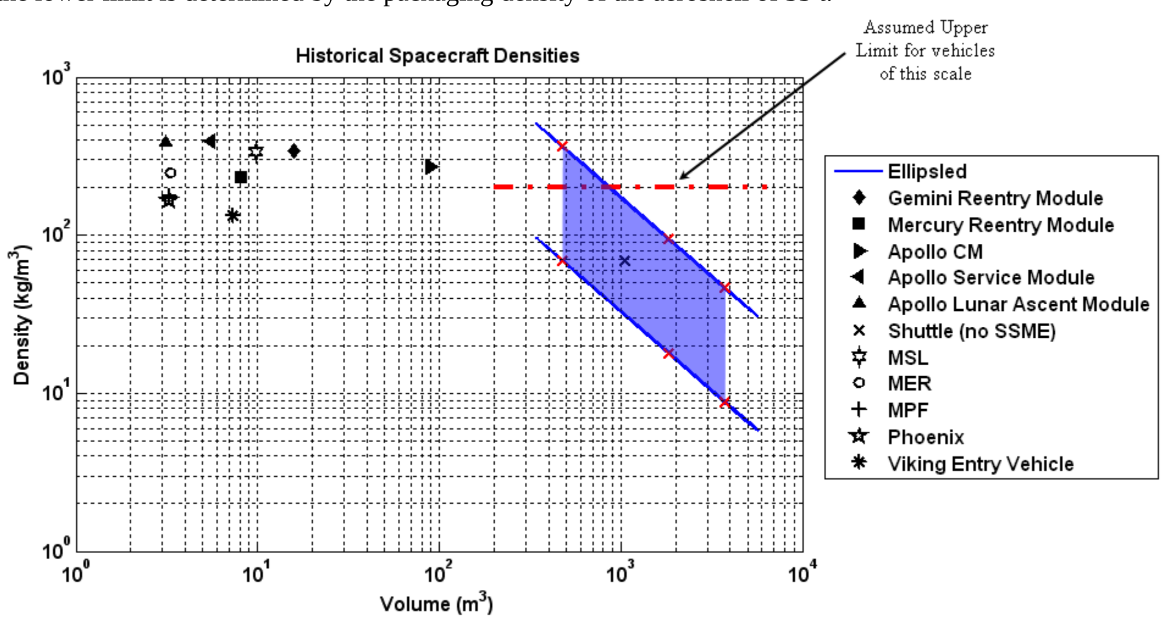


**Figure 3-32.** Aeroshell packing densities.

To understand the benefits and consequences of performing aerocapture at various L/D ratios and ballistic numbers, a parametric aerocapture assessment assuming optimal performace was employed. An open-loop lift-up to lift-down bank profile was used, using the L/D ratios of 0.3 to 0.7 and ballistic numbers of 100 kg/m$^2$ to 1800 kg/m$^2$. In these trajectories, the vehicle was assumed to enter the atmosphere at full lift-up (bank = 0 degree) and during the flight banks to full lift-down (bank = 180 degrees) to achieve the desired apoapsis altitude. Each trajectory was targeted to an entry flight path angle such that the peak deceleration load during the aerocapture pass reached 4 g's. This value was chosen as a nominal entry profile target. Once perturbations are taken into account, with close-looped guidance in a Monte Carlo simulation, the worst-case g-load would reach approximately 5.5 g's, which is an estimated acceptable upper limit for a human Mars entry. These trajectories, although simplified, present an adequate overview of the performance of a real guided aerocapture simulation.

In figure 3-33, the minimum altitude during the aerocapture pass using the full lift-up to full lift-down bank profile is shown. Each data point is targeted to a peak 4-g deceleration during the pass. Here the sensitivity to L/D can be seen, with the higher L/D vehicles achieving a higher minimum altitude. If a minimum pass altitude is desired, 25 km for example, the maximum ballistic number for an aerocapture vehicle will range from 600 kg/m$^2$ for a vehicle with 0.3 L/D to approximately 800 kg/m$^2$ for a vehicle with 0.7 L/D.

Also included on the plot are three Monte Carlo cases (indicated by the error bars), showing the dispersed range of minimum altitudes for a given vehicle based on closed-loop guidance with atmospheric perturbations. This indicates that the open-loop guidance assumption was valid; however, note that the variability in minimum altitude is approximately ±2 to 3 km.

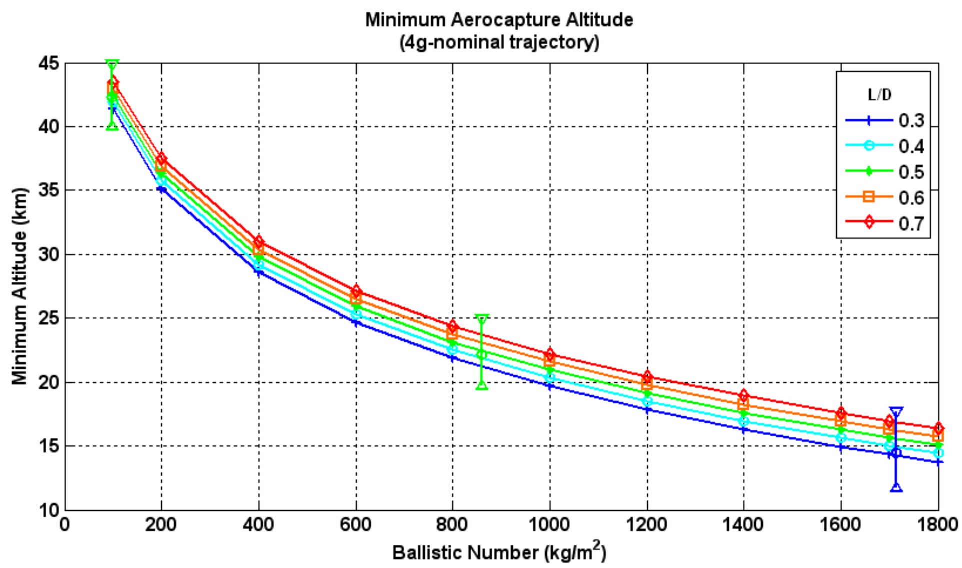


**Figure 3-33.** Minimum aerocapture altitude.

The entry flight path angle was targeted to achieve the desired 4-g deceleration as mentioned above and thus was different for each vehicle. To ensure proper margin, it was important to know how far the atmospheric entry interface was from the skip-out limit. The skip-out condition was defined as the shallowest flight path angle that was capable of achieving the target apoapsis while flying full lift-down with a 10% reduction in lift coefficient ($C_L$) and drag coefficient ($C_D$) and a 50% reduction in atmospheric density. The margin to skip-out is shown in figure 3-34. As can be seen in the figure, there is adequate entry flight path angle margin for the entire range of possible vehicles as estimated navigation errors for a Mars approach is approximately 0.25 degree 3σ, based on the recent robotic Mars probe missions.

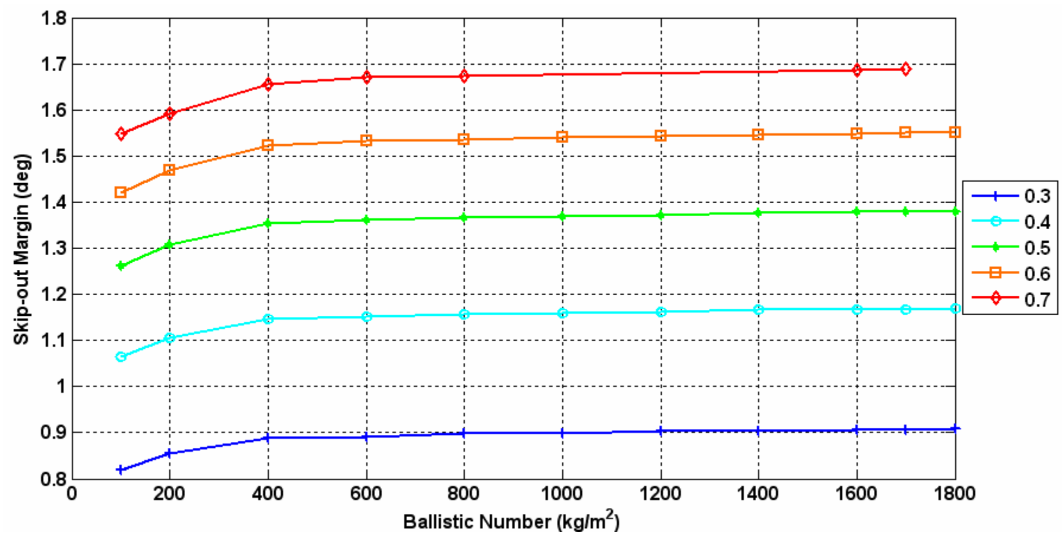


**Figure 3-34.** Aerocapture skip-out margin.

In figure 3-35, the peak heat rate and total integrated heat load sensitivities are shown as a function of aeroshell L/D and ballistic number for the aerocapture maneuver. In these cases, the peak rate and loads are determined for points on the windward aftbody, where turbulent heating is a maximum (as opposed to the stagnation point, which in this case is much cooler in a relative sense). These data indicate the fact that for moderate ballistic numbers (400–1,000 kg/m$^2$), the peak heating and total heat loads are well bounded by the TPS performance capabilities that are being developed for the Orion CEV lunar return conditions.

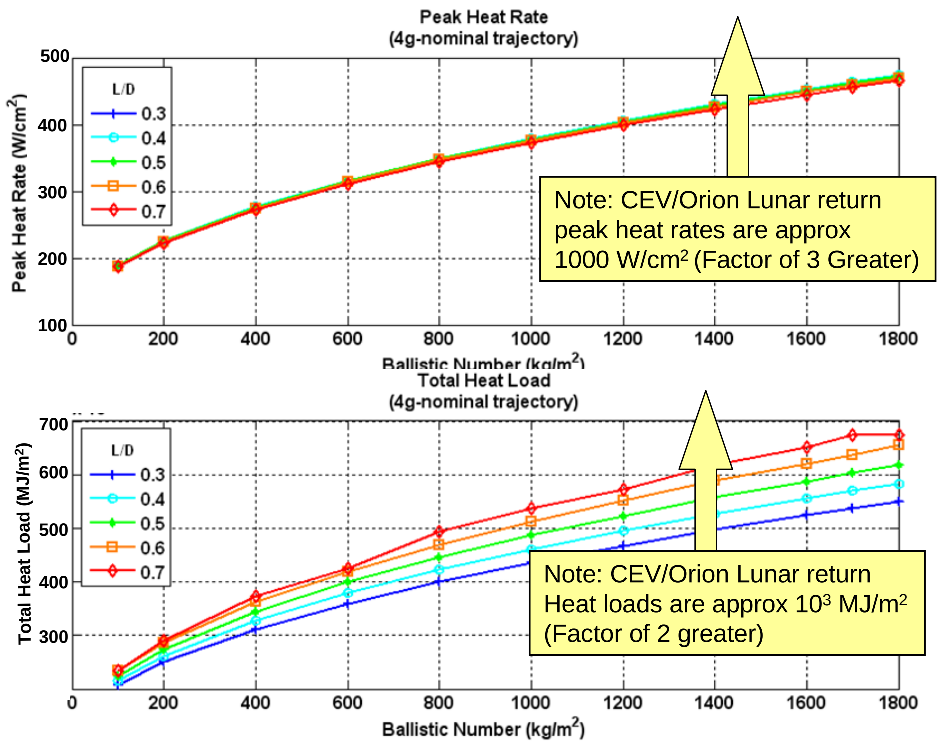


**Figure 3-35.** Mars aerocapture heat rate and heat load.

The results of the trade study can be overlaid in one set of plots, which are shown in figures 3-36 through 3-39. The minimum altitude for each L/D and ballistic number combination is shown as a contour plot (blue). As can be seen, the L/D has only a small effect on the minimum altitude during the pass. This happens because the entry flight path angle was changed accordingly to target the desired 4-g trajectory. The entry flight path angle margin to skip-out is also plotted (red contour). With the exception of the very small ballistic numbers, the skip-out margin is not affected by the ballistic number but is a function of the L/D. The higher L/D provides the largest margin. In each of the plots, different vehicles can be overlaid. In the case of figure 3-36, for example, a 12-m-diameter Apollo capsule geometry is assumed. The mass that is required to achieve the desired ballistic number and L/D is plotted (magenta), along with the associated packing density of the ellipsled (green).

For example, to achieve a ballistic number of 600 kg/m$^2$ with an L/D of 0.4, the vehicle mass must be 43,000 kg and will have a packing density of approximately 91 kg/m$^3$. This vehicle will fly to a minimum altitude of approximately 26 km, and will enter the atmosphere 1.9 degrees from the skip-out margin. Due to the dimension of the vehicle, the maximum L/D achievable is approximately 0.45. The spike in the mass curves for the small capsule is caused by the extrapolation errors of the drag coefficient from the existent aerodynamic database. A similar set of data is provided for 7.5-m × 12-m, 10-m × 25-m, and 12-m × 35-m ellipsled aeroshell geometries.

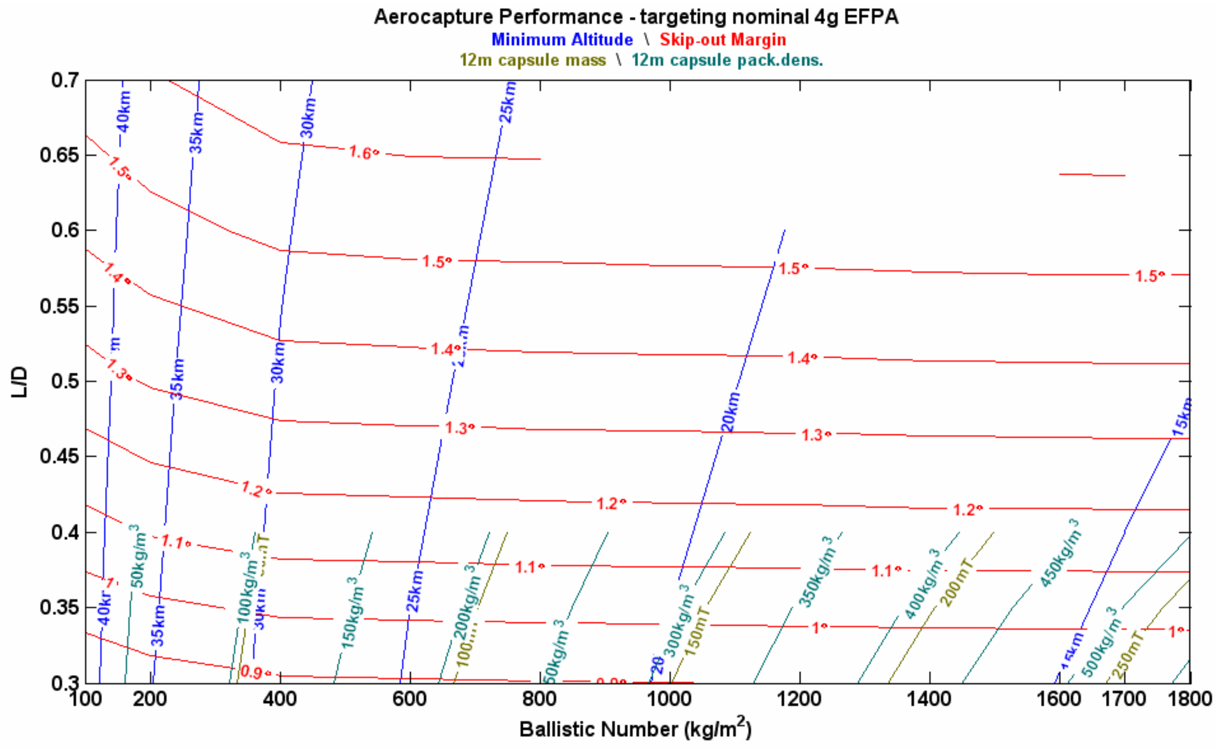


**Figure 3-36.** The 12-m-diameter capsule performance.

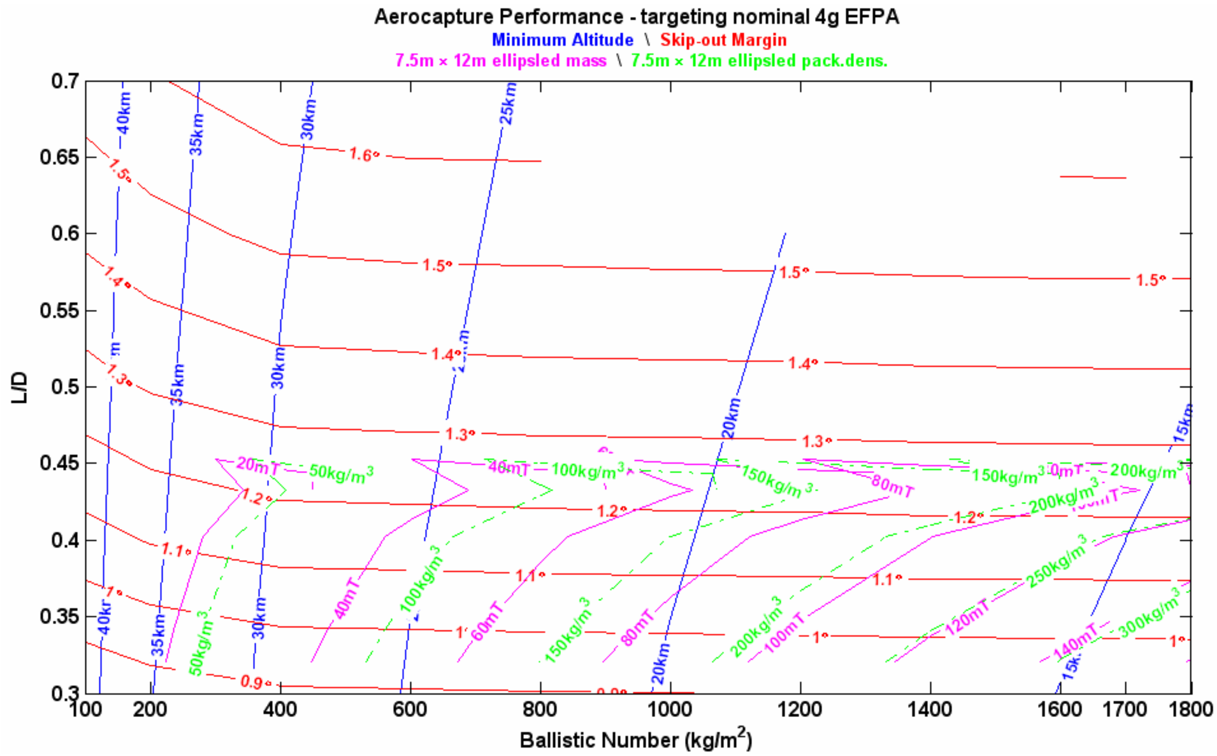


**Figure 3-37.** The 7.5-m x 12-m ellipsled performance.

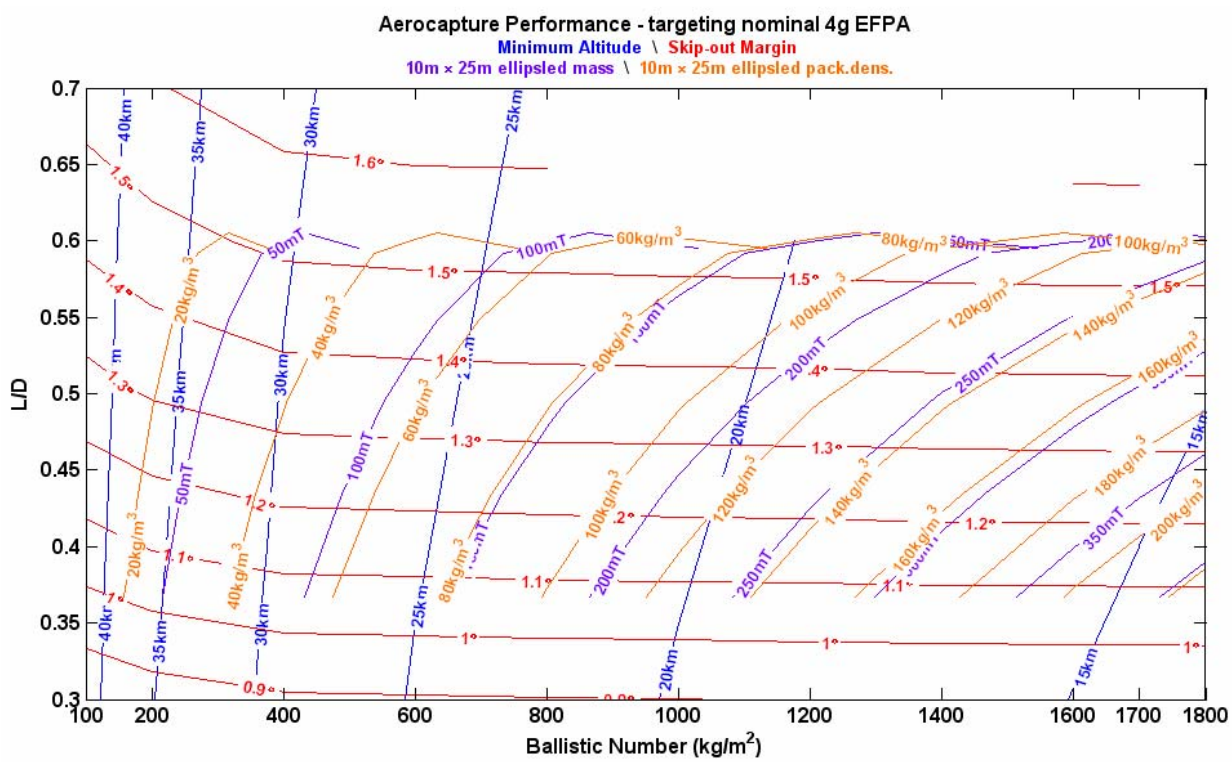


**Figure 3-38.** The 10-m x 25-m ellipsled performance.

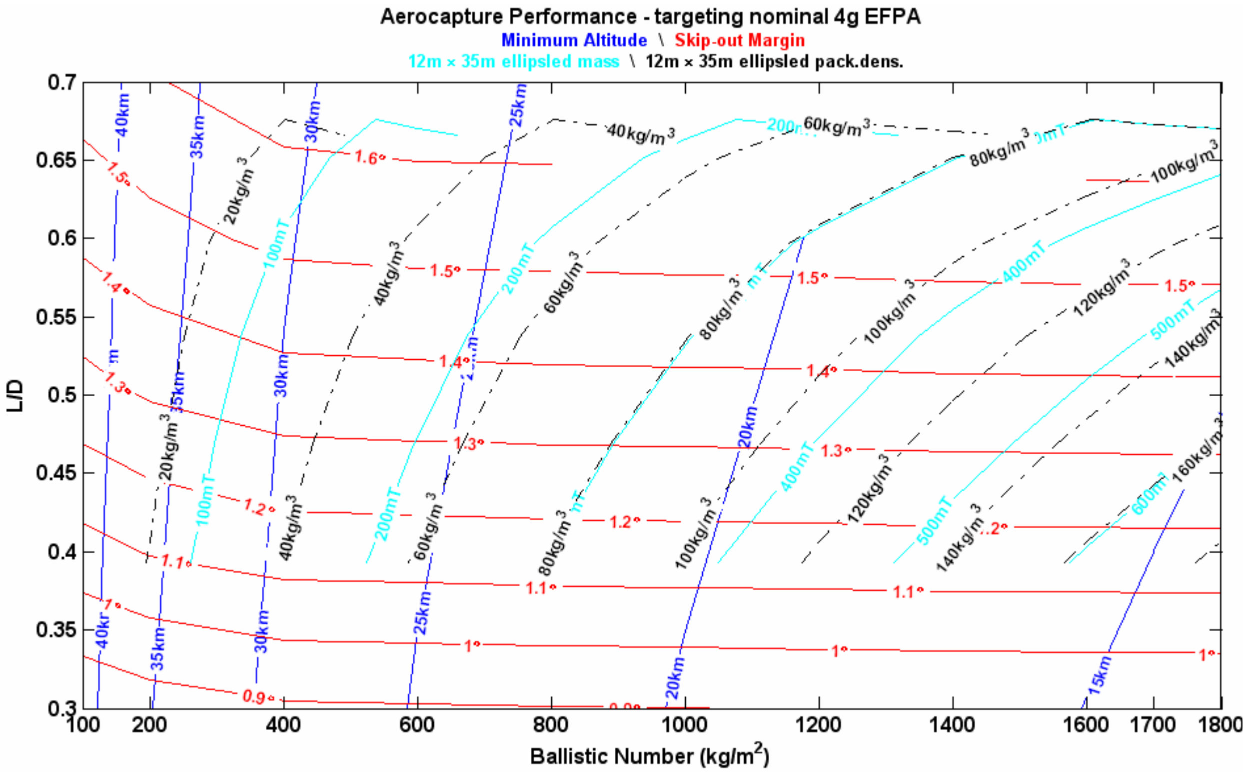


**Figure 3-39.** The 12-m x 35-m ellipsled performance.

At this point in the study, an aerocapture point design solution was developed that assumed a 40-t useful payload on the surface of Mars. Corresponding to this, an EDL system design, which will be described in the following section, resulted in a Mars arrival mass of 115 t. Using the assumptions in table 3-16, aerocapture trajectory performance was evaluated using a 2000-case Monte Carlo simulation, employing the Hybrid Predictor-corrector Aerocapture Scheme (HYPAS) guidance algorithm [2]. The algorithm has been analyzed extensively in aerocapture mission studies at Earth, Mars, Titan, and Venus.

**Table 3-16.** Aerocapture Assumptions

| | |
|---|---|
| Vehicle Aeroshell Dimension | 10 m × 30 m |
| $C_D$ | 2.957 |
| $C_L$ | 1.39 |
| Mass | 115,549 kg |
| Target Apoapsis | 33,793 km |
| (1 sol orbit period) | |
| Monte Carlo Dispersions | |
| Entry flight path angle | 0.35 degree 3σ |
| Aerodynamic | ±10% |
| Atmospheric variations | Mars GRAM |

Monte Carlo simulations were used to perform a performance capability verification, and results are shown in figures 3-40 through 3-43. The results that are shown in figure 3-40 indicate adequate targeting performance for this vehicle despite the high ballistic number (498 kg/m$^2$) and the high-energy orbit (exit velocity representing 97% of escape velocity) that increase the difficulty of targeting the desired orbit. Although the standard deviation on the apoapsis dispersion is approximately 1,400 km, further tuning of the guidance algorithm will improve this performance.

Overall performance of the aerocapture maneuver can be measured in terms of the post-aerocapture circularization burn requirements. Figure 3-41 indicates that the mean delta-V that is required is only 19 m/s, with a maximum 66-m/s case.

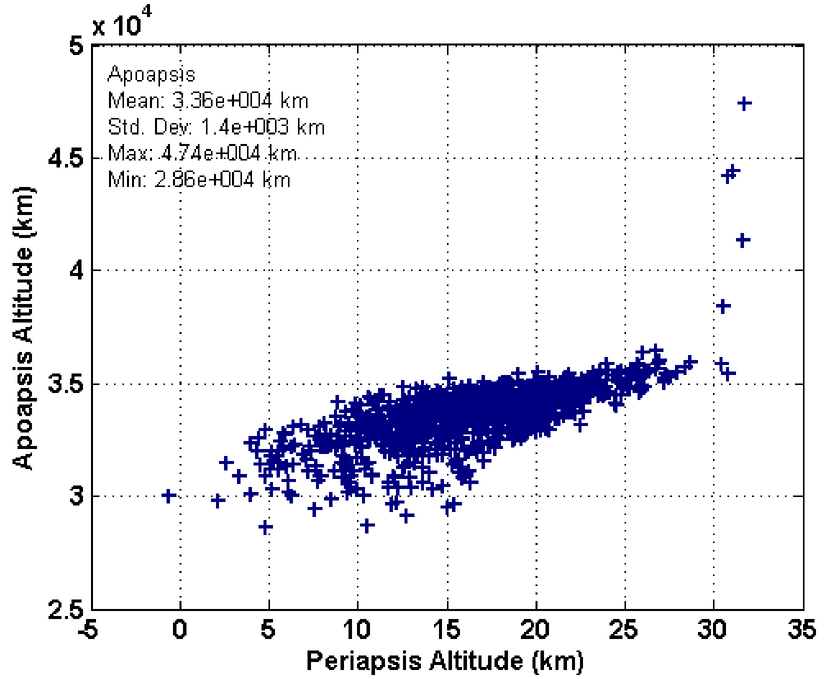


**Figure 3-40.** Monte Carlo apoapsis and periapsis dispersions.

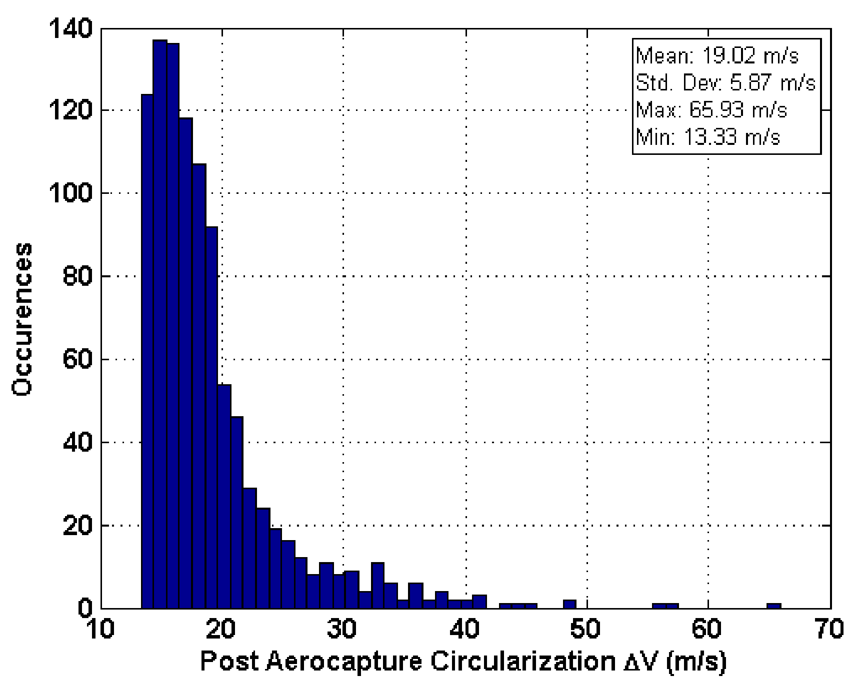


**Figure 3-41.** Monte Carlo post-aerocapture circularization delta-V.

The heating data that are shown in figure 3-42 indicate heat flux and heat loads referenced to a 1-m sphere using the Sutton-Graves heating equation (Sutton, 1971[79]). Finally, the nominal trajectory was targeted to achieve a peak 4-g deceleration to not exceed 5.5 g's during a worst-case dispersion, as seen in figure 3-43.

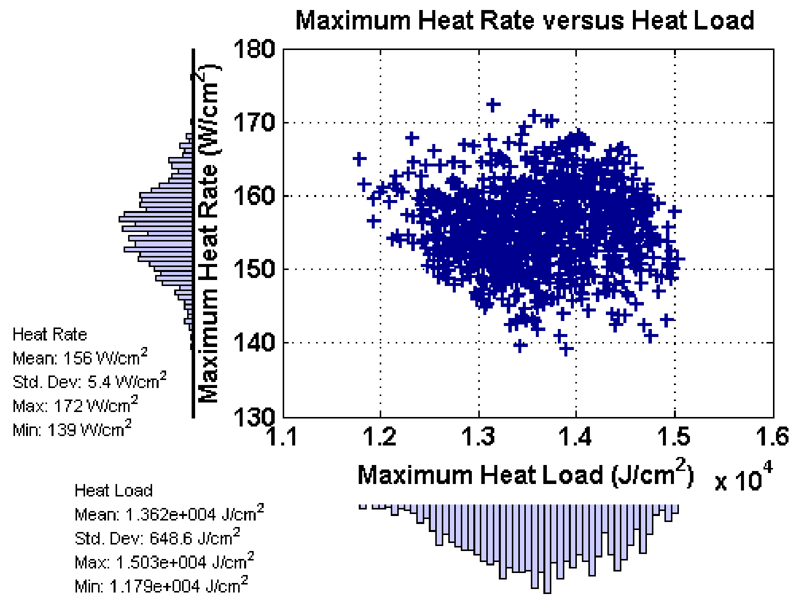


**Figure 3-42.** Monte Carlo maximum heat rate and total heat load.

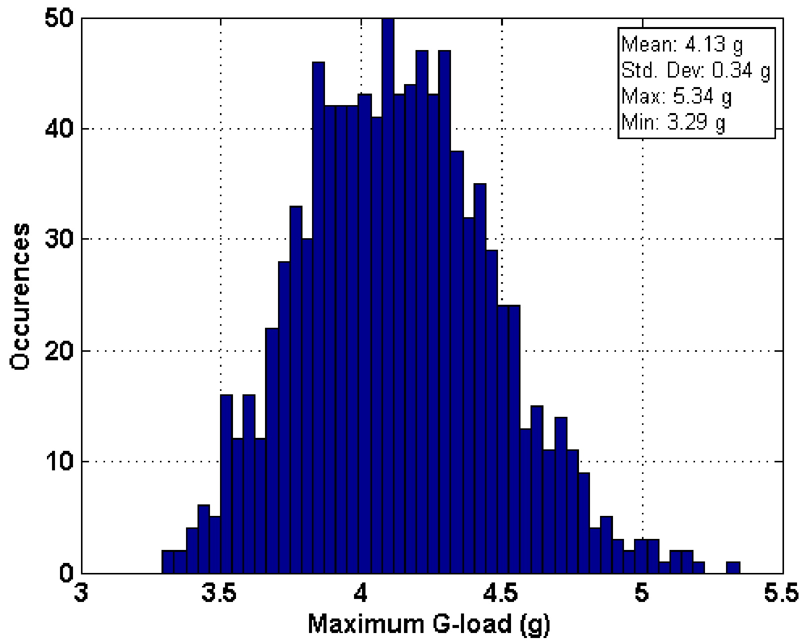


**Figure 3-43.** Monte Carlo maximum deceleration.

Based on the results of these parametric and Monte Carlo performance assessments, the aerocapture maneuver was deemed a feasible option for the large-scale, high-mass systems that are consistent with the human-class mission set.

[79]K. S. Sutton, and R.A Graves, Jr., "A General Stagnation-Point Convective-Heating Equation for Arbitrary Gas Mixtures", NASA TR R-376, November 1971

The next step was to conduct a direct reference architecture level comparison of the cargo mission aerocapture vs. a propulsive Mars orbit insertion maneuver. The architecture trade tree options 4, 6, 10, and 12 were examined and compared. These were aerocapture with NTR interplanetary propulsion (TMI/TEI stages), aerocapture with conventional chemical rocket propulsion stages, and the all-propulsive orbit insertions using NTR and chemical rocket propulsion stages, respectively. Mission-level trades were conducted, and the results are shown in table 3-17.

**Table 3-17.** Aerocapture vs. Propulsive MOI Trade Results

| **Aerocapture MOI** | | **Figure of Merit** | **All-Propulsive MOI** | |
|---|---|---|---|---|
| **CHEM** | **NTP** | | **CHEM** | **NTP** |
| 126 | | Mars Orbit Mass, post MOI (t) | 122 | |
| Rr | | Aeroshell + TPS mass – 10 $\times$ 30 m (t) | 41 | |
| 1378 | 911 | **Total IMLEO req** | 1728 | 998 |
| 12 | 10 | Total Number of Ares V Launches | 15 | 10 |
| 2.49 | 1.87 | MOI System Mass Ratio Sensitivity to IMLEO | 3.49 | 2.24 |

CHEM = chemical propulsion; MOI = Mars orbit insertion;

The aerocapture cases required slightly more mass in Mars orbit (~4 t) due in large part to the additional TPS that is required to execute the aerocapture maneuver, which would then be reused during the EDL phase. Detailed TPS sizing assessments were performed for both the aerocapture that was followed by entry and the entry-alone options to validate these data. However, these masses were small in comparison to the additional NTR and chemical propellant masses that are required to execute the all-propulsive burns for orbit insertion. Ultimately, the performance metric that was used for direct comparison was determined to be the IMLEO. The missions that use aerocapture achieve a significant savings in IMLEO requirements. In the case of conventional chemical propulsion options, the difference is 1,378 t IMLEO for the aerocapture case vs. 1,728 t IMLEO for the chemical propulsion MOI. These IMLEO masses would require 12 Ares V launches vs. 15 Ares V launches, respectively. The difference in aerocapture vs. propulsive capture for the NTR propulsion options is less dramatic; 911 t vs 998 t for the aerocapture and NTR propulsive capture options, respectively. In both cases, 10 Ares V launches will be required to deliver this IMLEO; however, the aerocapture cases would have more partial payload launches (five vs. two) for the NTR, which implies some additional margin or robustness for the aerocapture architecture. One additional FOM that wasexamined was the MOI system mass ratio sensitivity; that is, the amount of additional IMLEO required for each unit mass of payload that is required in Mars orbit. These cases also favor the use of aerocapture for both NTR and chemical propulsion options (smaller is better).

Risk and cost assessments were also conducted for the aerocapture vs. propulsive MOI options. Detailed quantitative loss of mission and of crew risk metrics were difficult to define; however, qualitative assessments of the risks that are associated with aerocapture were reviewed. Many systems analysis studies and projects have examined multiple targets (Earth, Mars, Venus, Titan, Neptune) with a variety of shapes (low-L/D sphere cones to mid-L/D slender bodies), with/without aerodynamic control surfaces, and a variety of guidance algorithms and have all concluded that aerocapture is a relatively low-risk technology. The overall TPS requirements for Mars aerocapture are much less stressing than those that are associated with either lunar or Mars Earth return (Orion/CEV Blocks II and III). Given the similarities between the aerocapture MOI maneuver and the skip-entry maneuver that may be used by the Orion CEV for lunar return, many of the risks that are associated with aerocapture, including guidance system performance and dual-use TPS, will be retired via the CEV/Orion development program. The use of aerocapture is felt to be a relatively small incremental cost to the larger, more challenging EDL system development costs and risks. The major engineering challenges and technology risk reduction efforts that are required for EDL system development will also serve to retire many of the risks that are associated with aerocapture technology. Some incremental technology development and risk reduction efforts will be required, but these are felt to be moderate and easily manageable. Preliminary risk analysis and modeling indicate no significant risk discriminators between aerocapture and propulsive capture (chemical or NTR).

The cost assessments that were performed indicate that there is a distinct long-term cost advantage to the aerocapture mission for the chemical propulsion option due to a large reduction in the number of launches and flight elements. The cost for the multiuse aeroshell design and production for three missions was estimated as a 6% increase over the cost of an entry-only aeroshell. There is some cost risk inherent to this assessment due to technological uncertainties in the development of the dual-use aerocapture aeroshell, but this was deemed to be small and the overall cost sensitivity to this assumption was minimal.

The cost advantage of the aerocapture option is reduced for the NTR-based propulsion due to the reduction in mass sensitivity that NTR provides. For the NTR systems, cost is not seen as a significant factor in the aerocapture trade. Aerocapture does provide increased launch margins that may have cost implications that are not captured, however.

**Conclusions and Recommendations**
The reference aerocapture system architecture is felt to be a conservative design that includes a large nonoptimized aeroshell with relatively large uncertainty margins, including TPS. However, mass estimates are based on engineering judgment and extrapolations primarily from much smaller-scale robotic EDL systems. There are other options yet to be explored, including dual-use launch vehicle/aerocapture shrouds, inflatable/deployable aeroshells, etc., that may substantially improve the performance of the aerocapture and EDL system as a whole. Therefore, recommendations for the development and use of aerocapture technology for the human Mars architecture are as follows:

1. Continue to include aerocapture for Mars orbit insertion on cargo missions as the reference approach until a decision on the propulsion option (chemical vs. NTP) is determined
2. Conduct detailed “pre-Phase A” point designs to validate mass models for both aerocapture and propulsive capture MOI
3. Continue to pursue options to improve aerocapture system performance and understand overall system-level performance, cost, and risk sensitivities and drivers
4. Take advantage of the Orion/CEV lunar return “skip-entry” qualification and flight data to retire risks that are associated with dual-use TPS and aerocapture guidance performance

It must be noted, that time constraints of the study limited detailed assessments of integrated systems design impacts including thermal soak-back, center of gravity control, and separation dynamics to name a few. Further assessments in these areas are necessary in order to adequately address the use of aerocapture techniques for capturing cargo elements into Mars orbit.

## *3.6 Mars Ascent Propellant (In-situ Resource Utilization)*

### 3.6.1 Introduction

Mars ISRU involves the production of critical mission consumables, such as propellant and life support consumables, from resources that are available at the site of exploration. The main rationale for incorporating ISRU technologies into a Mars mission is to attempt to reduce IMLEO by reducing landed mass (IMLEO being a first-order measure of cost and risk). Incorporation of ISRU can also significantly enhance, if not enable, more robust exploration capabilities while also providing redundancy of critical functions such as life support. Since propellants and life support consumables for a long surface stay make up a significant fraction of the mass that must be launched from Earth, ISRU can either reduce the total amount of mass that must be launched or replace propellant/consumable mass with extra payload or science. In particular, the potential benefits of ISRU were assessed for the Mars ascent vehicle propulsion system and for the creation of consumables for life support and EVA needs. Several ISRU technologies were analyzed for their mass-reduction benefits during the course of trade studies for DRA 5.0. Analyses must take into account the mass of all hardware that are needed to enable ISRU (including power systems), the total volume (including reagents brought from Earth), and any risk the use of ISRU contributes to loss of mission or crew. Past DRMs have also documented some of the benefits of ISRU technologies, but in a less comprehensive manner than documented here. Prior studies were limited to the investigation of Mars atmospheric resources (e.g., $CO_2$, $N_2$, and argon (Ar)), whereas this study also performed an initial investigation of the use of surface regolith material as a source of $H_2O$ as well. Mars ascent vehicle propellant options of liquid oxygen ($LO_2$)/$CH_4$, $LO_2$/$H_2$, and hypergolic propellants have all been examined and traded in the past. In the present study, emphasis was placed on $LO_2$/$CH_4$ as the clear choice from previous trades. The sizing of back-up life support consumables was also previously analyzed in DRM 3.0. What has not been previously studied, but was investigated this time, was:

1. A direct comparison of ISRU vs. no ISRU propagated all the way back to LEO and the corresponding number of launches
2. A comparison between using a 1-sol-Mars rendezvous orbit vs. a 500-km orbit for ISRU and no-ISRU in terms of how the choice that optimizes the ascent vehicle design compares to the ideal solution for the transportation system
3. Preliminary investigation of the use of hydrated minerals for the creation of mission-critical consumables
4. The impact associated with bringing hydrogen ($H_2$) from Earth for $CH_4$ production vs. bringing $CH_4$ fuel on total lander volume and ISRU option trade selections
5. The impact of ISRU on “campaigns” to the same surface location on Mars more than once

To assess the cost/benefits of the use of ISRU both for ascent propellant and for crew consumables, the trade tree cases, which are shown in figure 3-44 (1 and 3 for ISRU; 4 and 6 for non-ISRU), that corresponded to Conjunction Class missions using aerocapture and a pre-deploy strategy were the focus of analysis. Analyses were conducted without deciding whether nuclear or chemical propulsion would be used for the interplanetary stage (i.e., both options were carried forward).

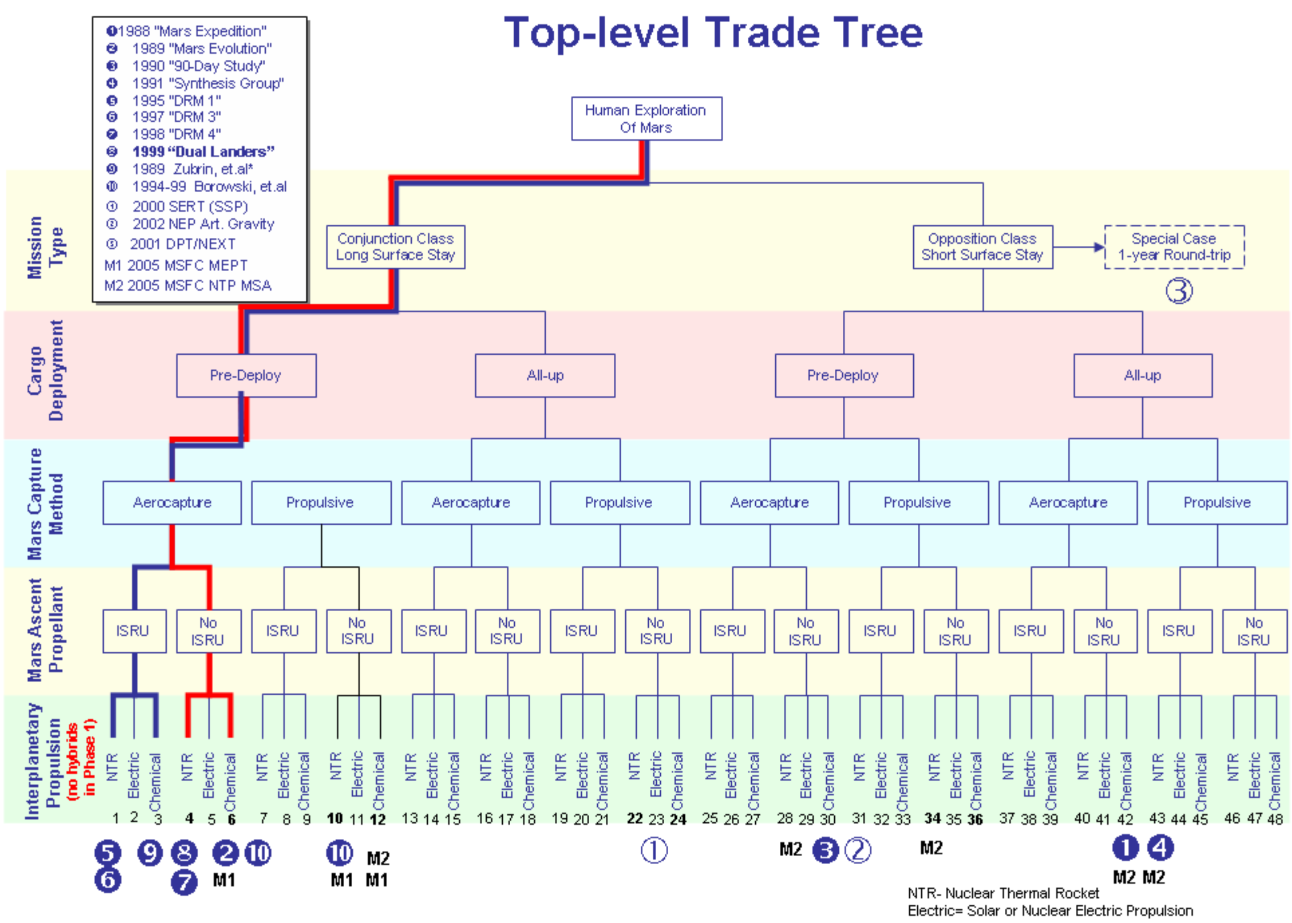


**Figure 3-44.** In-situ resource utilization decision trade tree options.

### 3.6.2 In-situ resource utilization operational concept

It is important to note that the use of ISRU for ascent propellant necessitates a different operational concept than sending a fully fueled MAV to Mars. Figure 3-45 illustrates this difference. If *not* employing ISRU, both the MAV and the SHAB would be sent to Mars 2 years before the launch of the crew. The SHAB would land at the target site while the MAV would loiter in orbit awaiting crew arrival over 2 years later. The crew members would then ride the MAV to the surface and be assured that they had a fully fueled ascent vehicle that is capable of ascending back to orbit if necessary. In the case *with* ISRU, a MAV with an un-fueled ascent vehicle and the SHAB is sent to Mars 2 years before the crew. In this case, the MAV would be the vehicle that would first land at the target site and then begin creating propellant while the SHAB would loiter in orbit awaiting crew arrival. Once the MAV sends a signal to Earth indicating that its propellant tanks are full, the crew would be launched. Upon arrival at Mars, the crew would transfer to the SHAB and ride it down to a location close to the MAV. A basic manifest of major items is also listed on the right side of the figure.

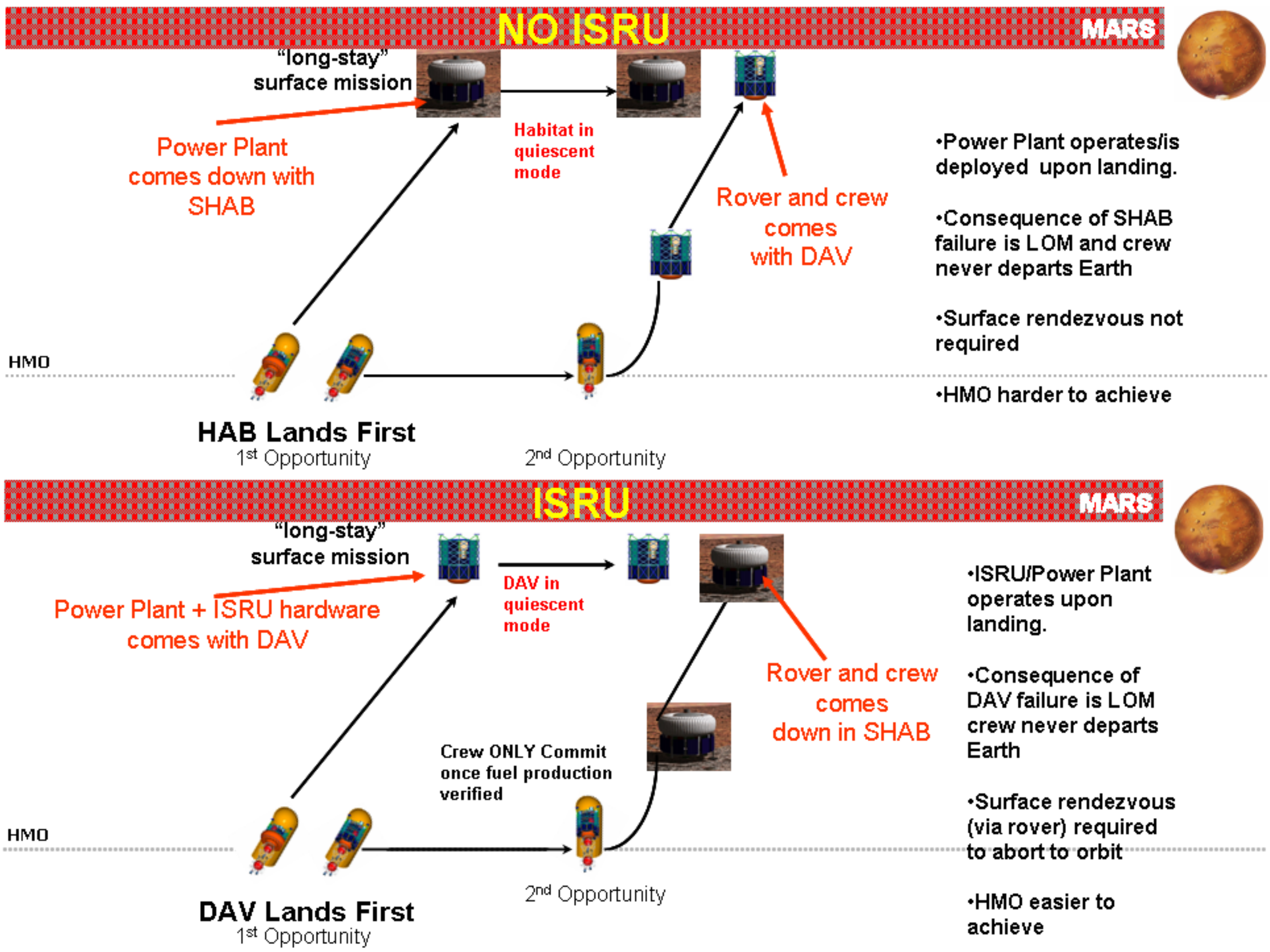


**Figure 3-45.** In-situ resource utilization operational concept.

### 3.6.3 In-situ resource utilization and aborts: an emphasis on abort to the surface

One perceived drawback of the ISRU propellant strategy is the lack of ATO capabilities inherent in the ISRU propellant-derived vehicle. The key leverage of the in-situ propellant production strategy is derived from the fact that ascent propellants are made at the planet (in-situ), thus dramatically reducing the overall transportation mass that is required. This results in a lander vehicle that cannot perform ATO maneuvers during the landing sequence. The ATO strategy has been a risk-reduction philosophy that has been followed since the early days of human exploration. During critical mission maneuvers, abort strategies with well-defined gates and sequences are established such that, if warranted, they can be exercised to place the crew in a stable position; i.e., in orbit. With the Mars in-situ propellant production strategy, ATO scenarios do not exist since the ascent propellants are produced on the martian surface and are not transported with the crew.

This lack of ATO capability inherent with in-situ propellant production has led many scientists to discount the overall strategy of ISRU. During development of Design Reference Architecture 5.0, the specific question of ATO was raised. The EDL community reviewed the typical entry sequence and concluded that, due to the physics involved during the atmospheric entry phase, ATO was probably not possible, and if it were required it would only be available during the final portion of the entry sequence, namely the terminal phase after separation from the aeroshell had occurred near the surface. At that point, the most critical phases of the entry maneuver have been completed. Thus, emphasis of the EDL philosophy changed from one of ATO to an *abort-to-surface strategy*; i.e.,, provide enough functionality and reliability in the EDL system to enable a surface rendezvous. In this sense, surface rendezvous must be within a distance that accessible by the crew, which includes the distance that a rover, taken with the crew, can reach.

### 3.6.4 Key findings

As part of the comprehensive ISRU analysis, several key findings were provided to summarize the numerous trades that were conducted and to provide guidance to decision-makers.

They key findings are as follows:

- The mass of an EDL system that is required to land a fully fueled six-person DAV may be prohibitive: ISRU production of ascent vehicle oxidizer ($O_2$) may be mission enabling

- In-situ production of $O_2$ and inert gases for life support ($N_2$ and Ar) allows mass savings on the SHAB by enabling closure of the environmental control and life support system (ECLSS)/EVA system or, at the very least, providing functional redundancy for the ECLSS and increased surface EVA mission flexibility

- The mass savings that are realized by in-situ production of ascent vehicle oxidizer (>25 t) can be used for delivery of a larger power plant and a large pressurized rover, or for additional margin
    - Production of ascent fuel with $H_2$ from Earth was ruled out due to the large increase in DAV size (~30 $m^2$)
    - Production of ascent fuel and life support consumables with $H_2O$ that is extracted from hydrated minerals is a promising avenue to investigate in continuing ISRU studies

- Nuclear power is significantly more mass and volume efficient, and potentially less operationally complex, in meeting ISRU propellant production needs in the martian environment

- ISRU results in fewer launches, the number of launches being a large contributor to overall risk and cost

### 3.6.5 In-situ resource utilization trades performed

The ISRU and mission trades that were performed in DRA 5.0 are more comprehensive then any performed in any previous human Mars mission study. Extra attention and effort was made toward comparing ISRU and non-ISRU missions on an equal basis, especially with respect to the impact of Mars rendezvous orbits. ISRU propellant production has two main influences on mission architectures: (1) reducing the mass of the lander, and (2) reducing the propulsive needs for both Mars capture and Mars departure by enabling higher rendezvous orbits compared to non-ISRU missions. Previous mission evaluations covered the first main influence but ignored the second main influence. Also, since the last human Mars mission study was performed, NASA and European Space Agency (ESA) orbital and surface robotic missions have determined that water is globally available in the Mars soil in varying concentrations and depths. Therefore, for the first time, Mars water was considered as a potential resource in this study with and without the use of Mars atmosphere resources. For DRA 5.0, several trade options for ISRU and their impact on mission mass/volume as well as optimum technology/ISRU process were evaluated to understand all of the potential mission implications and benefits of incorporating ISRU into human Mars exploration plans. These trades included the following:

- Mission consumables of interest: life support/EVA-only, or propellant and life support
- What, if any, consumable or reagent must be brought from Earth: $H_2$ for $H_2O$ and/or fuel production, or bring ascent fuel vs. making it in situ
- Mars resource of interest: atmosphere only or both atmosphere and soil/$H_2O$
- Power system associated with ISRU processing: solar or nuclear.

The trade options that were considered were organized into a trade tree (shown in figure 3-46). ISRU process systems and subsystems were developed for each of the trade tree branches. For atmospheric processing options, previous models were updated with new technology and hardware performance information. For Mars soil/$H_2O$ processing, new models for excavation and soil processing were created from recently created lunar ISRU regolith and processing models with Mars soil/$H_2O$ parameters applied. While not perfect, this allowed for first-order evaluation of system mass, volume, and power associated with Mars water resource collection.

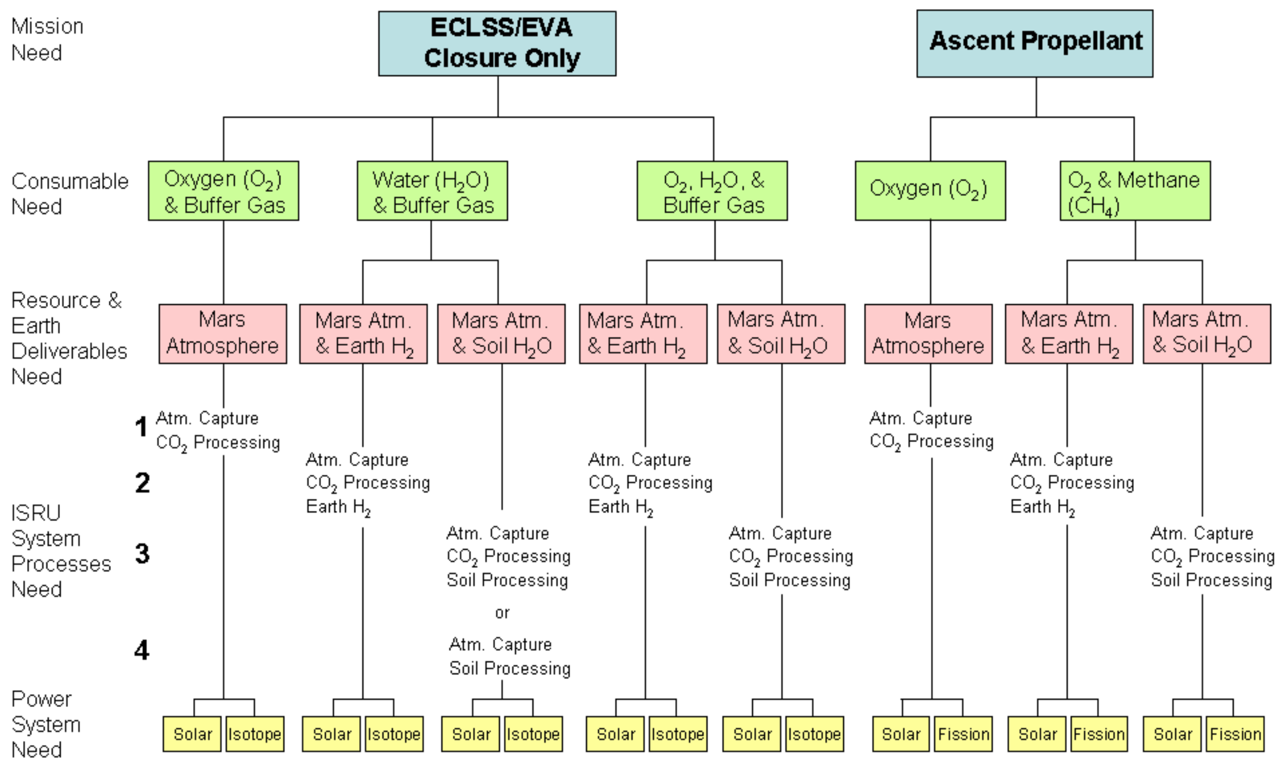


**Figure 3-46.** In-situ resource utilization trade tree.

### *3.6.5.1 Ground rules and assumptions*

To perform ISRU system trade studies and calculate system option mass, power, and volume, mission GR&As had to be established. For DRA 5.0, the common ISRU ground rules and assumptions that were used were the following:

- All mission consumable production was completed prior to crew departure from Earth. This allowed for 330 sols of operation: (300 days with 30 days' contingency)
- Surface crew size of six for 550 sols surface stay operations
- Sizing and risk-reduction philosophy:
  - Each ISRU system is single-fault-tolerant
  - Two ISRU systems are flown, each sized to complete the mission on their own
  - ISRU subsystems/concepts should be previously demonstrated on a robotic precursor mission at relevant scale and operation duration
- Nuclear power is used for continuous (all-day) operation of ISRU systems, and solar power is used for daylight-only (8 hours/day) operation

### *3.6.5.2 Mission consumables, resources, and Earth feedstock required*

Because ISRU has never been demonstrated before in an actual mission, mission planners have a major concern that that the risk of ISRU system failure in a critical mission role may outweighs the benefits. Therefore, at the start of DRA 5.0, two mission approaches were considered: (1) only provide consumables to close the life support system and enable extensive surface EVAs, and (2) produce ascent propellants and consumables for life support and EVAs. Because current EVA suits are open-loop systems that consume both $O_2$ for crew breathing and $H_2O$ for cooling, extensive EVAs by a crew of six over 500 sols on Mars require more $O_2$ and $H_2O$ than can easily be manifested. Based on DRA 5.0 mission models and assumptions for vehicles and surface operations, estimates for how much $H_2O$, $O_2$, and fuel were calculated to determine ISRU system mass, volume, and power. The amount of buffer gas that is required ($N_2$/Ar) was based on habitat leakage and airlock usage loss estimates. Also, since some consumables and ISRU processing options require reagent feedstock delivery from Earth, the type and amount of feedstock for each consumable need and Mars resource assumed was also determined. Table 3-18 depicts the range of $H_2O$, $O_2$, and fuel production needs for the two ISRU mission scenarios that were considered.

**Table 3-18.** Mission Consumable and Feedstock Needs (in kilograms)

| **Close Crew & EVA Consumables** | **$O_2$** | **$CH_4$** | **$H_2O$** | **$N_2$/Ar** | **Earth $H_2$** | **Earth $CH_4$** | **Comment** |
|---|---|---|---|---|---|---|---|
| Mars Atm. Processing only | 1906 | NA | | 133 | 399 | | – Closes $H_2O$ through making $H_2O$ and shortfall of $H_2$ closure brought from Earth |
| Mars Soil Processing only | | NA<br>NA | 2146<br>3586 | 133<br>133 | 160 | | – Closes $H_2O$ and shortfall of $H_2$ closure brought from Earth<br>– Closes $H_2O$ and $H_2$ shortfall through in-situ $H_2O$ only |
| Mars Atm. & Soil Processing | 1281 | NA | 2146 | 133 | | | – Closes $H_2O$ and covers $O_2$equivalent to $H_2$ closure shortfall |

| **Propellants & Crew Consulables** | $O_2$ | $CH_4$ | $H_2O$ | $N_2$/Ar | Earth $H_2$ | Earth $CH_4$ | Comment |
|---|---|---|---|---|---|---|---|
| Mars $O_2$ Propellant Production Only | 24891 | | | 133 | 399 | 6567 | – Requires tank and cryocooler for $H_2$ delivery and filled ascent $CH_4$ tank |
| Mars $O_2$/$CH_4$ Production w/Earth $H_2$ | 24891 | 6567 | | 133 | 2069 | | – Requires tank and cryocooler for $H_2$ delivery |
| Mars $O_2$/$CH_4$ Production w/Mars $H_2O$ | 24891 | 6567 | 16788 | 133 | | | – Requires excavation and soil processing |

It should be noted that the amount of $O_2$ and $CH_4$ that was required for ascent propulsion can vary depending on lander size and payload. The amount that is specified in the table should not be considered exact. Propellant quantity calculated were based on a notional two-stage Mars ascent vehicle of 40 t (wet mass) with a total ascent delta-V of 5625 m/s and using pump-fed $LO_2$/$CH_4$ engines with a mixture ratio of 3.5:1 and $I_{sp}$ of 369 seconds.

#### *3.6.5.3 Mars atmosphere-based options*

Production of $O_2$, $CH_4$, buffer gases, and $H_2O$ are all possible from Mars atmosphere resources with Earth-supplied $H_2$ required for $H_2O$ and/or $CH_4$ production. Excess $CH_4$ is produced since production and propulsion mass ratios of $O_2$ and $CH_4$ are not equivalent with Earth-supplied $H_2$ feedstock.

The Mars atmosphere is primarily $CO_2$ (95.5%), $N_2$ (2.7%), and Ar (1.6%). The significant benefit of Mars atmosphere $CO_2$, $N_2$, and Ar as a resource is that it is available globally at known concentrations. Nitrogen and Ar are very good buffer gases for crew breathing as well as purge gases for science experiments. (**Note:** Further assessments of the use of Ar should be conducted to ensure that it is a suitable buffer gas.) Carbon dioxide is a good source of both $O_2$ and carbon (C) for the production of $O_2$, $CH_4$, and other hydrocarbons that may be of interest. Pressurized gases (whether the bulk atmosphere or separated $CO_2$, $N_2$, or Ar) are also beneficial for inflating habitats and structures as well as use in science experiments and cleaning dust off of surfaces and sensitive areas. However, the Mars atmosphere is at low pressure (~0.1 pounds per square inch, absolute (psia)) at the surface. Before Mars atmospheric $CO_2$ can be used or processed, it must be collected, separated, and pressurized; typically at or above Earth ambient pressure (>14.7 psia) to increase the efficiency of $CO_2$ processing concepts.

Three primary methods for $CO_2$ collection and pressurization have been evaluated: mechanical pumps, micro-channel adsorption, and cryogenic separation ($CO_2$ freezing). To deliver $CO_2$ to a processing unit, a >100:1 compression ratio is required from Mars atmospheric pressure to $CO_2$ processing unit pressure. Since most mechanical pumps are efficient up to around 10:1 compression ratios, a two-stage compressor is required. While mechanical pump technology is very mature, the pumps can be very heavy and power intensive. Also, mechanical pumps can not be used on their own to separate $CO_2$ from other atmospheric gases. Microchannel adsorption is a technology that was developed by the Department of Energy (DOE)/Pacific Northwest National Laboratory (PNNL) that uses small beds with rapid adsorption/desorption cycles to minimize pressure drop and diffusion-limited capacity loss during collection, separation, and pressurization of $CO_2$ from the bulk Mars atmosphere. Microchannels allow for rapid heat exchange and high surface area to volume beds to make this approach feasible. The approach is attractive since it is very mass and power efficient, provides good separation of $CO_2$ from other gases, and is compatible with habitat and EVA life support system operations and designs. Cryogenic separation is based on the fact that the temperature difference between solid and gaseous $CO_2$ at Mars atmospheric pressure is very close to Mars nighttime temperatures. Therefore, a $CO_2$ freezer (solidification pump) with active cooling (cryocooler) to lower the atmospheric gas temperature in the pump to below 150 K (–123 °C) will solidify $CO_2$. The frozen $CO_2$ can then be heated in a controlled volume to supply the $CO_2$ at any desired inlet pressure for subsequent processing. The solidification pump is attractive because it allows small-volume and high-pressure $CO_2$ delivery, and can potentially simplify system design and development costs by sharing cryocooler hardware with the $O_2/CH_4$ propellant liquefaction and storage system.

Conversion of atmospheric $CO_2$ into $O_2$ can be performed in a number of different ways, depending on the resources that are available and the products that are desired. The three processes that have been examined the most due to process simplicity or commonality with life support systems are solid oxide $CO_2$ electrolysis (SOCE), Sabatier conversion of $CO_2$ to $CH_4$ and $H_2O$ (with subsequent $H_2O$ electrolysis), and reverse water gas shift (RWGS) conversion of $CO_2$ to CO and $H_2O$ (with subsequent $H_2O$ electrolysis). For both Sabatier and RWGS conversion of $CO_2$, $H_2$ is required. In the case of $O_2$ production using RWGS, the $H_2$ that is required is obtained from the subsequent $H_2O$ electrolysis, so $H_2$ is recycled. In the case of $O_2$ production using Sabatier, only half of the $H_2$ that is needed is recovered from the subsequent $H_2O$ electrolysis process. When considering Mars atmospheric resources alone, the remaining half of the $H_2$ must be obtained either from Earth (if $CH_4$ production is desired) or by processing the $CH_4$ that is produced to recover the $H_2$ if $O_2$-only production is desired. For any missions requiring $H_2O$ and/or $CH_4$ production on Mars using only atmospheric resources, $H_2$ delivery from Earth is required. The Sabatier process is the only ISRU process option that makes $CH_4$ fuel; however, $O_2$ and $CH_4$ are produced at a 2:1 $O_2$ to $CH_4$ mass ratio with Earth-supplied $H_2$. Since propulsion systems require $O_2$ to $CH_4$ mass ratios of between 3:1 and 4:1, excess $CH_4$ is produced. While the $H_2$ that is required may be a relatively low mass compared to $H_2O$ or $CH_4$ that is delivered from Earth, it requires over three times the volume. Table 3-19 depicts the three main $CO_2$ processing options and reaction equations.

**Table 3-19.** In-situ Resource Utilization Atmosphere Process Options and Reactions

| ISRU Process | Reaction |
|---|---|
| Sold Oxide $CO_2$ Electrolysis | $2CO_2 \rightarrow 2CO + O_2$ |
| Sabateir with Water Electrolysis* | $CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$<br>$2H_2O \rightarrow 2H_2 + O_2$ |
| Reverse Water Gas Shift with Water Electrolysis | $2CO_2 + 2H_2 \rightarrow 2CO + 2H_2O$<br>$2H_2O \rightarrow 2H_2 + O_2$ |

*Note imbalance between $H_2$ required in Sabateir vs. that produced from water electrolysis (WE).

Selection of the atmosphere collection, separation, and processing hardware was based on life support and EVA processing that need calculations. Table 3-20 depicts the mass, volume, and power that are associated with each subsystem option for atmospheric processing for both nuclear and solar power options. The combined atmosphere collection and processing system that was deemed best from a mass, power, and volume perspective was the rapid cycle adsorption pump (RCAP) with SOCE system. Because of technology development risk discussed in section 7.7 and potentially better commonality with life support system development, the RWGS with WE system was recommended as the backup option.

It should be noted that the amount of Mars atmosphere that is processed is a function of how much $CO_2$ is required. Therefore, although a notional amount of buffer gas replenishment was calculated (see table 3-20), the amount of $CO_2$ that is required will allow for a greater amount of buffer gas collection. This extra amount is listed in table 3-20.

**Table 3-20.** Atmosphere Processing Subsystem Sizing Estimates

| ISRU for Crew/EVA Consumables Only | (8-hour Operation) | | | (24-hour Operation) | | |
|---|---|---|---|---|---|---|
| | Mass (kg) | Power (kW) | Vol. (m) | Mass (kg) | Power (kW) | Vol. (m) |
| **$O_2$ and $N_2$/Ar Only & Bring $H_2$ from Earth** | | | | | | |
| RCAP – RWGSWE | 69 | 8.00 | 0.02 | 24 | 2.82 | 0.01 |
| MP – RWGS/WE | 195 | 7.28 | 0.32 | 66 | 2.58 | 0.11 |
| CFP – RWGS/WE | 135 | 7.44 | 0.20 | 47 | 2.63 | 0.10 |
| RCAP – SOCE | 39 | 4.70 | 0.05 | 16 | 1.58 | 0.03 |
| MP – SOCE | 217 | 3.76 | 0.44 | 73 | 1.25 | 0.16 |
| CFP – SOCE | 135 | 3.96 | 0.29 | 48 | 1.37 | 0.14 |
| $O_2$ liquefaction power | | 0.67 | | | 0.24 | |
| Amount of extra $N_2$/Ar | 103 | | | 103 | | |
| Total $N_2$/Ar storage | 250 | 0.02 | 0.60 | 250 | 0.01 | 0.60 |
| $H_2$ delivery from Earth ($H_2$ plus tank) | 580 | 0.34 | 8.60 | 580 | 0.34 | 8.60 |
| Total (Based on RCAP-SOCE) | 972 | 5.06 | 9.25 | 949 | 1.92 | 9.23 |

#### *3.6.5.4 Investigation Mars surface water-based options*

Production of $O_2$ and $H_2O$ using Mars surface $H_2O$ resources (hydrated minerals) also seems promising, but requires more extensive study (and scientific data from Mars) prior to a final recommendation. Buffer gas and $CH_4$ production are not possible from Mars $H_2O$ alone.

Some robotic missions to Mars have shown that $H_2O$, in the form of hydrated minerals, can be found globally across the Mars surface. In equatorial regions (±30), the Viking missions measured 1% to 3% $H_2O$ by mass, and Mars Odyssey mission data suggest regions with up to 8%–10% $H_2O$ by mass in the top 1 m. Mars Odyssey data also suggest that subsurface ice table may be within the top few meters in some localities in the mid latitudes (40° to 55°), near-surface subsurface ice tables may be widely prevalent at the high latitudes (55°–70°), and >50% water ice by mass is at or near the surface in the polar regions (+70°). To be conservative in estimating benefits and to minimize forward/back contamination and search for life issues, only surface soil/$H_2O$ (hydrated minerals) were considered as a potential resource. No ice or subsurface $H_2O$ reservoirs were considered. Also, since Viking is the only mission to date to provide "ground truth" measurements of $H_2O$ content, a 3% $H_2O$ by mass assumptions was used in sizing calculations with an evaluation of 8% $H_2O$ by mass to understand $H_2O$ content impact on system sizing. It is believed by experts that hydrated minerals and gypsum may be widely available at $H_2O$ concentrations between 20% and 30% at sites of science exploration interest. Since $H_2O$ can be electrolyzed, it can be used to make $O_2$ for life support and propulsion without requiring atmospheric resources. Methane production is not possible with $H_2O$-based options alone.

Like Mars atmospheric $CO_2$, before Mars $H_2O$ can be processed, it must first be collected and separated from the Mars soil. This is performed by a two-step process: excavate and deliver soil to a processing plant, and process soil to separate and collect the $H_2O$. To excavate and deliver Mars soil for processing, an excavation system model was developed that was based on lunar excavation concepts. The model evaluated the number and size of excavators, distance of the excavation from the processing plant, Mars soil and $H_2O$ content properties, and amount of time that is available for operations. Excavation and material property experience and data from the use of the arm/scoop on Mars Viking, wheel/soil interaction behavior and experience from the Sojourner and MERs, and potentially new excavation and surface material property data in the polar region from the Phoenix lander arm/scoop are good starting points for future Mars excavation and material transport development efforts. During the brief Mars DRA 5.0 study, a good amount of time was spent on trying to better understand and define Mars soil parameters for possible locations of scientific and human exploration interest including: $H_2O$ content, cohesion, internal friction angle, bulk density, compressive strength, and tool-soil adhesion. Force and structural calculations to size the excavation subsystem components on the vehicle (dump bin and digging tool) assumed a simple bucket excavator concept. To be as conservative as possible in this first evaluation of Mars soil/$H_2O$ resource processing and to minimize forward contamination and search for life issues, only $H_2O$ in the top surface was considered and only at 3% to 8 % concentration. No hydrated gypsum, permafrost, ice, or subsurface reservoirs were considered. Based on mission consumable need estimates, Mars $H_2O$ extraction that is to be used for propellant production for crewed missions requires that excavation and soil processing systems must be designed to excavate and process 77 kg (3% $H_2O$ content) to 30 kg (8% $H_2O$ content) every hour.

An $H_2O$ extraction unit that is based on recent work on $H_2$-reduction reactors for lunar $O_2$ production from regolith is used to extract adsorbed $H_2O$ from the martian soil. The unit consists of an inlet/outlet hopper, inlet/outlet auger, two soil reactors, two gas clean-up modules, and two $H_2O$ condensers. The inlet/outlet hopper and auger are used to receive soil from the excavator/hauler vehicle and transfer soil in and out of the reactors, respectively. Once the soil is in the reactor, it is heated to approximately 600 K. An inert gas flow fluidizes the soil to aid desorption of $H_2O$. The inert-water gas stream is sent to a gas clean-up process to remove any contaminants that are evolved during the process. The $H_2O$ is then collected on a condenser, which is actively cooled by a cryocooler. While lunar material is much drier compared to expected Mars soil, there are enough similarities in design and operation of the lunar $H_2$ reduction reactors for $O_2$ extraction that models and experience that are gained from this effort should benefit future Mar soil processing and $H_2O$ extraction/separation development efforts. Water electrolysis alone is required for processing to produce mission-consumable $O_2$. Hydrogen that is produced during the electrolysis process is vented.

Selection and sizing of the atmosphere collection and separation hardware is based on results in section 3.6.5.5. Table 3-21 depicts the mass, volume, and power that are associated with hardware for excavation, transportation, and soil processing to extract water for both nuclear and solar power options. Excavation and soil processing estimates are based on 3% water content and 1,000 kg/m$^3$ soil density. More work needs to be done to verify the mass estimates of the support equipment (e.g., the mass associated with excavators and consumables transport).

**Table 3-21.** Soil/Water Processing Subsystem Sizing Initial Estimates

| ISRU for Crew/EVA Consumables Only | 8-hour Operation | | | 24-hour Operation | | |
|---|---|---|---|---|---|---|
| | Mass (kg) | Power (kW) | Vol. (m) | Mass (kg) | Power (kW) | Vol. (m) |
| **Water Only & Bring $H_2$ from Earth** | | | | | | |
| $N_2$/Ar Extraction (no extra) | 13 | 1.75 | 0.00 | 5 | 0.58 | 0.00 |
| Excavators (2) | 425 | 1.01 | 2.23 | 350 | 0.81 | 1.53 |
| Soil Processor for $H_2O$ Extraction | 266 | 3.43 | 9.32 | 219 | 2.89 | 3.26 |
| Total $N_2$/Ar storage | 155 | 0.01 | 0.40 | 142 | 0.00 | 0.40 |
| $H_2$ delivery from Earth | 94 | 0.25 | 3.79 | 94 | 0.25 | 3.79 |
| Total | 954 | 6.44 | 15.75 | 810 | 4.53 | 8.89 |
| **Water Only** | | | | | | |
| $N_2$/Ar Extraction (no extra) | 13 | 1.75 | 0.00 | 5 | 0.58 | 0.00 |
| Excavators (2) | 541 | 1.34 | 3.63 | 444 | 1.06 | 2.46 |
| Soil Processor for $H_2O$ Extraction | 324 | 4.33 | 15.41 | 246 | 3.23 | 5.29 |
| Total $N_2$/Ar storage | 155 | 0.01 | 0.40 | 142 | 0.00 | 0.40 |
| Total | 1,033 | 7.42 | 19.45 | 837 | 4.87 | 8.15 |

### *3.6.5.5 Mars atmosphere and water-based options*

Production of $O_2$, $CH_4$, buffer gases, and $H_2O$ are all possible from the combination of Mars atmosphere and surface $H_2O$ resources at any quantities.

The combined atmosphere and water-based ISRU system consists of the excavator and soil processing units, and atmospheric collection and Sabatier/water electrolysis processes. While the combined system requires more hardware and power infrastructure than either option alone, it provides much more mission flexibility and, potentially, much more mass/volume savings than either option alone. For atmospheric processing alone, the amount of $H_2O$ that is produced in situ is limited by the amount of $H_2$ that is brought from Earth. Therefore, if more $H_2O$ is required due to failures or performance degradation, atmospheric processing alone will not allow recovery of inadequate $H_2O$ supply. Water processing alone allows for any in-situ production need for both $O_2$ and $H_2O$, but does not allow for production of $CH_4$ fuel for fuel cells, surface hopping, or ascent to Mars orbit. Combining both atmosphere and water-based ISRU allows for all consumable needs and production flexibility to overcome unforeseen shortfalls in mission consumables.

Even when assuming only 3% $H_2O$ by mass in the Mars soil, this option was extremely competitive. Since these calculations were based on first-order approximations and conversion of lunar excavation and soil processing models to Mars application and predicted soil/ $H_2O$ properties, further work is required to fully evaluate this combined option.

Using process selection and sizing of the atmosphere collection, separation, and processing hardware and excavation and soil processing hardware from the last two sections, table 3-22 depicts the mass, volume, and power that are associated with combined atmosphere and soil/$H_2O$ processing ISRU systems for both nuclear and solar power options. Excavation and soil processing estimates are based on 3% $H_2O$ content and 1,000 kg/m$^3$ soil density.

**Table 3-22.** Combined Atmosphere and Soil/Water Processing Subsystem Sizing Estimates

| ISRU for Crew/EVA Consumables Only | 8-hour Operation | | | 24-hour Operation | | |
|---|---|---|---|---|---|---|
| | Mass (kg) | Power (kW) | Vol. (m) | Mass (kg) | Power (kW) | Vol. (m) |
| **ISRU $O_2$ and** $H_2O$ | | | | | | |
| $O_2$ Production from Mars Atm. | 29 | 3.17 | 0.04 | 11 | 1.06 | 0.02 |
| Excavators (2 full for redundancy) | 425 | 1.01 | 2.23 | 350 | 0.81 | 1.53 |
| Soil Processor for $H_2O$ Extraction | 266 | 3.43 | 9.32 | 219 | 2.89 | 3.26 |
| $O_2$ liquefaction power | | 0.45 | | | 0.17 | |
| Total $N_2$/Ar storage | 155 | 0.01 | 0.40 | 142 | 0.00 | 0.40 |
| Total | 876 | 8.07 | 11.99 | 722 | 4.92 | 5.21 |

### 3.6.6 Nuclear vs. solar power for Mars in-situ resource utilization

As was stated in the ISRU GR&As, all mission consumables that are produced by ISRU systems must be completed before the crew leaves Earth. Therefore, during the period prior to crew landing, power must be provided to process in-situ resources either for crew and EVA consumables only or for propellant in addition to crew/EVA consumables. For the case of nuclear fission power, it is assumed that the ISRU plant would be operated continuously for a period of at least 300 sols to produce the necessary resources. In the case of solar power, the total energy would be the same, but operation of the ISRU plant would be limited to 8 hours per day at three times the power level of the nuclear case. This daytime-only operation avoids the need for large quantities of fuel cell reactants that would be needed to provide around-the-clock production, but may result in production inefficiencies that could require additional margin for the solar power case.

The power requirement for the consumables-only atmosphere-only ISRU case is 2 kWe continuous or approximately 5 kWe for 8 hours/day operations. The mass of a single 5-kWe photovoltaic (PV) module is estimated to be 1,980 kg (including 20% contingency). A 2-kWe nuclear dynamic isotope power system (DIPS) is estimated to be approximately 200 kg.(with 20% contingency). From previous Mars ISRU DRM studies, it was known that production of large amounts of $O_2$ and fuel would be power intensive. Previous studies estimated 20 to 40 kWe were required, and although new $H_2O$ electrolysis and liquefaction technologies could help reduce previous power estimates, drastic reductions were not anticipated. ISRU production of $O_2$ for life support and propulsion is estimated at 26 kWe for continuous and approximately 80 kWe for 8-hour operations. The addition of 450 m$^2$ of solar array to accommodate $O_2$ propellant production would increase this overall power system mass from 1,980 kg to about 12,500 kg (including 20% contingency). Because of the large power that is required for $O_2$ production, a fission surface power system (FSPS) is required instead of a DIPS for continuous operation. The mass of a FSPS is a variable with power output, which is primarily based on the size of radiator that is needed to reject waste heat. The estimated mass for a 23-kWe reactor that might be used for habitat operation is 7,300 kg (including 20% contingency). The mass for a 30-kWe reactor that could accommodate propellant ISRU is estimated at about 8,000 kg. Figure 3-47 depicts the power estimates for solar vs. nuclear power for ISRU $O_2$ production and the power estimates for habitat operations.

Besides mass differences, there is significant concern for solar power systems due to deployment difficulties and degradation due to dust settling and potentially lengthy dust storms. Because ISRU operations need to occur before the crew leaves Earth, no crew will be available for deploying and cleaning solar array power systems. Based on both the mass savings and lower risk due to deployment and dust degradation issues, nuclear power was selected for all ISRU mission options.

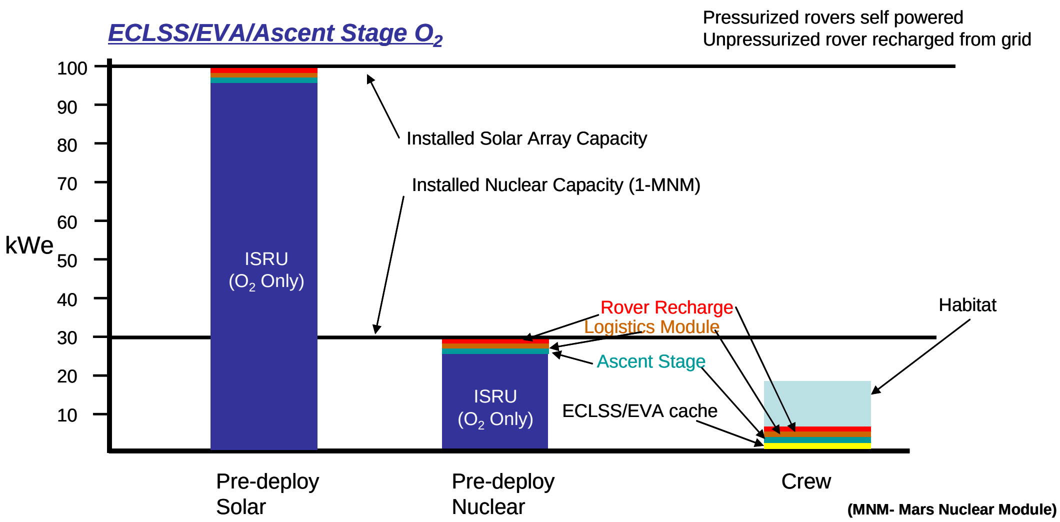


**Figure 3-47.** ISRU and habitat power estimates for solar and nuclear.

### 3.6.7 In-situ resource utilization trade study results

Once process options for atmospheric collection and $CO_2$ processing were selected, an evaluation of the three main ISRUs for production of all consumables was performed:

- Make $O_2$ and $H_2O$ from the atmosphere and Earth $H_2$ but bring $CH_4$ fuel
- Make $O_2$, $H_2O$, and $CH_4$ fuel from the Mars atmosphere and Earth $H_2$
- Make $O_2$, $H_2O$, and $CH_4$ fuel from the Mars atmosphere and soil/$H_2O$

Mass, power, and volume calculations for the three main ISRU options for meeting all mission consumable needs are depicted in table 3-23 and are graphically shown in figure 3-48. These estimates assume that nuclear power is available for continuous operation. Production of $O_2$ alone is the highest mass (because it requires delivery of $CH_4$ fuel from Earth), but also has the lowest volume and power impact to the mission. Production of $O_2$ and $CH_4$ from atmosphere resources and Earth $H_2$ requires lower mass than $O_2$ production alone; however, the volume is significantly higher than all other options. Production of both $O_2$ and $CH_4$ with atmosphere and soil/$H_2O$ resources is the lowest mass but highest power option. It should be noted that by changing the soil $H_2O$ content from 3% to 8%, there was a significant reduction in both volume and power associated with the combined atmosphere and soil resource ISRU option.

**Table 3-23.** In-situ Resource Utilization Process Mass, Power, and Volume

| **Ascent to 250 × 33,793 km Mars Orbit (Delta-V = 5,625 m/s)** | **Mass (kg)** | **Power (kW)** | **Vol. ($m^3$)** |
|---|---|---|---|
| *$O_2$ Only for Propulsion w/Earth $CH_4$* | | | |
| Plant Type: Zirconia Cells | | | |
| $CH_4$ from Earth | 6,567 | 0.02 | |
| $H_2$ delivered from Earth | 399 | | |
| Tank and cryocooler for $H_2$ delivery from Earth* | 201 | 0.37 | 8.62 |
| Total $N_2$/Ar storage | 285 | 2.85 | 0.80 |
| ISRU Atm. Processing Plan | 47 | 19.65 | 0.11 |
| Liquefaction Subsystem | 13 | 3.19 | 0.06 |
| Total without $CH_4$ | 945 | 26.08 | 9.59 |
| **Total** | **7,512** | **26.08** | **9.59** |
| *$O_2$/$CH_4$ Propellant for Propulsion w/Earth $H_2$* | | | |
| Plant Type: Sabatier/Zirconia Cells | | | |
| $H_2$ delivered from Earth | 2,069 | | |
| Tank and cryocooler for $H_2$ delivery from Earth* | 541 | 0.75 | 30.02 |
| Total $N_2$/Ar storage | 285 | 2.85 | 0.80 |
| ISRU Atm. Processing Plan | 356 | 29.70 | 0.25 |
| **Total** | **3,251** | **33.30** | **31.07** |
| *$O_2$/$CH_4$ Propellant for Propulsion w/Mars $H_2O$ (3%)* | | | |
| Plant Type: Sabatier Water Electrolysis | | | |
| Amount of $H_2O$ Needed to Make Propellant and ECLSS | 16,788 | | |
| Soil Excavators (2) | 1,183 | 1.53 | 11.48 |
| Soil/$H_2O$ Extraction Plant (3%) | 615 | 31.90 | 7.05 |
| Total $N_2$/Ar storage | 285 | 2.85 | 0.80 |
| ISRU Atm. Processing Plan | 545 | 23.11 | 0.61 |
| Liquefaction Subsystem | 30 | 4.38 | 0.18 |
| **Total** | **2,658** | **63.77** | **20.12** |
| *$O_2$/$CH_4$ Propellant for Propulsion w/Mars $H_2O$ (8%)* | | | |
| Plant Type: Sabatier Water Electrolysis | | | |
| Amount of $H_2O$ Needed to Make Propellant and ECLSS | 16,788 | | |
| Soil Excavators (2) | 704 | 0.80 | 4.28 |
| Soil/$H_2O$ Extraction Plant (8%) | 474 | 15.81 | 4.11 |
| Total $N_2$/Ar storage | 285 | 2.85 | 0.80 |
| ISRU Atm. Processing Plan | 527 | 24.26 | 0.49 |
| Liquefaction Subsystem | 30 | 4.38 | 0.18 |
| **Total** | **2,021** | **48.10** | **9.86** |

*$H_2$ cryocooler to prevent boiloff during 8-month trip.

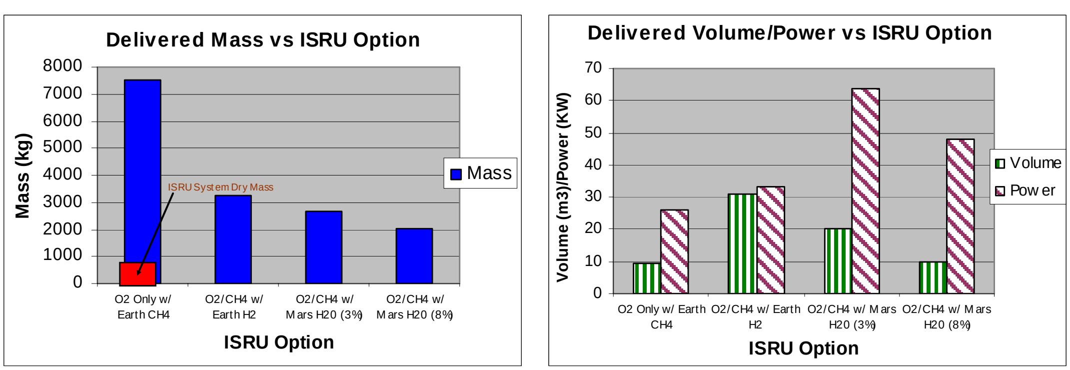


**Figure 3-48.** Mass, power, and volume of in-situ resource utilization strategies.

Besides mass, power, and volume calculations, technology readiness and resource uncertainty are also important aspects to consider when selecting the best ISRU option for future human exploration missions. Significant work has been performed on Mars atmosphere processing technologies and systems (see section 7.7). Also, as noted previously, the Mars atmosphere is globally available at known concentrations and can therefore be assumed to be available no matter where future human missions may land on Mars. While past and current Mars robotic missions are finding out more and more information about soil properties and $H_2O$ content on Mars, much more information is needed on soil physical and mineral characteristics, compaction, density, and excavation forces; and $H_2O$ content as a function of type, depth, and local distribution is required before it can be realistically considered as a resource for mission-critical consumable production. Also, while Viking landers (and soon the Phoenix lander) have excavated and heated Mars surface soil, more technology and system development work is required to raise the TRL of Mars soil/ $H_2O$ ISRU to an acceptable level for serious mission mass, power, and volume estimates to be performed. Although work on lunar regolith processing for $O_2$ extraction is providing confidence that Mars soil excavation and $H_2O$ extraction processing is realistically feasible, work specific to Mars is required. Table 3-24 lists the strengths and weaknesses that are associated with atmosphere and soil resource ISRU options.

**Table 3-24.** ISRU Atmosphere and Soil Resource Processing Strengths and Weaknesses

| | Atmosphere Resource Processing | Soil Resource Processing |
|---|---|---|
| **Strengths** | Atmosphere resources are globally obtainable (no landing site limitations) | Surface material characteristics being studied by robotic landers and rovers |
| | Production of $O_2$ only makes >75% of ascent propellant mass | Water (in the form of hydrated minerals) identified globally near the surface by orbiters |
| | Significant research and testing performed on several methods of collection and processing | Lunar regolith excavation and thermal processing techniques are applicable to Mars |
| | | Low concentrations of $H_2O$ in surface hydrated minerals (3% to 8%) still provide tremendous mass benefits |
| **Weaknesses** | Production of $CH_4$ or $H_2O$ requires delivery of $H_2$ from Earth, which is volume inefficient | Risk associated with the complexity of required surface infrastructure needs must be evaluated. Significant autonomous operations required |
| | Mars optimized ISRU processing does not currently use baseline life support technologies | Local/site dependency on $H_2O$ resource concentration and form |
| | | Concerns from planetary protection and search for life with $H_2O$ excavation, especially at higher concentrations |
| | | No previous work on Mars regolith excavation or soil processing at scale or number of operations required for ISRU |

Based on the ISRU process and power system evaluations and the strengths and weaknesses that are associated with ISRU resource options on Mars, the following decisions were made with respect to ISRU integration into the DRA 5.0 mission:

1. For the ascent propellant, all options requiring the creation of $CH_4$ (fuel) from the atmosphere were dropped because of the need to bring seed $H_2$ from Earth and the corresponding large increase in tank volume on the lander. Carrying a large tank for $H_2$ would increase the size of the lander and make EDL and packaging more difficult. This would ripple back all the way to LEO, resulting in an increased number of launches.

2. For the ascent propellant, the solar power option was dropped because it was deemed to be inefficient given the power requirements of ISRU. The complexity that is associated with deploying the necessary solar arrays was also viewed as prohibitive. Additionally, an 8-hour duty cycle followed by downtime is a detriment to ISRU system operability (vs. continuous power by a nuclear system).

3. For the ECLSS/consumables, all options requiring soil processing were dropped because the complexity of surface operations are not yet well understood. While great potential exists in extracting $H_2O$ from

hydrated minerals, the mass and operations that are associated with the systems that are needed for soil processing and transfer of $H_2O$ may grow larger than current estimates and add an unacceptable amount of risk.

4. For the ECLSS/consumables, the branches that call for production of just $O_2$ or just $H_2O$ were dropped in favor of a more mass-efficient strategy that allows for the production of both using a combined system.

5. For the ECLSS/consumables, the solar power option was once again dropped for the reasons outlined in item 2 above.

In light of the decisions that are itemized above, the ISRU trade tree was reevaluated; branch selections for the two main mission needs are highlighted in figure 3-49. These branches provide the lowest power/volume and lowest risk/least complex ISRU implementation options.

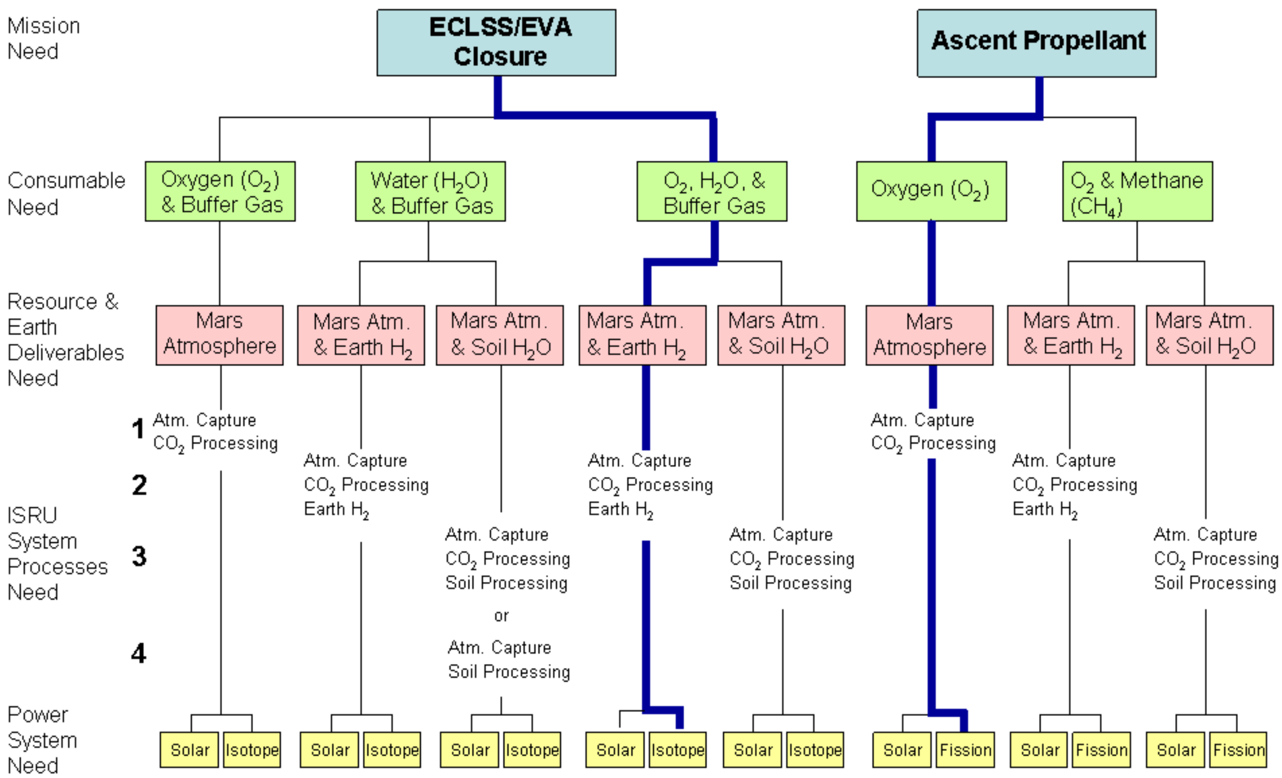


**Figure 3-49.** Selected branches of the in-situ resource utilization trade tree.

The combined atmosphere collection and processing system that was deemed best for Mars DRA 5.0 was the RCAP with SOCE system and a small amount of Earth-supplied $H_2$ for production of buffer gases, $H_2O$, and $O_2$ for life support, EVA, and ascent propulsion. Mars ascent $CH_4$ fuel was also brought from Earth. It was also concluded, that Mars soil/$H_2O$ characteristics and soil excavation and soil/$H_2O$ processing should be further evaluated and developed, with the goal of selecting RCAP with Sabatier and WE processing, and Mars soil/$H_2O$ extraction and incorporating these into future human Mars missions because of the higher mission flexibility and commonality with lunar ISRU and life support system development.

### 3.6.8 Parametric sizing of Mars ascent vehicle

Using the general results of the trade studies that are outlined above, a MAV was parametrically sized to compare the stage masses with and without the inclusion of ISRU for ascent propellant production. Through some rough analysis it was determined that a two-stage ascent vehicle performed better than a single-stage ascent vehicle. Further optimization of staging (e.g., optimized use of common tanks and engines) is required as part of forward work. In the case of a pre-fueled MAV, a mini-habitat that could sustain the crew for up to 30 days upon landing was included. For the ISRU case, such a feature was not needed in the MAV since the crew would land and acclimatize in the SHAB. Table 3-25 provides a mass summary by stage and subsystem of a MAV that creates $O_2$ from the Mars atmosphere while bringing $CH_4$ from Earth. The vehicle summary is shown in tables 3-26 and 3-27.

**Table 3-25.** Vehicle Mass Breakdown for In-situ Resource Utilization

| **Descent Module – Option 1** | **Qty** | **% of Vehicle Dry Mass** | **Mass (kg)** | **Volume ($m^3$)** | **Avg Power (W)** | **Peak Power (W)** | **Source** |
|---|---|---|---|---|---|---|---|
| 1.0 Structure | | 30 | 2,657 | 0 | 0 | 0 | |
| 2.0 Protection | | 0 | 0 | 0 | 0 | 0 | |
| 3.0 Propulsion | | 29 | 2,624 | 41 | 10 | 0 | |
| 4.0 Power | | 4 | 324 | 1 | 65,448 | 0 | |
| 5.0 Control | | 1 | 101 | 0 | 0 | 0 | |
| 6.0 Avionics | | 0 | 30 | 0 | 0 | 0 | |
| 7.0 Environment | | 5 | 415 | 1 | 283 | 0 | |
| 8.0 Other | | 9 | 767 | 0 | 0 | 0 | |
| 9.0 Growth | | 23 | 2,075 | 13 | 0 | 0 | |
| 10.0 Non-cargo | | | 274 | 0 | 0 | 0 | |
| 11.0 Cargo | | | 0 | 0 | 0 | 0 | |
| 12.0 Non-propellant | | | 3 | 0 | 0 | 0 | |
| 13.0 Propellant | | | 12,027 | 0 | 0 | 0 | |
| | | | | | | | |
| Dry Mass | | 100 | 8,993 | | | | |
| Inert Mass | | | 9,267 | | | | |
| Total Vehicle | | | 21,297 | | | | |

| **Ascent Stage 1** | **Qty** | **% of Vehicle Dry Mass** | **Mass (kg)** | **Volume ($m^3$)** | **Avg Power (W)** | **Peak Power (W)** | **Source** |
|---|---|---|---|---|---|---|---|
| 1.0 Structure | | 30 | 2,139 | 0 | 0 | 0 | |
| 2.0 Protection | | 0 | 0 | 0 | 0 | 0 | |
| 3.0 Propulsion | | 39 | 2,775 | 1,981 | 3,295 | 0 | |
| 4.0 Power | | 6 | 430 | 0 | 0 | 0 | |
| 5.0 Control | | 1 | 101 | 0 | 0 | 0 | |
| 6.0 Avionics | | 1 | 68 | 0 | 0 | 0 | |
| 7.0 Environment | | 0 | 0 | 0 | 283 | 0 | |
| 8.0 Other | | 0 | 27 | 0 | 0 | 0 | |
| 9.0 Growth | | 23 | 1,662 | 594 | 1,431 | 0 | |
| 10.0 Non-cargo | | | 454 | 0 | 0 | 0 | |
| 11.0 Cargo | | | 0 | 0 | 0 | 0 | |
| 12.0 Non-propellant | | | 0 | 0 | 0 | 0 | |
| 13.0 Propellant | | | 20,245 | 0 | 0 | 0 | |
| | | | | | | | |
| Dry Mass | | 100 | 7,202 | | | | |
| Inert Mass | | | 7,656 | | | | |
| Total Vehicle | | | 27,902 | | | | |

Subtract 15,746 kg of $O_2$ (offload) from the above stage summary when bringing $CH_4$ from Earth.

| **Ascent Stage 2** | **Qty** | **% of Vehicle Dry Mass** | **Mass (kg)** | **Volume ($m^3$)** | **Avg Power (W)** | **Peak Power (W)** | **Source** |
|---|---|---|---|---|---|---|---|
| 1.0 Structure | | 15 | 771 | 0 | 0 | 0 | |
| 2.0 Protection | | 0 | 0 | 0 | 0 | 0 | |
| 3.0 Propulsion | | 24 | 1,214 | 24 | 1 | 1,215 | |
| 4.0 Power | | 15 | 793 | 1 | 28 | 872 | |
| 5.0 Control | | 0 | 0 | 0 | 0 | 0 | |
| 6.0 Avionics | | 6 | 334 | 0 | 1,276 | 1,943 | |
| 7.0 Environment | | 13 | 689 | 2 | 861 | 1,125 | |
| 8.0 Other | | 3 | 169 | 1 | 25 | 50 | |
| 9.0 Growth | | 23 | 1,191 | 9 | 0 | 0 | |
| 10.0 Non-cargo | | | 420 | 0 | 118 | 1,149 | |
| 11.0 Cargo | | | 0 | 0 | 0 | 0 | |
| 12.0 Non-propellant | | | 0 | 0 | 0 | 0 | |
| 13.0 Propellant | | | 12,253 | 0 | 0 | 0 | |
| | | | | | | | |
| Dry Mass | | 100 | 5,160 | | | | |
| Inert Mass | | | 5,580 | | | | |
| Total Vehicle | | | 18,540 | | | | |

Subtract 9,210 kg of $O_2$ (offload) from the above stage summary when bringing $CH_4$ from Earth.

**Table 3-26.** In-situ Resource Utilization Vehicle Mass Summary for Oxygen Production

| | | |
|---|---|---|
| Ascent Stage 2 | 9,330 kg | Includes $CH_4$ brought from Earth |
| Ascent Stage 1 | 12,156 kg | Includes $CH_4$ brought from Earth |
| ISRU Equipment and Power Plant | 11,280 kg | |
| Descent Stage | 21,297 kg | |
| **TOTAL** | **54,062 kg** | |

**Table 3-27.** In-situ Resource Utilization Vehicle Mass Summary for No Oxygen Production

| | | |
|---|---|---|
| Ascent Stage 2 | 18,540 kg | Includes $CH_4$ and $O_2$ brought from Earth |
| Ascent Stage 1 | 27,902 kg | Includes $CH_4$ and $O_2$ brought from Earth |
| 30-day mini-habitat | 5,687 kg | |
| Descent Stage | 27,300 kg | |
| **TOTAL** | **79,428 kg** | |

As can be seen in the above tables, the mass savings that were gained by using ISRU are substantial. Tracing those savings all the way back to their IMLEO value will result in fewer launches during a given mission.

### 3.6.9 In-situ resource utilization costing

The cost estimates for the Mars ISRU system were generated using the NASA/Air Force Cost Model (NAFCOM) tool and data set. The ISRU systems that were studied were broken down into the subsystem level to determine cost. For each subsystem, an appropriate analogy to historical human spacecraft subsystems was chosen (see table 3-28). The NAFCOM multivariant cost-estimating relationships for the appropriate analogy were used to determine the design, development, test, and evaluation (DDT&E) and unit costs for the subsystems; a systems integration cost was added to determine the overall element costs. The results of this are shown in table 3-29.

**Table3-28.** In-situ Resource Utilization Costing Analogies

| ECLS* Subsystem | Cost Analog |
|---|---|
| Atmosphere Acquisition | ECLS |
| $O_2$ Generation | ECLS |
| Cryocoolers | Active Thermal |
| Storage Tanks | Structure – Tank |
| Regolith Feed/Removal | Structures & Mechanisms |
| Water Extraction Reactor | Electrical Power |
| Excavators | Crewless Rovers** |

*ECLS = environmental control and life support
**Excavator costs estimated outside of NAFCOM

**Table 3-29.** In-situ Resource Utilization Surface Systems Cost (relative to Case 1b)

| Architecture | DDT&E | To First Mission | Through Third Mission |
|---|---|---|---|
| *Case 1a*: ECLSS $O_2$ and $N_2$/Ar with Earth $H_2$ – Solar Powered | 86% | 88% | 90% |
| *Case 1b*: ECLSS $O_2$ and $N_2$/Ar with Earth $H_2$ – Nuclear Powered | 100% | 100% | 100% |
| *Case 2a*: ECLSS $O_2$ and $N_2$/Ar with Mars $H_2O$ – Solar Powered | 140% | 146% | 152% |
| *Case 2b*: ECLSS $O_2$ and $N_2$/Ar with Mars $H_2O$ – Nuclear Powered | 131% | 132% | 134% |
| *Case 3*: ECLSS and Propellant $O_2$ and $N_2$/Ar with Earth $H_2$ and $CH_4$ – Nuclear Powered | 103% | 104% | 106% |
| *Case 4*: ECLSS and Propellant $O_2$ and $N_2$/Ar with Mars $H_2O$ – Nuclear Powered | 191% | 202% | 213% |

The increase in cost from Case 1b to Case 3 represents the additional cost of adding enough capability to produce oxygen for both ECLSS and propellant requirements over the cost of ECLSS requirements alone. While there is expected to be some cost increase by developing ISRU systems, this is more than offset by the cost savings of reducing the size of the stages. The true cost difference between the architectures comes in the cost saving of reducing the total number of Ares-V launches for each mission.

The cost of extracting resources from Mars soil is higher than extracting from the atmosphere alone due to the additional equipment required. Case 2b would have lower costs over Case 1b if the number of Ares-V launches was reduced by two or more through the third mission. Case 4 would have a lower cost compared to Case 1b if the number of Ares V launches was reduced by four or more through the third mission.

### 3.6.10 In-situ resource utilization risk considerations

The ISRU reliability analysis was conducted by applying heritage, analog component failure data to the ISRU Master Equipment List (MEL) and component sparing, precursor, and reliability maturation assumptions. The failure data set that was used was a surrogate set that was developed from a combination of component failure data from the ISS modular auxiliary data system (MADS)) database and "order of magnitude" subject matter expert estimates (see figure 3-50). This is a preliminary analysis of ISRU reliability, which is intended to help stakeholders discriminate among design alternatives on the basis of reliability; it should not be taken as a forecast of the actual fielded reliability of the ISRU system.

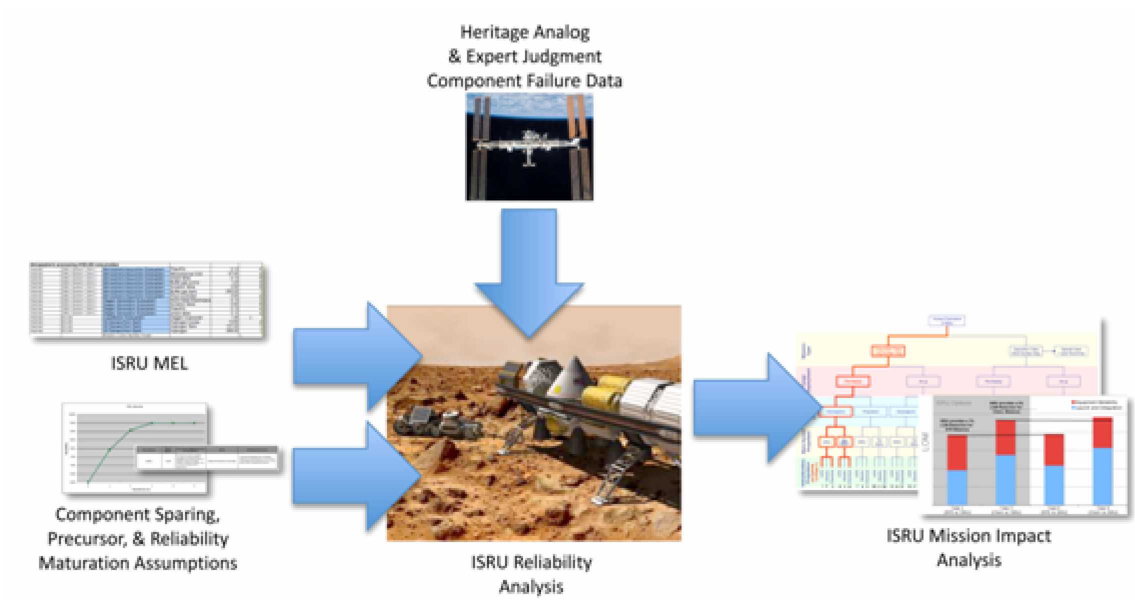


**Figure 3-50.** In-situ resource utilization reliability analysis strategy.

The analysis was conducted assuming a system operational exposure time of 540 days on the surface of Mars, which implies that the ISRU would require an additional 107 kg of optimized sparing hardware to bring it to an acceptable reliability level at maturity. The mission assumes an order of magnitude increase in component reliabilities based on system maturation and component reliability improvement programs included as part of the precursor efforts. The final reliability and component maturation level will be dependent upon possible synergies with ground testing, robotic precursor missions, and potential lunar experience.

Although ISRU provides for a mass reduction for Mars EDL, the required propellant tanks do not allow for a change in the Mars entry system moldline design. However, the reduction in mass will be beneficial in reducing the energy that is required to be dissipated on entry. Due to the uncertainty in the EDL system, no reliability credit for ISRU was taken for this analysis. Therefore, ISRU will only affect LOM in the areas of **equipment reliability** and **launch and integration**. The ISRU system operational time increases the equipment reliability contributing to the loss of mission (LOM) risk for Cases 1 and 3. However, the launch and integration LOM risk reduction provides these missions with a slight edge for LOM over the equivalent non-ISRU missions (Cases 4 and 6). Trade-tree Case 1 with NTR propulsion and ISRU has a 1% LOM advantage over the similar non-ISRU option (Case 4). Trade-tree Case 3 with chemical propulsion and ISRU has a 3% LOM advantage over the similar non-ISRU option (Case 6). The launch reduction ISRU provides a significant reliability benefit. However, from a mission-reliability perspective, ISRU does not appear to be a clear-cut discriminator based on this analysis. Results are illustrated in figure 3-51.

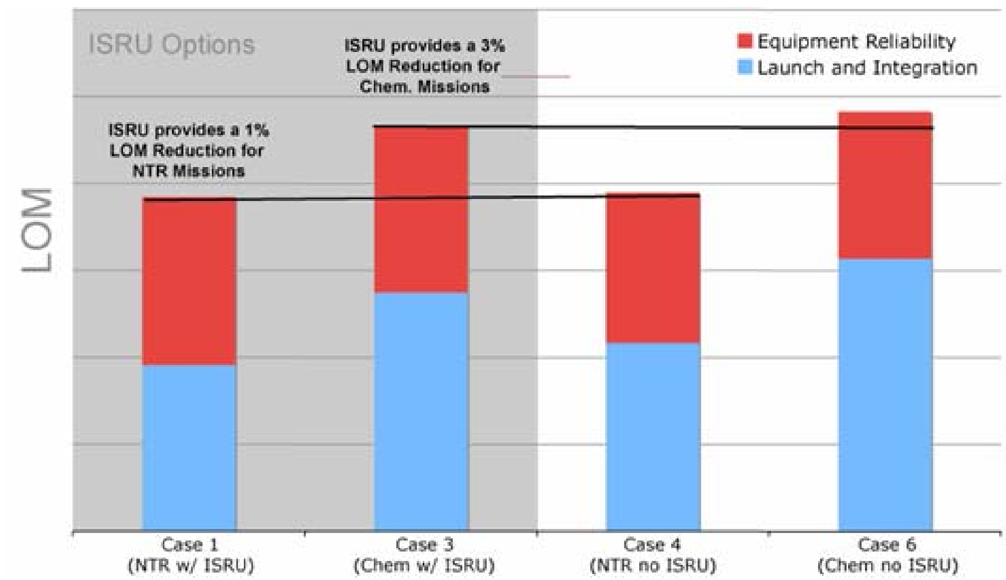


**Figure 3-51.** In-situ resource utilization reliability estimates.

From a loss of crew (LOC) perspective, it is important to point out that ISRU could provide additional $O_2$ for backup life support during failures or leaks that are affecting the primary ECLSS consumables. ISRU could also prove beneficial to LOC by enabling the use of a pressure-fed ascent propulsion system because it would allow for the compensation of a lower ISP by producing additional propellant on the surface.

### 3.6.11 In-situ resource utilization conclusions and future work

Figure 3-52 lists the main FOMs against which the use of ISRU was measured. The first three – DAV Landed Mass, IMLEO, and Number of Ares V launches – are closely related and provide a good first-order indication of the relative risk and cost of the two approaches. Fewer launches have the greatest effect on bringing down both risk and cost. For both ascent propellant and ECLSS consumables, the use of ISRU necessitates greater power, which comes at the cost of mass. This is one apparent advantage of not using ISRU. However, significant surface power capability will most likely be required to support the surface activities and infrastructure regardless. Likewise, using the more traditional approach of landing a fully fueled MAV will make surface operations (e.g., transfer of consumables) simpler and, therefore, more mass efficient. Another important FOM is a catch-all that is termed “Mission Flexibility.” ISRU gives the ability to produce fuel for roving, EVAs, and other activities that would otherwise be limited by a fixed consumables budget. This additionally flexibility provides an advantage over the non-ISRU case.

The final two FOMs were risk and cost. The ISRU and non-ISRU cases were judged to be similar in terms of loss of mission probability for different reasons. Using ISRU increases the risk of component failure on some of the specialized systems but decreases the risk of launch and assembly (one of the greatest risk contributors) by requiring fewer launches (lower IMLEO). In terms of costs, analysts found very little difference between the two for almost the same reason, as in the case of the risk analysis. The cost of the more expensive ISRU systems is offset by the reduction in number of launches over a program.

**Human Exploration Of Mars**

| ISRU Propellants (Oxidizer) | Figure of Merit | No ISRU Propellants |
|---|---|---|
| Lower Mass: 54 mT | DAV Landed Mass | Higher Mass: ~79 mT |
| ~ 820 mT (Nuclear); ~1250 mT (Chem.) | IMLEO Mass for 1 mission | ~910 mT (Nuclear); 1420 mT (Chem.) |
| 9 (Nuclear); 12 (Chemical) | Number of Ares V Launches Per Mission (125 MT Launch System) | 10 (Nuclear); 14 (Chemical) |
| More | Power Required | Less |
| More complex operations and infrastructure; surface rendezvous required for abort to orbit | Complexity of Surface Operations | Less Complex operations; no surface rendezvous required for abort to orbit |
| Greater Flexibility do to the ability to produce more fuel for roving/hopping/EVA | Mission Flexibility (contingency re-planning) | Less Flexibility |
| Advantage | PLOM | Slightly higher risk attributed to more launches and higher IMLEO |
| No discriminator | Cost Through First Mission | No discriminator |
| Advantage | Cost Through Third Mission | Disadvantage |

**Figure 3-52.** In-situ resource utilization figure of merit summary.

With respect to further evaluation and refinement of the ISRU mission concept, there are several aspects that Future Work should focus on to further increase mission mass reduction, increase mission flexibility, and lower mission and LOC risk. Areas of further work interest are:

1. Continue to work on defining the global distribution and concentration of $H_2O$ in the form of hydrated minerals and concepts for excavation and soil/ $H_2O$ processing on Mars. It is important to continue preliminary identification of surface/near-surface $H_2$ (interpreted as hydrated minerals) with a global distribution and locally substantial concentrations. Tremendous ISRU potential exists at even low concentrations of $H_2O$ in hydrated soils; however, higher concentrations would reduce the amount of time that is required to process or reduce the size and power of the hardware that is required to obtain this resource. Also, continue evaluation of lunar regolith excavation and thermal processing techniques for applicability to excavate and extract $H_2O$ on Mars, and continue to define hardware concepts. The complexity (and mass impact) of required supporting infrastructure is currently very preliminary and could preclude this avenue of ISRU if it is found to be too difficult or complex.

2. Evaluate pump-fed vs. pressure-fed $LO_2/CH_4$ propulsion systems for Mars descent and ascent vehicles. Pressure-fed propulsion systems are inherently less risky than the currently baselined pump-fed propulsion systems, albeit at the expense of lower performance and increased overall ascent vehicle mass. Because ISRU can provide significant mass savings, it might be possible to use some of this mass savings to incorporate a lower-performing, but higher-reliability, pressure-fed propulsion system into the Mars landers. ISRU may be enabling for Mars pressure-fed propulsion vehicles.

3. Evaluate ISRU hardware limitations and risks as a result of Mars dust. Since acquisition and collection of the Mars atmosphere for processing may contain dust, filter clogging, catalyst degradation, and process performance reductions may occur due to physical and/or chemical aspects of Mars dust infiltration into the system.

4. Perform more in-depth risk assessments associated with the surface mobility that is required for ISRU strategies.

## 3.7 Mars Surface Power

The assessment of a power supply for the crewed lunar base is intimately tied to the Mars mission architecture and concept of operations. For human-scale activities, two main options are available to provide outpost power: solar and nuclear fission. An additional option, the use of large-scale (1–5 kWe) radioisotope power systems (RPSs) can be considered for backup power needs, as well as for surface transportation applications.

### 3.7.1 Power requirements

A major consideration in developing the power trade is the power requirement that is imposed by the mission. For the mission architectures that were considered, two major phases are defined with separate power requirements.

#### *3.7.1.1 In-situ resource utilization phase*

The first phase, which commences shortly after landing of the cargo vehicle and extends to athe rrival of the crew, is the ISRU phase. During this period, power must be provided to process in-situ resources either for crew and EVA consumables only ($O_2$ and $N_2$/Ar) or for propellant $O_2$ production in addition to consumables. Estimates for power requirements for these two scenarios vary with the assumed power source. For the case of nuclear fission power, it is assumed that the ISRU plant would be operated continuously for at least 300 days to produce the necessary resources. In the case of solar power, the total energy would be the same, but the operation of the ISRU plant would be limited to 8 hours per day at three times the power level of the nuclear case. This daytime-only operation avoids the need for the large quantities of fuel cell reactants that would be necessary to provide round-the-clock production; but, in turn, substantially larger surface arrays must be packaged, outfitted on the cargo lander, and then deployed. Daytime-only production may also result in inefficiencies that will need to be evaluated further to determine whether additional margin should be provided to the solar power requirement.

Current estimates place the power requirement for the consumables-only ISRU case at 2 kWe continuous, or approximately 6 kWe for 8 hours/day operations. When $O_2$ propellant production is added, these power requirements rise to 26 kWe and approximately 96 kWe, respectively.

#### *3.7.1.2 Crewed phase*

The second major phase of the mission is the crewed phase, which commences with arrival of the crew at the outpost site. Power requirements for this phase vary among the three scenarios that were considered for the mission architecture, depending on extent of mobility provided and the presence of a dedicated habitat.

**Scenario 1: MOBILE HOME**

In the "Mobile Home" scenario, the crew would live in two large, long range rovers. These rovers would be required to provide all of the power that is necessary to support the crew members during their stay, as well as providing the considerable energy that is required for roving expeditions lasting up to 30 days, during which time the rovers would traverse as much as 200 km. No central habitat would be included in this scenario, although any central power supply that is needed to support ISRU prior to crew arrival would be available to power any systems at the landing site, as well as to provide recharge of batteries or resupply of fuel cell reactants between rover excursions. Current estimates for the power that is required to support the crew on each large rover are 5 kWe for daytime operations, dropping to 3.5 kWe at night.

Power that is required for mobility varies with the mass of the rover and the speed at which it is required to travel. Given the science-driven desire to perform excursions of up to 200 km in 30 days, an average rover speed of about 3 km/hr is required, assuming 5 hours of driving every other day. This driving pattern would allow average traverses of 15 km in a day, followed by 1 day of stationary science activities; it therefore seems a reasonable model for planning purposes. To achieve the required 3-km/hr speed a 15,000-kg rover would need about 47 kWe for mobility alone. This imposes a very high requirement for power generation or energy storage. For comparison, the same rover traveling at 0.5 km/hr would require only 8 kWe for mobility. However, the adoption of this lower speed would limit the distance that is traveled by the rover in 30 days to approximately 38 km, or, if the 200-km distance was still required, the trip duration would stretch to >160 days.

**Scenario 2:** **COMMUTER**
The "Commuter" scenario includes a central habitat in addition to two smaller pressurized rovers. The central habitat would provide services to the full crew in between rover excursions, maintaining a minimum crew of two when both rovers are in the field. Estimated power requirement for the habitat is 12 kWe during both day and night.

The two pressurized rovers in this scenario are estimated to require 3.4 kWe daytime power for the crew, dropping to 2.4 kWe at night. Mobility power is variable as for the larger rover, but the lower mass of these rovers (~7,500 kg) brings this requirement down to approximately 25 kWe for a speed of 3 km/hr.

**Scenario 3:** **TELECOMMUTER**
No pressurized rovers are included in the "Telecommuter" scenario. The habitat is included, however, and power requirements are estimated to be the same as for the Commuter scenario that was discussed above. This scenario also includes two long-range robotic rovers. These rovers are expected to use an isotope system due to anticipated higher power levels than those of the MERs. The unpressurized crew transportation rovers are likely be similar to the Apollo LRV [lunar rover vehicle]. While not studied in detail, it is assumed that power for the rover would primarily be an energy storage device.

### 3.7.2 Power profiles
The power systems that were used for these missions must accommodate all mission phases. A graphical representation of the power profile is illustrated in figure 3-53. As shown in the figure, the ISRU phase, which occurs prior to crew arrival, requires substantial power to be available from shortly after the cargo vehicle lands to ensure sufficient resource buildup. The implication of this is that the main power system for the outpost must be delivered with the cargo and ISRU equipment, and must be autonomously deployed and activated within a reasonable amount of time after landing (assumed to be ~30 days).

In addition to the base power loads for ISRU and habitat power, the figure shows graphically the additional power that is required to support outpost systems during both the ISRU and the crewed phases of the surface mission. These loads include about 1.5 kWe each for the logistics module, ECLSS/EVA cache, and ascent vehicle maintenance power. An additional 1.5 kWe are also reserved for charging unpressurized rovers and other miscellaneous power loads.

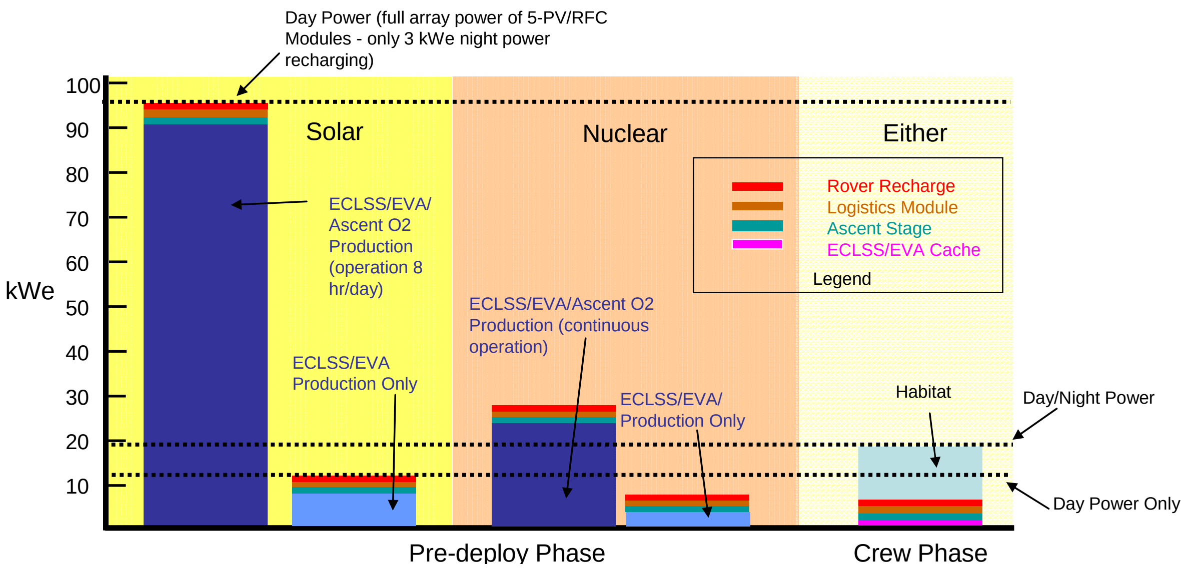


**Figure 3-53.** Power requirements profile.

The propellant ISRU requirement for power dominates other loads and becomes the main factor in sizing the base power system. For the nuclear case, the ISRU phase power totals to about the 26.5 kWe that are required on a

24-hour/day basis. The solar case requires a total daytime power capability of about 94 kWe for 8 hours/day to support ISRU (including margin to allow for at least one 50-day dust storm), in addition to providing 4.5 kWe continuous power for auxiliary loads. These total power requirements drop following the completion of ISRU activities, with total continuous power loads during the crew phase requiring about 17–20 kWe.

### 3.7.3 Power system concepts

A number of power system options can be applied to the Mars exploration architectures for main base power as well as for backup and mobility power. For purposes of evaluating the major power system trades, conceptual designs were adopted for the main contenders. These include: solar, nuclear fission, and large-scale radioisotope systems as addressed in the following sections.

#### *3.7.3.1 Solar power system concept*

Solar PV power systems have recently shown themselves on the MER missions to be capable of long-duration operation on the martian surface. Although solar arrays face a number of challenges on Mars, the relative simplicity and technical maturity of PV systems makes them a candidate for application even to large-scale human missions.

For the present study, it was decided that an optimum approach for solar power would be to develop a modular PV system that would be capable of providing 5 kWe continuous power. An optimal number of these units could be deployed to provide the power that is necessary to support base operations. Additional units could be provided for redundancy.

An artist's illustration of a power system consisting of five of these 5 kWe modules is shown in figure 3-54. Each module would consist of one or more solar array wings providing a total of 290 $m^2$ of solar array area. Solar arrays would be populated with 29% efficiency triple junction cells. The arrays would be fixed at an inclination angle that would allow evening out-of-power output over the course of the day, and would facilitate automated dust removal systems. The solar arrays feed power to a central box that contains power management and distribution equipment, as well as five regenerative fuel cells that would provide 5 kWe of power for nighttime operations.

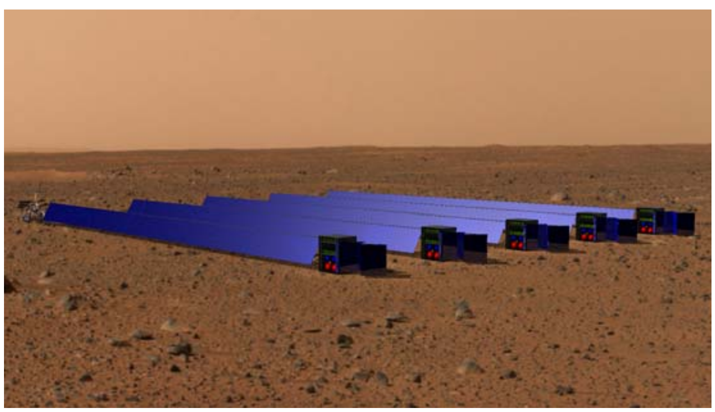

**Figure3-54.** Solar power concept.

To meet the base power needs for the crewed phase of the Commuter or Telecommuter scenarios, five of these modules would be employed. The nominal power requirements would be met by four of these units, with one unit provided for redundancy. These five modules would also be sufficient for consumables-only or propellant production ISRU, operating at approximately 100 kWe for 8 hours/day while supplying approximately 3 kWe nighttime power through the RFCs, even if a dust storm should cause ISRU operations to be suspended for up to 50 days.

The mass of a single 5-kWe module is estimated to be 1,980 kg (including 20% contingency). Thus, the total power system mass for the non-propellant ISRU case would be about 10,000 kg. The addition of 450 $m^2$ of solar array to accommodate the propellant ISRU production would increase this overall power system mass to about 12,500 kg (including 20% contingency).

### *3.7.3.2 Fission surface power system*

The FSPS design is taken directly from the recent work that was performed to develop a low-cost, low-temperature system for the lunar architecture. One of the key features of this power system design is that it is adaptable to use either on the lunar or martian surface. A sketch of the 40-kWe design that was developed for the moon is shown in figure 3-55.

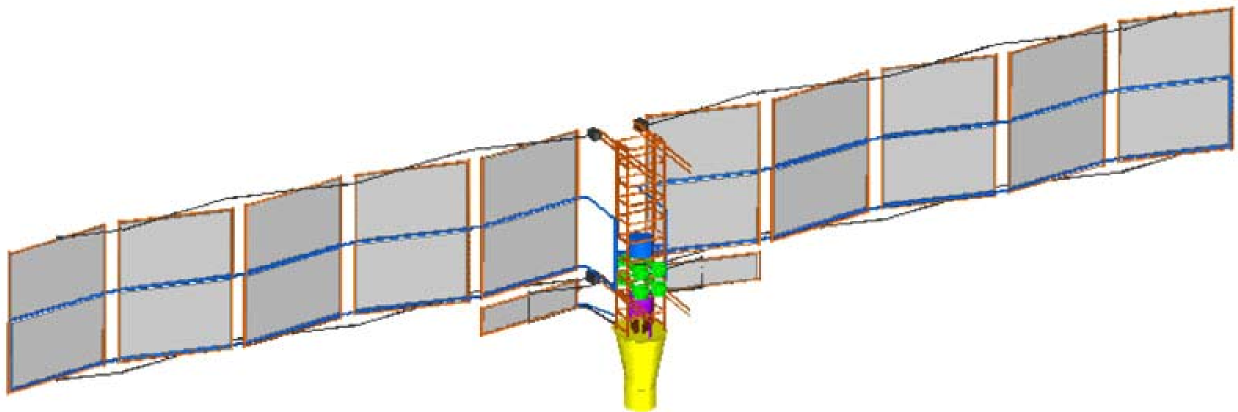

**Figure 3-55.** Conceptual fission surface power system configuration.

The reactor is located at the base of the power system and, for the Mars case, is surrounded by a substantial radiation shield – one that is preferably thickened in the direction of the base. This design allows the reactor to be sited at 1 km separation from the base, achieving a dose rate during reactor operation of <5 rem/year at the base. The shaped shield provides a dose rate of <50 rem/year in all other directions. This implementation of the FSPS would require the reactor to be landed with its own mobility system, which would autonomously drive the FSPS to a distance of 1 km from the landing site, deploying a power cable as it goes. Once the implementation site is reached, the FSPS would deploy its radiators, and startup of the reactor would be performed. From the end of startup operations, full power would be available to the base essentially independent of time of day or atmospheric conditions.

Mass for the FSPS is variable with power output, which is primarily based on the size of radiator that is needed to reject waste heat. The estimated mass for a 20-kWe reactor that might be used for the non-propellant ISRU cases is 6,800 kg (including 20% contingency). The mass for a 30-kWe reactor that could accommodate propellant ISRU is estimated at about 7,800 kg.

### *3.7.3.3 Large-scale radioisotope power system*

An additional power system concept that can be considered for applications such as backup power and mobility is the large-scale RPS. Large-scale RPS designs that are based on Stirling engine technology have been under development in power levels up to 10 kWe. For this study, a 5-kWe RPS has been considered (as shown in figure 3-56). This system consists of a heat source made up of 54 general-purpose heat source (GPHS) modules containing a total of 32.4 kg of $^{238}$Pu. For comparison, this is the same amount of plutonium that is currently being used to power the Cassini spacecraft at Saturn. While the Cassini RTGs are able to provide about 1 kWe to the spacecraft, the much greater efficiency of the Stirling generator would enable such a large-scale RPS to generate approximately 5 kWe from the same amount of fuel. The RPS would provide a continuous power source from the time that it is fueled, with a power output estimated to fall off by about 0.8% per year as a result of natural decay of the $^{238}$Pu fuel.

The current design for the 5 kWe RPS estimates its mass to be about 450 kg (including 20% contingency). A smaller 2.5-kWe system is estimated to be about 230 kg.

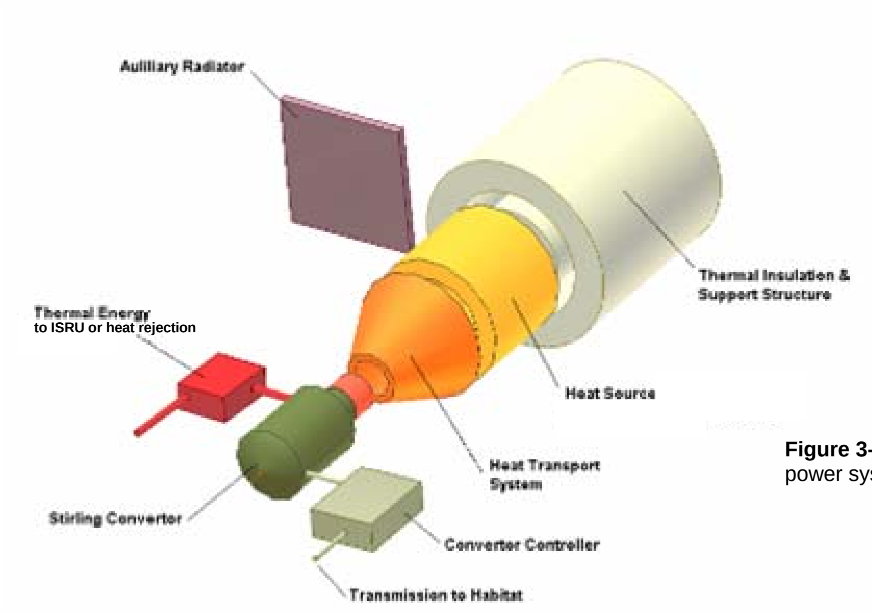


**Figure 3-56.** Conceptual radioisotope power system configuration.

### 3.7.4 Implementation considerations

A number of issues affect the power system trade. The following sections highlight some of the key considerations that must be taken into account.

#### *3.7.4.1 Dust deposition*

Accumulation of dust on both horizontal and vertical surfaces has been a salient feature of Mars surface missions. The MERs and the earlier Mars Pathfinder mission witnessed power output drops of approximately 0. 2% per day resulting from dust. The surprising longevity of the MERs has been a result of “clearing events” seen by both rovers that temporarily mitigated the output losses; however, dust buildup has been seen to resume following these cleaning events.

The design of a solar power system that is critical to mission success will not be able to rely on these cleaning events, which are incompletely understood. It will be necessary for a solar power system to incorporate some form of autonomous dust-mitigation technology prior to crew arrival. Technologies are in development (e.g., piezoelectric vibration) that show great promise in this area, but it will be necessary to further develop these technologies before they can be relied on. For the purposes of this study, a dust obscuration of 10% has been baselined that assumes that some form of autonomous dust mitigation system is available at the time of the mission.

Dust deposition will have a minimal effect on the FSPS. Dust adhesion to the radiator surfaces can potentially result in a slight decrease in emissivity, but this should not significantly affect operation.

#### *3.7.4.2 Dust storms*

Perhaps the greatest threat to the solar-powered system is the incidence of large-scale dust storms on Mars. Regional and global dust storms can dramatically reduce the amount of sunlight reaching the surface, thereby reducing solar power to a fraction of its nominal levels. Recent experience on the MERs has shown a decrease in power output during the worst days of a storm (down to 15% of pre-storm capability). The solar power system must be designed to provide at least minimal survival power during dust storms, which may last for 1 to 2 months. This can be provided by including an extra solar array area, additional fuel cell capacity, or a combination of the two. Analyses for the current study included sizing for crew survival during a dust storm lasting up to 50 days. This would require an additional solar array of approximately 4,300 m$^2$ to be deployed prior to the dust storm. The concept that was considered for the study would entail having the crew deploy a thin-film array blanket to provide extra power for the duration of the storm. The mass of the extra blanket is estimated at about 7,800 kg.

Again, the dust storm conditions will have little effect on the FSPS. The radiator will see the daytime temperatures drop while night temperatures increase, resulting in no significant change in overall performance.

#### *3.7.4.3 Deployment*

Autonomous deployment of very large structures in space is inherently complicated, and historically has posed, in crewed missions, serious problems that have been had to be remedied by crew involvement. Solar arrays provide the most memorable examples of this issue, with Skylab and the ISS both suffering deployment failures, although the ISS example was fully recoverable with crew intervention. Autonomous deployment of the very large solar array wings that will be required for the surface power implementation will be a critical task, and a system must be engineered and tested to assure reliable operation.

Likewise, the FSPS must deal with deployment of its large radiators. While the area is smaller on these radiators than the solar arrays, the radiators have the additional feature of containing jointed fluid lines requiring a small level of extra complexity. The FSPS also must rely on its integral mobility system to drive it to its emplacement site at least 1 km from the landing area. It must also deploy a high-voltage power cable during this traverse.

#### *3.7.4.4 Latitude constraints*

A major distinguishing feature of the FSPS is its ability to operate at any latitude on the martian surface. The solar power system, however, will be more limited in its geographical range. Previous studies have shown that the applicability of a solar power system is best between latitudes of about 15°S and 30°N, with system efficiency falling off quickly beyond this region. This equatorial band, which is shown in figure 3-57, is overlaid on the map of 58 sites of potential interest that were identified by the HEM-SAG. Of these sites, approximately 26–28 fall within the latitudes where solar power would be a viable option.

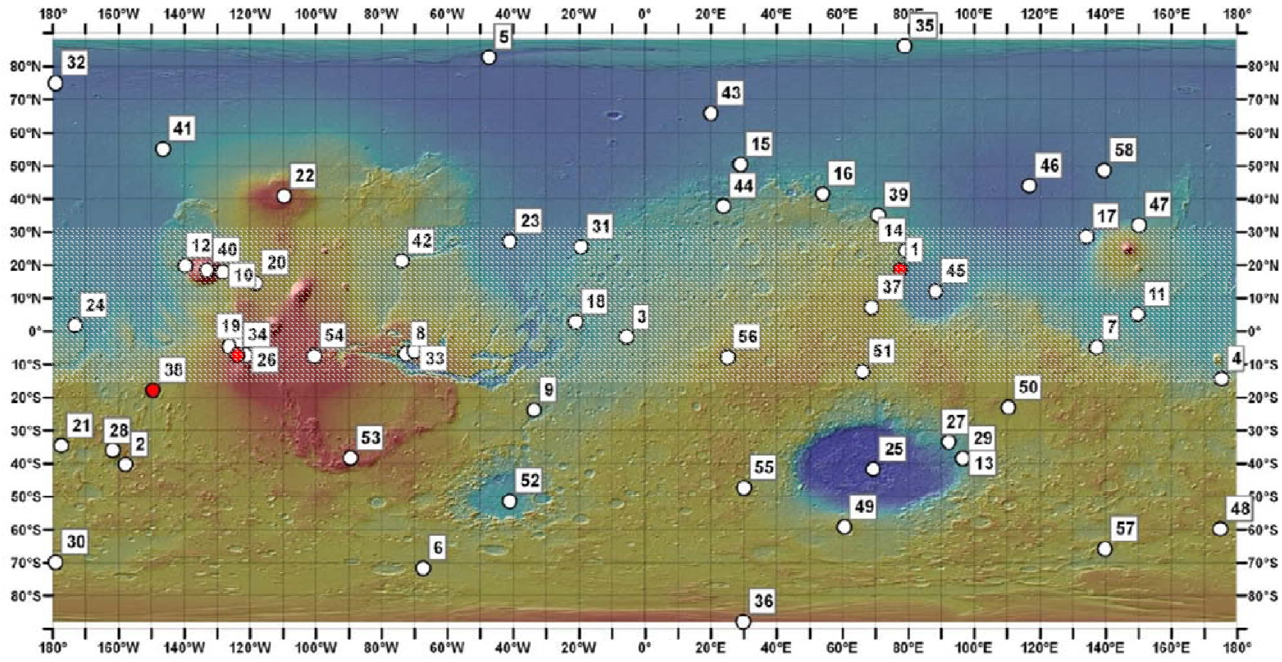


**Figure 3-57.** Latitude band of effective solar power applicability with sites of interest.

#### *3.7.4.5 Operational restrictions*

One special consideration that is peculiar to the FSPS is the operational restrictions that are posed by radiation shielding implementation. The use of a shaped shield facilitates an acceptable radiation dose rate (design guideline used is 5 rem/year) at the base site while keeping the overall shield mass manageable. However, it does result in a restriction in the area of the base that is included in the shielded region. For the LAT study design, the protected base diameter was set at 200 m for a separation distance of 1 km. Regions outside of this zone would be shielded to a dose rate of <50 rem/year (figure 3-58). The higher radiation areas would be safe for transit and limited operations without significant increase in crew dose. Only areas that are significantly closer to the FSPS would pose health risks for short-duration exposure.

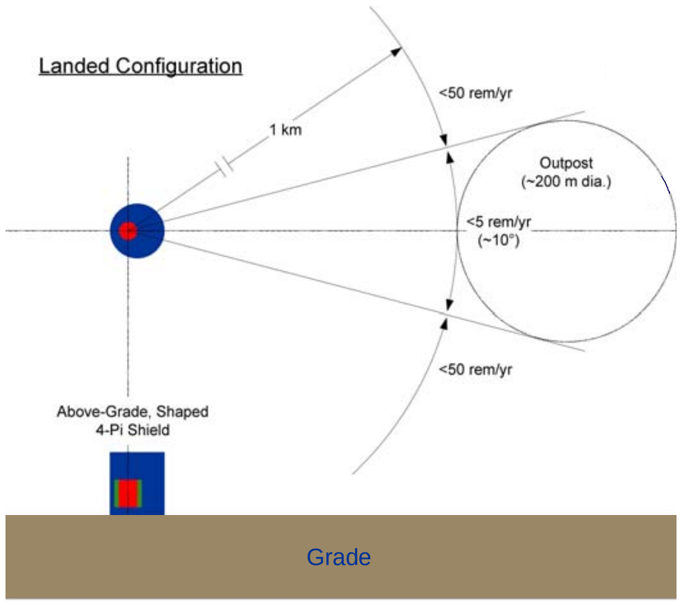


**Figure 3-58.** Fission surface power system radiation protection zone.

### *3.7.5 Surface Power Recommendations*

A summary of the FOMs that are relative to the power system trade is shown in figure 3-59. In this figure, areas in which one system is seen to have an obvious advantage over the other are shaded green.

Consideration of these FOMs and the information that is presented in the sections above has resulted in the following recommendations for Mars surface power:

1. Any power architecture that is implemented on which the crew depends should incorporate a reliable backup power system that is capable of supplying survival power. The incorporation of one or more large-scale RPSs could provide this backup capability, or it could be provided by a sufficiently robust auxiliary PV system with adequate energy storage capacity.

2. It is recommended that the FSPS be the primary power source for mission scenarios incorporating propellant ISRU. It is recognized that these missions could also be implemented using solar power, subject to latitude restrictions, however, the FSPS would result in a significantly lower mass and arguably simpler implementation, thus providing a steadier and more robust power source that would benefit the ISRU process efficiency. Additionally, it is felt that the complexity and criticality of successful deployment of the full complement of solar arrays that is needed for the ISRU phase would pose a significant challenge.

3. Reliability and cost estimates for a nuclear system are strongly dependent on the development and test programs that precede the Mars mission. Development of the nuclear system and its use in the lunar environment would greatly reduce the cost and significantly reduce the residual uncertainties in long-term operational reliability that would remain subsequent to terrestrial developmental testing.

4. The power estimates for crew operations in this study did not take into account the possibility of the nighttime or emergency power modes that could be implemented in case of a dust storm. Such a low-power mode could greatly reduce the additional solar array area that is needed to accommodate dust storm periods, thus simplifying the solar option.

5. It is noted that while the FSPS design that was considered in this trade is the product of a fairly detailed design study that was performed for the LAT, the solar power system did not benefit from such an effort. Any further consideration of solar power systems should begin with a design study that would develop a detailed implementation concept that is more fully tailored to this application.

**Base Power Supply (Commuter and Telecommuter Options)**

| Solar Power | Figure of Merit | Fission Power |
|---|---|---|
| 22.5 | Total landed mass (mt) | 7.8 (w/propellant ISRU) |
| High | Autonomous Deployment complexity | Moderate |
| Variable with dust settling and atmospheric obscuration | Power level stability | Continuous |
| High | Sensitivity to dust storms | Low |
| High | Reliability | High |
| Moderate | Ability to repair | Low |
| None | Increase in crew radiation exposure | Small (5 rem/year) |
| Mass increase with latitude | Latitude flexibility | No restrictions |
| Linear with power | Scalability | Relatively moderate increase with power in ranges of interest |
| Moderate | Development Complexity | High |
| Moderate | Similarity to lunar system | High |
| None | Surface access exclusion zone | Small areas of forbidden access, moderate areas of limited access |
| Disadvantage | Cost Through First Mission | Advantage (assuming lunar dev.) |
| Disadvantage | Cost Through Third Mission | Advantage (assuming lunar dev.) |

**Figure 3-59.** Figures of merit summary.

## *3.8 Planetary Protection*

### 3.8.1 Basis for planetary protection policy

A strong motivating factor for the exploration of the solar system is the search for extraterrestrial life. However, this search could be permanently compromised if spacecraft traveling to the more promising planetary environments carry Earth life that can contaminate the places that are explored. Additionally, samples that are returned to Earth from other places could contain living organisms that might reproduce on Earth and damage our biosphere. The practice of minimizing the probability of either type of contamination occurring is called "planetary protection."

Planetary protection entered into international law as Article IX of the 1967 Outer Space Treaty, which states in part that:

> "...parties to the Treaty shall pursue studies of outer space including the Moon and other celestial bodies, and conduct exploration of them so as to avoid their harmful contamination and also adverse changes in the

environment of the Earth resulting from the introduction of extraterrestrial matter and, where necessary, shall adopt appropriate measures for this purpose..."[80]

The COSPAR, which is a committee of the International Council for Science, maintains an international planetary protection policy that , in consultation with the United Nations Committee on the Peaceful Uses of Outer Space, serves as the consensus standard for biological contamination avoidance under the 1967 Outer Space Treaty. NASA Planetary Protection Policy is consistent with the COSPAR policy, and is documented in NASA Policy Directive NPD 8020.7. The current policy is applicable to human interplanetary missions, although specific requirements for human missions have not yet been issued. The requirements for robotic missions are given in NPR 8020.12. In general, any planetary protection restrictions depend on the nature of the mission and the target planet, with landed missions to planets of specific interest for biological studies being protected to the greatest extent. These requirements are determined "through recommendations from both internal and external advisory groups, but most notably from the Space Studies Board of the National Academy of Sciences," according to NASA policy. Specific measures may include: constraints on spacecraft operating procedures; inventory of spacecraft organic and biological contamination; reduction of such contamination; and, for sample return missions, restrictions on the handling of returned samples.

The Planetary Protection Subcommittee of the Science Committee of the NASA Advisory Council was formed to provide detailed review and advice regarding the requirements that are levied on each outgoing mission that might pose a contamination hazard, and on every sample return mission. The detailed requirements for each mission are documented in a "Planetary Protection Plan" that represents the contract between mission management and the Planetary Protection Officer regarding the means by which the mission will meet them.

### 3.8.2 Planetary protection for human missions

A number of workshops were held in the 1990s and early 2000s, both within the U.S. and jointly with international partners, that have resulted in an international consensus on planetary protection policy and its implementation for human missions.[81] This international consensus will be used as a basis for COSPAR guidelines, and will feed into the development of an NPR document for human mission implementation. One outcome of these workshops has been the recognition that there are no basic differences between planetary protection principles for human and robotic missions. Thus, a set of fundamental assumptions regarding human mission activities underlies the application of planetary protection policy and requirements.

Because humans invariably carry associated microbial populations that are necessary for our survival, forward contamination is a significantly greater risk with human missions than robotic missions. For this reason, the greater capabilities of human explorers can contribute to the astrobiological exploration of the solar system only if human-associated contamination is controlled and understood. Even with improvements in human support technologies, it will be not be possible for all human-associated processes and mission operations to be conducted within entirely closed systems. Even so, for some targets, such as Mars, it may be sufficient to restrict biological contaminants only from certain limited areas of the planet – the so-called "Special Regions" – in which Earth organisms may be able to propagate.

Backward contamination is an ongoing risk for human missions both during operations and the return of the crew to Earth, in contrast to robotic missions for which contamination can be controlled effectively by containment of samples after return. Crew members who are exploring other planets will inevitably be exposed to planetary materials,

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[80] "Treaty on Principles Governing the Activities of States in the Exploration and Use of Outer Space, Including the Moon and Other Celestial Bodies." (entered into force, October 10, 1967).

[81] Safe On Mars: Precursor Measurements Necessary to Support Human Operations on the Martian Surface. (2002) National Research Council, Space Studies Board, National Academy Press, Washington, D.C., <www.nap.edu>.

Planetary Protection Issues in the Human Exploration of Mars, Pingree Park Final Workshop Report. (2005) Criswell, M.E., M.S. Race, J.D. Rummel, and A. Baker (ed.s), NASA / CP-2005-213461.

Life Support & Habitation and Planetary Protection Workshop Final Report. (2006) Hogan, J.A., J.W. Fisher, M.S. Race, J. Joshi, and J.D. Rummel (ed.s), NASA / TM-2006-213485.

Joint NASA/ESA Workshop on Mars Planetary Protection and Human Systems Research and Technology. Held May 19-20, 2005 at the European Space Research and Technology Centre (ESTEC) in Noordwijk, Netherlands.

NASA Advisory Council Workshop on Science Associated with the Lunar Exploration Architecture. Held Feb. 27-March 2, 2007 in Tempe, Arizona, USA.

as was first demonstrated during the Apollo Program. The recent consensus on planetary protection for human missions argues that to the maximum extent practicable, these exposures should occur under controlled conditions; but it is understood that exposure cannot be eliminated entirely. Accordingly, careful planning will be required to understand the nature and consequences of such exposures to avoid the need for decisions about whether crew members will be allowed to return to Earth. For some missions, the potential that human explorers may be exposed to extraterrestrial life must be part of the plan. Nevertheless, safeguarding the Earth from harmful backward contamination must always be the highest planetary protection priority.

These assumptions lead directly to a set of general policy considerations that should be applied to all human missions. Clearly, to mitigate potential danger to astronauts and the Earth, planetary protection must be considered a critical element for the success of human missions, and evaluation of planetary protection requirements should be considered in all human mission subsystems development. However, planetary protection risks are among the many risks that a mission faces, and they should be identified and evaluated together with other mission risks that are to be reduced, mitigated, or eliminated to enable mission success. To ensure proper implementation of planetary protection provisions during the mission, general human factors will need to be considered along with planetary protection issues when developing technologies and procedures. Likewise, planetary protection considerations should be included in human mission planning, training, operations protocols, and mission execution.

Finally, to facilitate compliance and rapid mitigation when required, a crew member who is on board the mission should be given primary responsibility for implementation of planetary protection provisions affecting a crew during the mission. Planetary protection provisions are too important, and in a crisis they may become too urgent, to build in the requirement that discussions are subject to a 20-minute round-trip delay; i.e., with ground control.

### 3.8.3 Considerations for planetary protection implementation

Several factors will contribute to the control of forward contamination during human missions. Exploration, sampling, and base activities must be designed and developed to ensure effective operations while maintaining the required level of planetary protection activity. Particular challenges involve the processes that are associated with exploration, including EVA activities; therefore, egress/ingress-specific technologies and procedures will need to be developed, characterized and optimized. Systems will be required to allow controlled, sterile surface and subsurface sampling operations so that uncontaminated samples can be obtained, probably using robotic assistants. An inventory of microbial populations and organic materials that is carried aboard spacecraft should be established prior to launch and maintained throughout a mission to provide a record of contamination that could be potentially released by human-associated spacecraft and transportation systems. Monitoring technologies will be required to evaluate the level of contamination that is released by human-associated activities on an ongoing basis, as will technologies to mitigate contamination resulting from an off-nominal release event. The inventory and monitoring activities will support both planetary protection and crew health objectives.

The ability to maintain the crew in a healthy state is critical to ensure mission success. As part of normal crew health monitoring, basic tests of the medical condition of the crew members and their responses to pathogens or adventitious microbes should be developed, provided, and employed regularly during the mission. This information will also be essential for evaluating the effects of exposure events; i.e., to understand their severity and assess the need for quarantine measures. To permit the isolation of potentially contaminated or infectious crew member(s), a quarantine capability for both individual crew members as well as for the entire crew should be provided during the mission. After the mission, a quarantine capability and appropriate medical testing should be provided, and could be implemented in conjunction with a health monitoring and stabilization program as the crew members are integrated back into the general population.

To minimize the potential for harmful exposure events, operations for human missions shall include isolation of humans from direct contact with planetary materials until initial testing can provide verification that exposure to the material is safe for humans. Exploration, sampling, and base activities shall be performed in a manner that will limit inadvertent exposure of humans to material(s) from untested areas. For the initial landing site, testing will probably have been performed as a part of precursor mission activities; but a means for allowing controlled access to untested areas, or areas that are considered unsafe, must be provided during human missions. Sterilized and cleanable robots, which are under appropriate operational constraints, are one suitable approach for ensuring appropriate access.

### 3.8.4 Operational constraints for human missions to Mars

The surface of Mars is very cold and dry; in most places, it is too cold or dry to permit the growth and reproduction of Earth organisms. However, the subsurface of Mars is likely to be warmer and wetter, and, therefore, more hospitable to Earth life. Certain geological formations on the martian surface suggest that liquid water may occasionally be present, and such formations have been termed Special Regions that merit special protection. A Mars Special Region is currently (2007) defined by COSPAR as "a region within which terrestrial organisms are likely to propagate, OR A region that is interpreted to have a high potential for the existence of extant Martian life forms." Thus, Special Regions, as currently defined, encompass both certain features on the surface of Mars and, conservatively, the entire subsurface below the depth where surface equilibrium conditions prevail.

A future definition of Special Regions is likely to invoke a combination of specific parameters that can be measured accurately. The MEPAG has proposed two parameters as being suitable: temperature and water activity. Water activity is a measure of the availability of water to participate in chemical or biological reactions, and, in most cases, it can be considered as equivalent to the relative humidity of an environment divided by 100. The Planetary Protection Subcommittee has recommended to NASA that limits be set on these two parameters to define Special Regions: a "water activity" of 0.5 or greater, and (simultaneously) a temperature of –25°C or warmer. These numeric limits will be revisited regularly and modified, as appropriate, based on the most up-to-date scientific information. The intent is to define as Special Regions only those locations on Mars that have water available at a temperature that could support life. Under our current understanding, this encompasses a small fraction of the surface of the planet, excluding both equatorial and polar latitudes.

In line with current planetary protection policy for robotic missions, human missions to Mars shall avoid the inadvertent introduction of Earth organisms or organic molecules into Mars Special Regions, as well as the inadvertent exposure of humans to martian materials. Mission cleanliness and containment capabilities will feed directly into landing site selection and operational accessibility to scientifically desirable locations on Mars. Exploration of Special Regions, including access to subsurface ice or water, shall be restricted appropriately relative to the microbial and organic cleanliness of the human-associated or robotic systems that are used. Calculations that are based on this approach will determine the levels and kinds of contamination that will be allowed for a particular human mission activity.

Astronaut safety is one of the highest priorities for human missions. The SSB has recommended that a set of operational constraints be implemented for human mission activities that are designed to ensure the safety of astronauts[82]. These constraints include the designation of zones of minimum biological risk (ZBRs), which are regions that have been demonstrated to be safe for humans; astronauts will only be allowed in areas that have been demonstrated to be safe. Initial identification of ZBRs for human landing sites shall be performed through direct investigation by precursor missions, either on the ground or remotely (figure 3-60). Areas around human habitats shall be cleared as "safe" through appropriate robotic exploration, after which human EVAs would be allowed. Special Regions shall only be accessed using sterilized clean equipment to prevent forward contamination. Facilities for transfer of collected samples under appropriate contamination control will be required to prevent backward contamination.

### 3.8.5 Guidelines for practical implementation

Although specific requirements for human missions to Mars have not yet been established, a set of guidelines to assist planning and early decision-making can be assembled based on the consensus outcomes from the various NASA and international workshops that were held over the last decade. Listed guidelines cover four major areas of human activity: initial landing sites, human habitats, EVAs, and the potential for ISRU.

Landing sites shall be selected such that nominal or off-nominal mission operations shall have a low probability of allowing mission-associated microbial or organic contamination to enter Mars Special Regions either horizontally or vertically. This includes mission-induced Special Regions.

[82]See the SSB recommendations that are given in the 2002 Safe on Mars report.

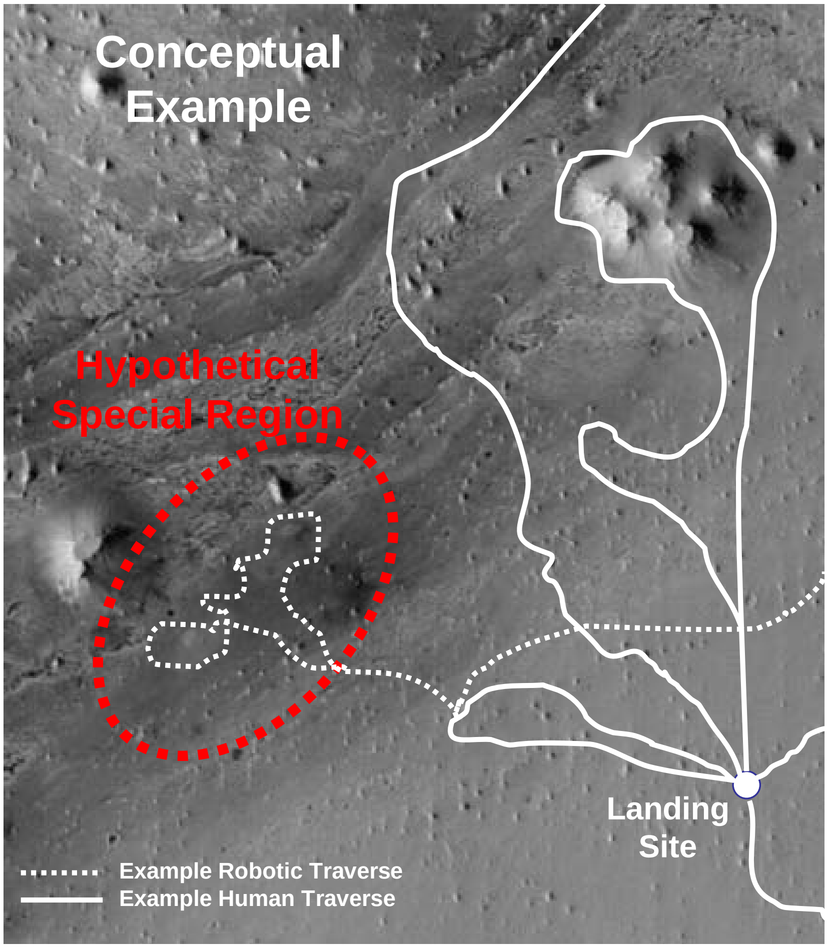



**Figure 3-60.** Notional operational scenario near safe zones.

Human habitation modules shall be located and operated so as to ensure that mission-associated microbial or organic contamination shall have a low probability of entering the Mars Special Regions. Closed-loop life support and recycling systems that release minimal contamination should be developed. Distances from Special Regions should be determined based on determinations of contaminants that are released and data addressing transport of material by surface winds and other processes. Calculations should include a conservative safety margin.

Human EVAs shall be planned and executed to ensure that mission-associated microbial or organic contamination shall have a low probability of entering Mars Special Regions. Tools that are capable of attaining and retaining the required cleanliness shall be used to explore and sample these regions. Appropriate equipment shall be provided to enable transfer of materials from collection devices to study facilities while maintaining the required levels of cleanliness and containment.

ISRU activities shall be planned and executed to ensure that mission-associated microbial or organic contamination shall have a low probability of entering the Mars Special Regions. Approaches for ISRU shall protect humans and human-associated systems from uncontrolled contact with material from Mars Special Regions.

### 3.8.6 Responding to off-nominal events

Off-nominal events must be anticipated for any mission, and appropriate planning must be used to mitigate the effects. Example events that could result in forward contamination of Mars include a spacecraft crash, habitat or mobility systems breach, a waste containment breach, and poor sterilization of systems that are accessing Special Regions. Example events that could produce backward contamination of human astronauts and their support systems include laboratory accidents and breaches in a Mars sample containment facility, a habitat, or mobility systems. Of immediate concern for astronaut survival would be failures in human support systems, including advanced life support systems or components, habitat integrity, EVA systems such as suits or rovers, power systems, etc.

Amelioration of planetary protection concerns would involve identification and documentation of the incident, followed by remediation when possible.

#### 3.8.7 Summary

In order for human exploration to be conducted safely, and to contribute successfully to our understanding of the rest of the solar system, planetary protection considerations must be part of overall mission planning and execution. In the specific case of the human exploration of Mars, planetary protection requirements will be imposed to protect astronauts, the Earth, and the potential for scientific discovery. Compliance with those requirements will be challenging, but they will also be worthwhile. Decades of experience with robotic missions and the lessons learned from the Apollo experience have provided us with important insights into both the nature and timing of the restrictions that make sense from the standpoint of the mission, and in terms of protecting the Earth from those unknowns that the missions will, in part, be sent to address. Accordingly, the development of appropriate technologies and procedures for planetary protection compliance must be included as part of the ongoing mission design process, and must be eyed when developing other support and mobility systems for the human explorers. The result will be scientifically sound and productive missions that preserve the pay-off potential of future exploration, and ensure that our astronauts can come safely home.

# 4 DESIGN REFERENCE ARCHITECTURE 5.0 OVERVIEW

In previous sections, the groundwork for this current Mars DRA were discussed, including the goals and motivations for exploring the surface of Mars and results of trade studies that were conducted to narrow the implementation options for these exploration missions. This section will describe the resulting DRA as a consolidated whole. The risk mitigation approach that was applied to the trade studies that lead to this Reference Architecture, and which will be used for future analyses and trades, is explained. In addition, this section will describe the rationale behind one of the key ground rules for this DRA – the choice of a six-person crew size – and characteristics of essential infrastructure – mission operations, communications, and navigation – that have not been previous discussed.

## *4.1 Mission Overview*

The Mars DRA describes the systems and operations that would be used by humans on the first three missions to explore the surface of Mars. These missions would occur on three consecutive trajectory opportunities sometime within the next several decades. A three-mission set was chosen for this Reference Architecture for several reasons:

- The development time and cost to achieve the basic capability to carry out a single human Mars mission are of a magnitude that a single mission or even a pair of missions could not be justified.
- Three consecutive missions will require approximately 10 years to complete, a period of time that is sufficient to achieve basic program goals and acquire a significant amount of knowledge and experience, making this a likely point in time to consider new goals and improved architectures to achieve them.

In addition, the first three human Mars missions are assumed to have been preceded by a sufficient number of test and demonstration missions on Earth, in LEO, on the moon, and at Mars (by robotic precursors) to achieve a level of confidence in the architecture such that the risk to human crews is considered acceptable.

A crew of six will be sent on each of these missions, and each crew would visit a different location on Mars. The rationale for a crew of this size will be discussed in more detail in a subsequent section, but can be summarized as being judged to be a reasonable compromise between the skill mix and level of effort for missions of this complexity and duration balanced with the magnitude of the systems and infrastructure that would be needed to support this crew. Visiting three different sites is based on a recommendation from a special committee of the MEPAG, which was described in a previous section. The science and exploration rationale for visiting three different sites reflects a recognition that a planet as diverse as Mars is not likely to be adequately explored and understood from the activities that can take place at a single site. However, this three-site assumption does not preclude returning to any of the sites should there be a compelling need to do so.

With these assumptions and the results from a series of trade studies, which were previously described in detail, many but not all of the key features of the Mars DRA can be summarized. Based on these trade studies, each of the three missions would use the so-called long-stay trajectory option. A portion of the assets of each mission, specifically the SHAB and other surface mission equipment as well as the crew ascent vehicle, are sent to Mars one opportunity prior to the crew. This is the so-called "pre-deploy" or "split-mission" option. This option allows a lower-energy trajectory to be used for these pre-deployed assets, which results in more useful payload mass to be delivered to Mars for the propellant that is available. In addition, a decision was made to take advantage of the aerocapture technique on arrival at Mars to further enhance the amount of useful payload that can be delivered due to the favorable trade of using the atmosphere of Mars to capture these payloads into orbit around Mars when compared with an equivalent propulsion system. It should be noted that the human crew does not use this aerocapture technique, due primarily to the size of the vehicle that is transporting them. The decision to pre-position some of the mission assets better accommodates the decision to make part of the ascent propellant at Mars, using the atmosphere as the raw material source for this ascent propellant. This use of in-situ resources and the equipment to process them into useful commodities results in a net decrease in the total mass that is needed to complete a mission. A nuclear power source was found to be better suited, when compared to an equivalent solar system, for producing this ascent propellant. This choice was further supported by the fact that this power system would be more than adequate to meet the needs of the human crew members when they arrive, which occurs after all of the necessary propellants have been produced.

An expanded discussion of key aspects of the Reference Architecture, given the results of these trade studies and their interaction, can be found in the following subsections of section 4.1.

Insufficient time and resources were available in this analysis to make a recommendation for two major areas of the Reference Architecture: the in-space propulsion system type (a high $I_{sp}$ chemical system vs. a nuclear thermal system) and specific details of the surface systems to be used by the crew. Alternative approaches for exploring the surface are still under discussion and are anticipated to be examined for the foreseeable future. (These alternate approaches will be discussed in a later section.) Under these circumstances, one of the approaches that most closely follows previous DRAs was selected as the nominal approach so that affected systems can be sized and reported here. An expanded discussion of the surface mission portion of the DRA will be discussed in section 4.2 below.

### 4.1.1 Mars Design Reference Architecture interplanetary trajectory and mission analysis

Specific high-thrust trajectories were analyzed for round-trip crewed missions to Mars with Earth departure dates ranging from 2030 to 2046. Mission opportunities occur approximately every 2.1 years in a cycle that repeats every 15 years. (The trajectories from one 15-year cycle to the next do not match exactly, but they are very similar and are sufficient for initial planning purposes. The duration that is required for a more exact match is 79 years.) Along with the crewed missions, one-way cargo delivery trajectories are also generated that depart during the opportunity preceding each crewed mission. Each cargo mission delivers two vehicles to Mars.

#### *4.1.1.1 Methodology and assumptions*

The trajectories that would be used for human crews balance low interplanetary trip times with the cost (i.e., propellant) of achieving the missions. This is facilitated by allowing long Mars stay times. For each opportunity, the outbound and inbound transit times are minimized such that desired departure energies (determined by $V_\infty$) are not exceeded. Again, the Mars stay time is allowed to vary so that the lowest interplanetary flight times are possible. The supporting cargo flights follow minimum energy trajectories with no restriction on the outbound transit times. The cargo departures occur approximately 2.1 years before each crew mission. This allows confirmation that the cargo elements have successfully reached their destinations and are functioning properly before the crew leaves Earth.

In this analysis, all vehicles depart from a 407-km circular orbit, and a two-burn Earth escape is performed to reduce the gravity-loss penalties. At Mars, the vehicles are inserted into a 1-day sol orbit (250 km × 33,793 km). For the cargo missions, both propulsive and aerocapture cases were investigated; while for the crewed vehicles, only propulsive orbital insertions were considered. These and the other trajectory assumptions are given in table 4-1.

**Table 4-1.** Overall Trajectory Assumptions

| | |
|---|---|
| **All Vehicles** | |
| Earth departure orbit: | 407 km (circular) |
| # of departure burns: | 2 |
| Mars arrival orbit: | 250 km by 33,793 km (1 day sol) |
| **Cargo Vehicles** | |
| Considered Mars capture method: | Aerocapture<br>Propulsive |
| Maximum Mars arrival $V_\infty$ | 5.450 km/s (Aerocapture cases) |
| **Crew Vehicle** | |
| Maximum(*) TMI $V_\infty$: | 4.290 km/s |
| Mars capture method: | Propulsive |
| Maximum MOI $V_\infty$: | 4.176 km/s (set by 2031 opportunity) |
| Mars stay time: | Not restricted |
| Maximum TEI $V_\infty$: | 3.487 km/s |
| Earth capture method: | Direct entry |
| Maximum Earth return $V_\infty$: | 6.813 km/s |

(*) Exceeded on 2039 and 2041 opportunities to keep flight times low (≤ 225 days)

#### *4.1.1.2 Cargo and crew mission profiles*

Figures 4-1 and 4-2 show representative trajectories for the crewed and cargo missions, respectively. The displayed crewed profile corresponds to the all-propulsive 2037 opportunity with transit times of 174 days outbound and 201 days inbound. The Mars stay time is 539 days, and the total mission duration is 914 days. Again, note that the majority of the mission duration is spent on the surface of Mars, while the interplanetary transit times are reduced to minimize the exposure of the crew to harmful solar and GCR. The supporting cargo vehicle departs Earth a little more than 2 years before the crewed mission and follows a minimum-energy trajectory. The trip time of 202 days is the quickest cargo flight time that was observed over the dates that were analyzed.

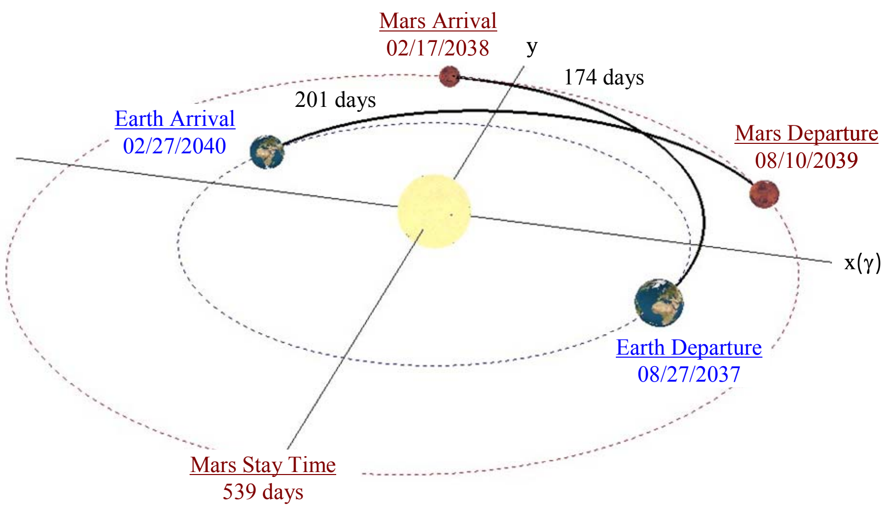



**Figure 4-1.** Crewed 2037 all-propulsive Mars mission.

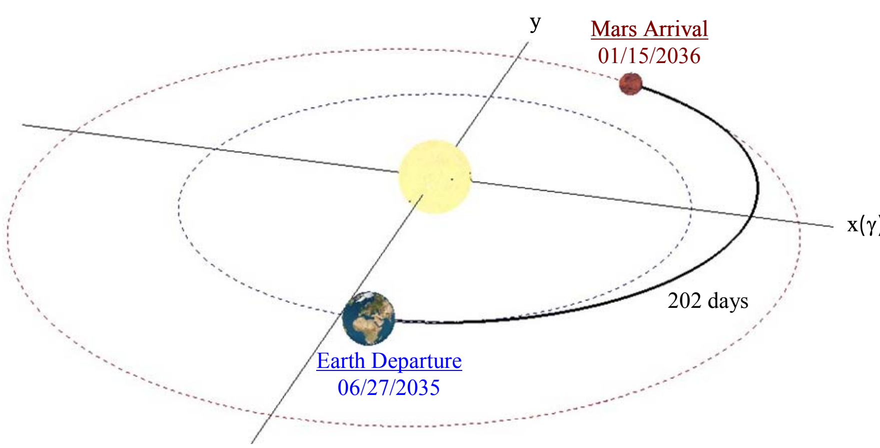



**Figure 4-2.** All-propulsive cargo trajectory.

#### *4.1.1.3 Mission information*

Figure 4-3 shows the mission timelines for all of the analyzed opportunities. Each cargo mission represents two vehicles that launch approximately 2 years before the supporting crewed mission. Included in the plot are the estimated LEO loiter times for the cargo vehicles. These loiter times allow a more reasonable Earth-to-orbit launch schedule between the crewed missions and the cargo vehicles that are supporting the subsequent opportunities.

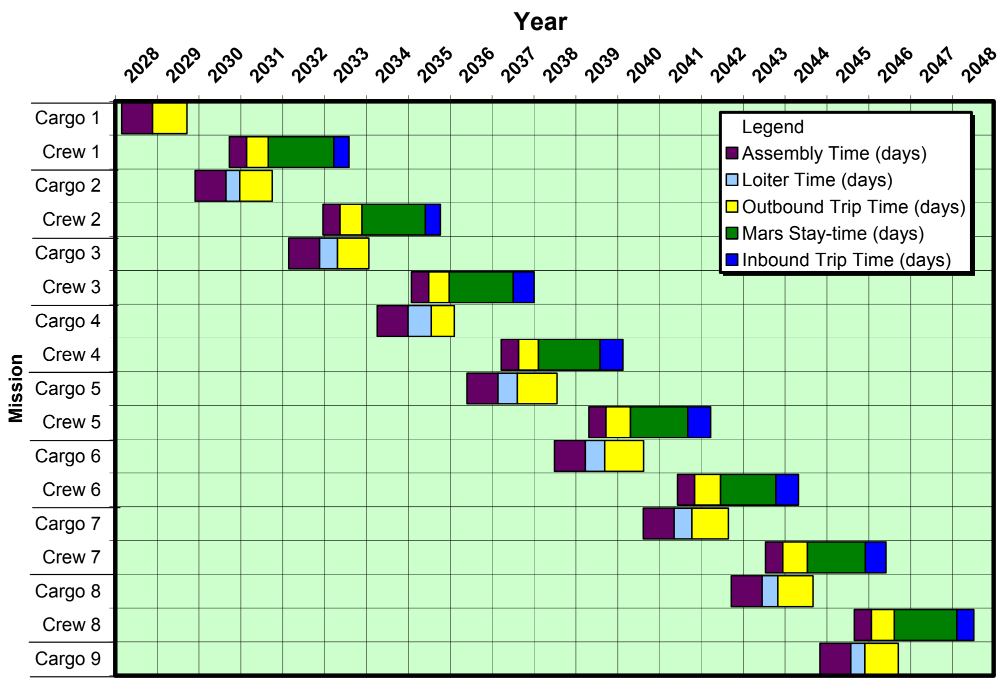



**Figure 4-3.** Crewed and cargo mission timelines.

Figures 4-4 and 4-5 show the crewed and cargo mission delta-Vs over the dates of interest. For the crewed missions, all TMI maneuvers are designed to achieve the maximum allowed Earth departure $V_\infty$, except for the 2031 and 2035 opportunities. As the time of flight is reduced in the 2031 case, the maximum allowed Mars arrival $V_\infty$ is exceeded before the Earth departure limit is reached. The 2035 case that is shown in figure 4-4 includes a 180-day outbound trajectory. In fact, the flight time can be reduced to around 140 days before violating the end-point constraints. The longer flight time is shown to indicate that, if desired, more crewed payload can be delivered with a reasonably fast transit. For the cargo missions that are shown in figure 4-5, fewer restrictions are placed on the trajectories. The flight times are allowed to vary to minimize the total effect of the TMI and MOI (in the all-propulsive missions) maneuvers. The all-propulsive vehicle design is determined by the worst-case delta-Vs, which, for the TMI maneuver, occurs in 2037 while the 2030 opportunity contains the maximum MOI requirement. In general, the aerocapture TMI variation resembles that of the propulsive MOI cases with slightly lower values. This is due to the relaxed effect of the Mars arrival velocity. The 2043 cases appear to contradict this statement, but the aerocapture case represents a fast transit (240 days) that is feasible because no propellant is required for MOI. If the aerocapture vehicle follows a longer trajectory, the TMI delta-V is less than or equal to that of the all-propulsive mission.

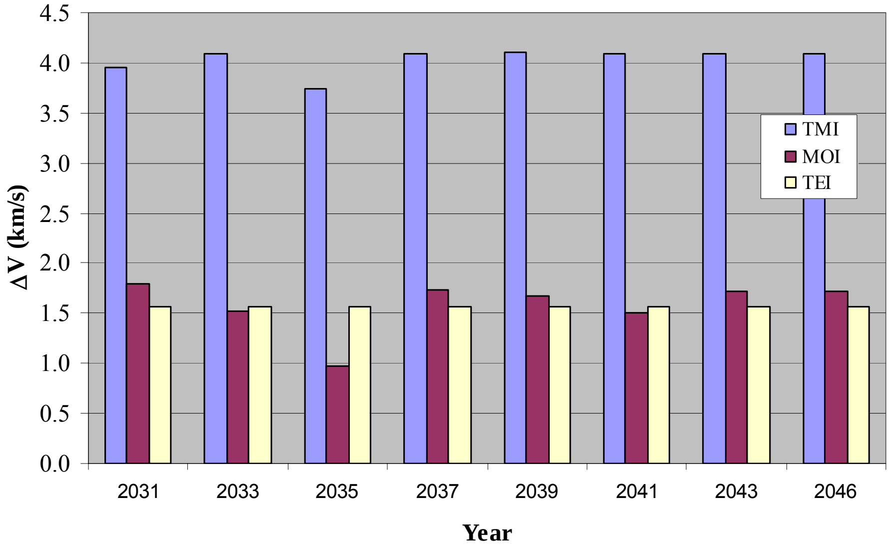



**Figure 4-4.** All-propulsive crewed departure and capture maneuvers.

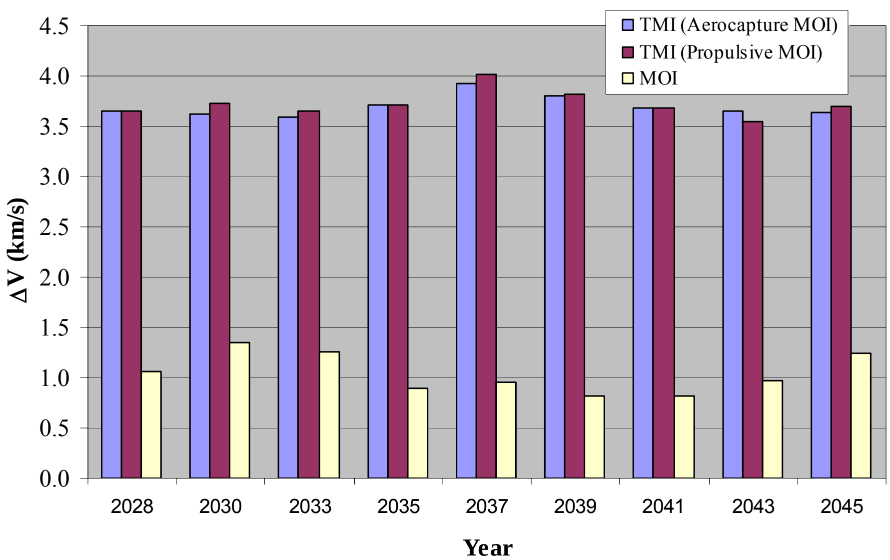



**Figure 4-5.** Cargo departure and capture maneuvers.

#### *4.1.1.4 Trajectory data*

Detailed trajectory information for the crewed missions is provided in table 4-2. Along with the mission times and propulsive maneuvers, the required LEO inclinations at departure are also given. For LEO inclinations greater than 28.5 degrees, plane changes, which are not included in the TMI delta-Vs, are required. The appropriate time to perform these maneuvers is at apogee passage after the first TMI burn. Note that quicker outbound trajectories (not shown) exist for the 2033 opportunity and have higher LEO inclinations. If propellant is included for the large plane change that is required in 2039, the faster outbound flight is possible in 2033.

**Table 4-2.** Crewed Mission Trajectory Data (Propulsive MOI into 1-day sol orbit)

| Earth Departure | | | | Mars Arrival | | | Mars Departure | | | Earth Return | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| Earth Departure Date (GMT) | LEO Incl. (deg) | TMI $V_\infty$ (km/s) | TMI ΔV** (km/s) | Outbound TOF (days) | MOI $V_\infty$ (km/s) | MOI ΔV** (km/s) | Mars Stay Time (days) | TEI $V_\infty$ (km/s) | TEI ΔV** (km/s) | Return TOF (days) | Earth Return $V_\infty$ (km/s) |
| 2/23/2031 | 28.5 | 3.940 | 3.955 | 190 | 4.176 | 1.789 | 572 | 3.847 | 1.573 | 134 | 6.813 |
| 5/21/2033* | 34.6 | 4.290 | 4.090 | 191 | 3.756 | 1.517 | 553 | 3.847 | 1.573 | 133 | 5.762 |
| 7/3/2035 | 28.5 | 3.327 | 3.746 | 180 | 2.792 | 0.971 | 560 | 3.847 | 1.573 | 183 | 4.438 |
| 8/27/2037 | 32.6 | 4.290 | 4.090 | 174 | 4.101 | 1.739 | 539 | 3.847 | 1.573 | 201 | 5.598 |
| 10/01/2039 | 50.2 | 4.318 | 4.100 | 213 | 4.005 | 1.675 | 502 | 3.847 | 1.573 | 199 | 6.813 |
| 11/14/2041 | 36.3 | 4.301 | 4.094 | 225 | 3.733 | 1.502 | 486 | 3.847 | 1.573 | 197 | 6.797 |
| 12/24/2043 | 28.5 | 4.290 | 4.090 | 215 | 4.067 | 1.716 | 507 | 3.847 | 1.573 | 180 | 6.813 |
| 2/6/2046 | 28.5 | 4.290 | 4.090 | 200 | 4.077 | 1.723 | 547 | 3.847 | 1.573 | 148 | 6.813 |

*Other potential trajectories exist with a shorter outbound time of flight (TOF) but higher LEO inclinations.
**All delta-Vs include gravity losses but no plane changes.

The one-way trajectory data that are provided in the following tables represent approximate minimum energy transfers, with no constraints placed on the transit durations. Table 4-3 gives the mission information for cargo vehicles that use engine thrust to capture into Mars orbit, while table 4-4 provides the data for cargos that perform aerocapture maneuvers at Mars. Note the LEO plane change that is required in the all-propulsive 2033 opportunity.

**Table 4-3.** Cargo Mission Trajectory Data (Propulsive MOI into 1-day sol orbit)

| Earth Departure Date | LEO Inclination (deg) | TMI $V_\infty$ (km/s) | TMI $C_3$ ($km^2/s^2$) | TMI ΔV* (km/s) | TOF (days) | MOI $V_\infty$ (km/s) | MOI ΔV* (km/s) |
|---|---|---|---|---|---|---|---|
| 11/23/2028 | 28.9 | 3.020 | 9.120 | 3.657 | 300 | 2.970 | 1.063 |
| 12/25/2030 | 28.5 | 3.282 | 10.772 | 3.733 | 283 | 3.482 | 1.350 |
| 04/17/2033 | 55.3 | 3.014 | 9.085 | 3.655 | 200 | 3.318 | 1.254 |
| 06/27/2035 | 28.5 | 3.218 | 10.356 | 3.715 | 202 | 2.631 | 0.892 |
| 08/16/2037 | 28.5 | 4.091 | 16.736 | 4.013 | 349 | 2.767 | 0.958 |
| 09/19/2039 | 28.5 | 3.560 | 12.674 | 3.823 | 340 | 2.474 | 0.818 |
| 10/20/2041 | 28.5 | 3.133 | 9.816 | 3.689 | 319 | 2.484 | 0.823 |
| 11/11/2043 | 28.6 | 3.009 | 9.054 | 3.553 | 307 | 2.794 | 0.972 |
| 12/11/2045 | 28.5 | 3.145 | 9.891 | 3.703 | 291 | 3.276 | 1.238 |

*All delta-Vs include gravity losses but no plane changes.

**Table 4-4.** Cargo Mission Trajectory Data (Aerocapture MOI into 1-day sol orbit)

| Earth Departure Date | LEO Inclination (deg) | TMI $V_\infty$ (km/s) | TMI $C_3$ ($km^2/s^2$) | TMI ΔV* (km/s) | TOF (days) | MOI $V_\infty$ (km/s) | MOI ΔV* (km/s) |
|---|---|---|---|---|---|---|---|
| 12/09/2028 | 28.5 | 3.008 | 9.048 | 3.653 | 222 | 4.901 | 0.000 |
| 02/20/2031 | 28.5 | 2.871 | 8.243 | 3.615 | 319 | 5.450 | 0.000 |
| 04/28/2033 | 28.5 | 2.789 | 7.779 | 3.593 | 274 | 4.379 | 0.000 |
| 06/23/2035 | 28.5 | 3.192 | 10.189 | 3.707 | 195 | 2.696 | 0.000 |
| 09/06/2037 | 28.5 | 3.854 | 14.853 | 3.925 | 395 | 3.346 | 0.000 |
| 09/27/2039 | 28.5 | 3.490 | 12.180 | 3.800 | 361 | 2.704 | 0.000 |
| 10/19/2041 | 28.5 | 3.132 | 9.809 | 3.689 | 316 | 2.486 | 0.000 |
| 11/23/2043 | 28.5 | 3.005 | 9.030 | 3.652 | 240 | 4.249 | 0.000 |
| 01/23/2046 | 28.5 | 2.930 | 8.585 | 3.632 | 331 | 5.179 | 0.000 |

*All delta-Vs include gravity losses.

## *4.2 Surface Reference Mission Overview*

For the long-stay mission sequence, two cargo elements are pre-positioned to support the crew's surface mission: the DAV and an SHAB; other surface equipment is divided between these two cargo elements so as to give each of them approximately the same total mass at launch. Both of these cargo elements are launched in the same minimum-energy opportunity just over 2 years prior to the launch of the crew. The cargo elements arrive at Mars approximately 8 months later, and the SHAB is placed into an orbit where it can later rendezvous with the crew members when they first arrive. The DAV lands autonomously at the selected surface exploration site, where it deploys the nuclear reactor and begins producing ascent propellant and other commodities that are needed by the crew. Both of these vehicles are checked for proper function and then placed into a minimal operating configuration to remain in this state for over 2 years before the arrival of the crew. The next minimum-energy window (for the next cargo elements) opens shortly before the fast transit trajectory window for the first crew, but these launch windows are still close enough that a combined launch campaign at KSC is required. The launch campaign for the second crew's cargo and for the first crew begins as much as 1 year before either windows open so that all of these elements are ready for their respective departures. The first crew arrives before the cargo elements for the second mission and nominally uses the assets that were launched more than 2 years previously. However, should either the DAV or SHAB suffer a failure between the time the first crew launches from Earth and when it leaves Mars to return to Earth, the second set of cargo elements can be used, thus potentially preventing LOM or LOC. This is a unique feature of the pre-deployment strategy when applied to the long-stay mission; this overlap of assets is not available for any of the short-stay options or for the all-up strategy. This sequence is depicted in figure 4-6.

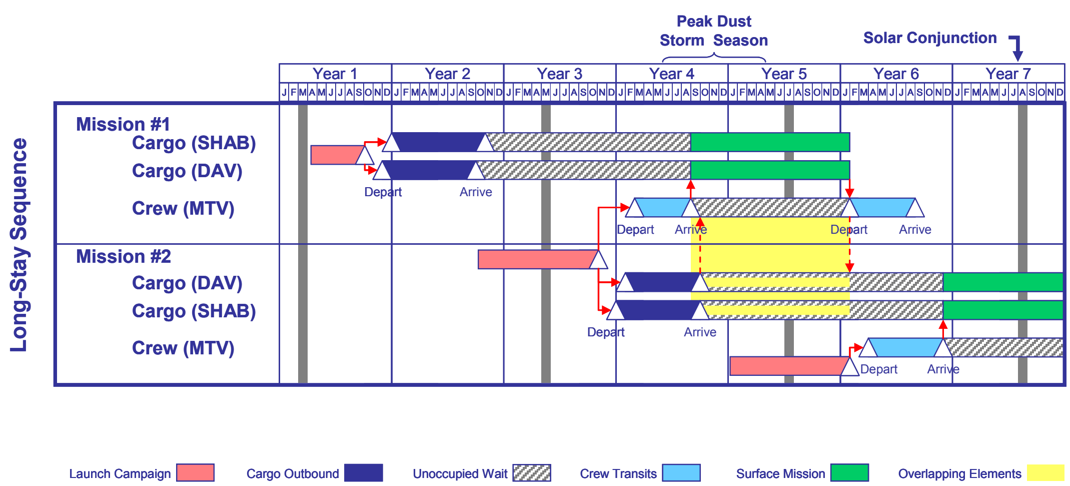



**Figure 4-6.** Nominal launch and arrival sequence for the first two human Mars missions.

Candidate surface sites will be selected based on the best possible data that are available at the time of selection, the operational difficulties associated with the site, and the collective merit of the science and exploration questions that can be addressed at the site. Data that are available for site selection will include remotely gathered data sets plus data from any landed mission(s) in the vicinity plus interpretive analyses that are based on these data.

Figure 4-7 illustrates a notional series of traverses to features of interest at the junction of the Isidis Planatia and Syrtis Major regions. No particular preference is being given to this particular site; it is included here to illustrate some general features of a human exploration mission and the resulting implications for operations at such a site.

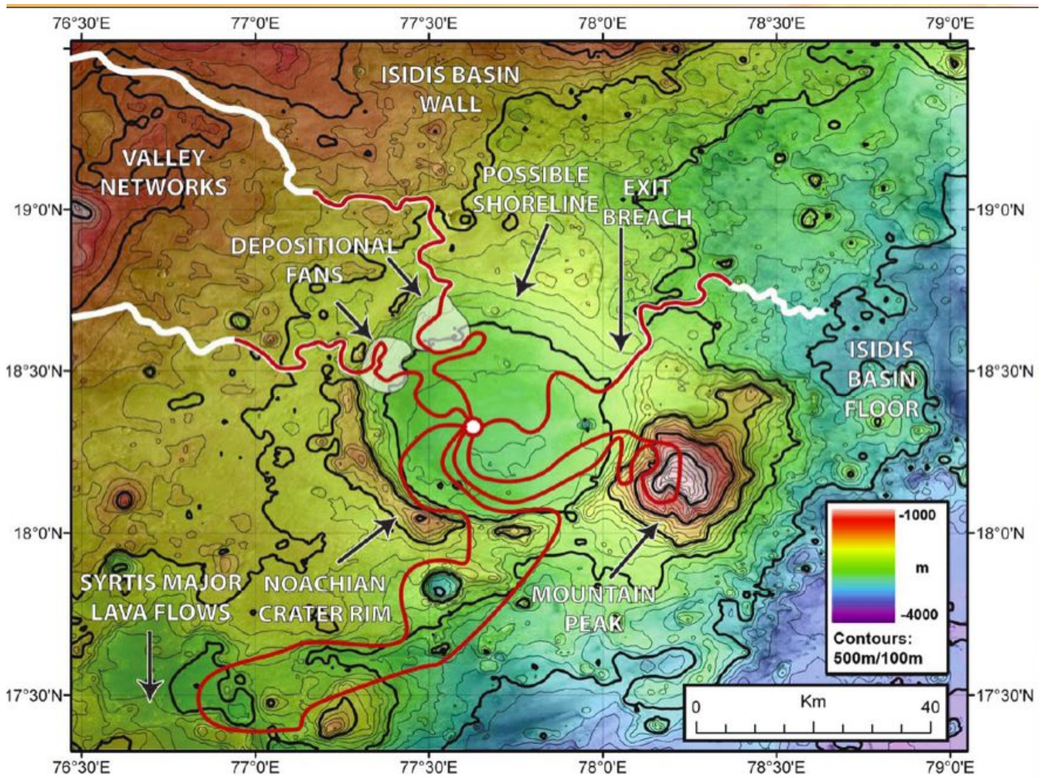



**Figure 4-7.** Notional traverses near the junction of the Isidis Planatia and Syrtis Major.

From an operational perspective, this location has a relatively broad, relatively flat, centrally located area where the cargo elements can land in relative safety. However, this places these systems and the crew at large distances from features that are of interest to the crew and the science teams. The scale at the lower right indicates that these features of interest are beyond what is currently considered a reasonable walking range for the crew (determined by the distance a crewperson can walk during one charge of power and breathing gases in a portable life support system – roughly 20 km total). Although sites with much more closely space features of interest are certainly possible, they are usually found at the expense of a relatively safe landing site.

One feature of interest is not illustrated here – the subsurface. Understanding the vertical structure of the site will also be of interest, indicating that a drilling capability would be included for each mission and site. The ability to move this drill from location to location would also be desirable.

Three possible approaches to this combined horizontal and vertical exploration need have been developed and are described in more detail on the following pages.

The nominal surface mission scenario that was adopted for this Reference Architecture would have a centrally located, monolithic habitat, two small pressurized rovers, and two unpressurized rovers (roughly equivalent to the LRV that was used in the Apollo missions to the moon) (figure 4-8). Power for these systems would be supplied by a nuclear power plant that would have been previously deployed with the DAV and would be used to make a portion of the ascent propellant. Traverses would be a significant feature of the exploration strategy that is used in this scenario, but these traverses would be constrained by the capability of the small pressurized rover. In this scenario, these rovers have been assumed to have a modest capability: notionally a crew of two, 100 km total distance before being resupplied, and no more than 1 week duration. Thus, on-board habitation capabilities would be minimal in these rovers. However, these rovers are assumed to be nimble enough to place the crew in close proximity to features of interest (i.e., close enough to view from inside the rover or within easy EVA walking distance of the rover). Not all of the crew members would deploy on a traverse, so there would always be some portion of the crew in residence at the habitat. The pressurized rovers would carry (or tow) equipment that would have the capability to drill to moderate depths – 100's of meters – at the terminal end of several traverses.

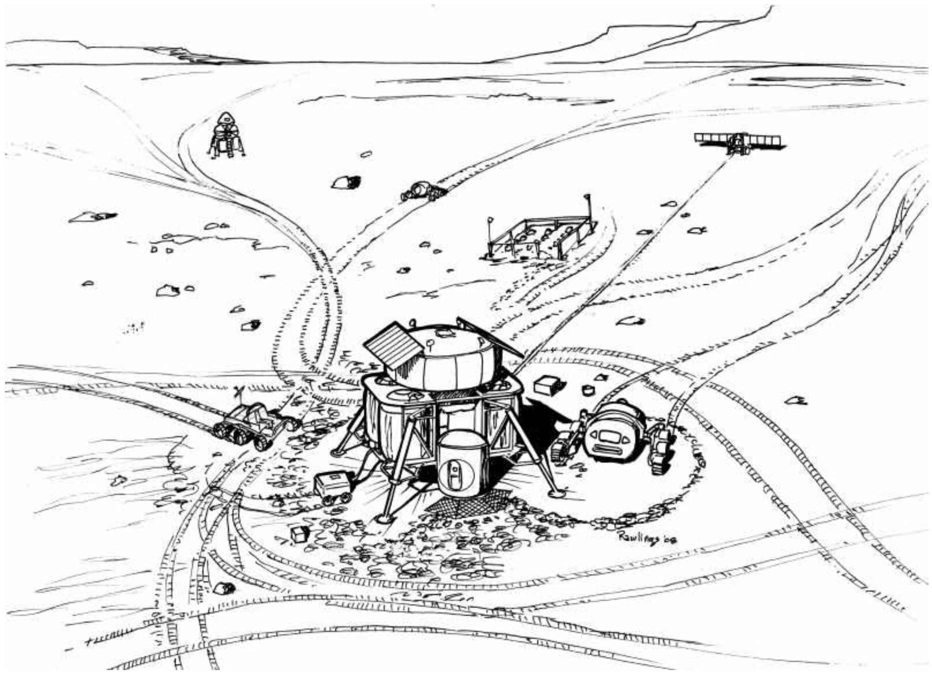


**Figure 4-8.** Notional view of the surface systems that are used in the Mars surface exploration (Rawlings 2007[1]).

The primary habitat would have space and resources allocated for on-board science experiments. The pressurized rovers would carry only the minimal scientific equipment that is deemed essential for field work (in addition to the previously mentioned drill); samples would be returned to the primary habitat and its on-board laboratory for any extensive analysis.

One approach to accomplishing the desired long traverses would be to use the pressurized rovers (or, possibly, the robotic rovers) to pre-position supplies in caches along the proposed route of travel prior to the "full-duration" traverse. Thus, a typical traverse would begin with the crew (or robotic rovers) traveling out a nominal distance (~15 km, or EVA walk-back distance) and establishing a cache of commodities for life support and power (possibly emergency habitation) before returning to the habitat. Some amount of exploration-related activities may be accomplished during this cache deployment phase, but the primary purpose is route reconnaissance and cache establishment. The crew then makes another traverse, establishing a second cache a like distance beyond the first cache. This process continues until all caches in this chain are built up sufficiently for the crew, in the two pressurized rovers, to make the entire round-trip traverse for the time duration needed to accomplish traverse objectives. The amount of time that is required to set up and retrieve these supply caches would depend on the specific conditions for a traverse. However, the timeline in figure 4-9 illustrates how much can be accomplished if approximately 2 weeks are allocated for establishing this string of caches and another 2 weeks to retrieve them. In addition, not all traverses would be long enough to require this type of support. A mixture of cache-supported and -unsupported traverses has been illustrated. Finally, some amount of time would be required to repair and restock the pressurized rovers after each traverse, as well as to conduct any local experiments and plan for the next traverse. A notional 2 weeks between short traverses and 4 weeks between long traverses is illustrated in figure 4-9.

[1] Drawing courtesy of Rawlings, 2007

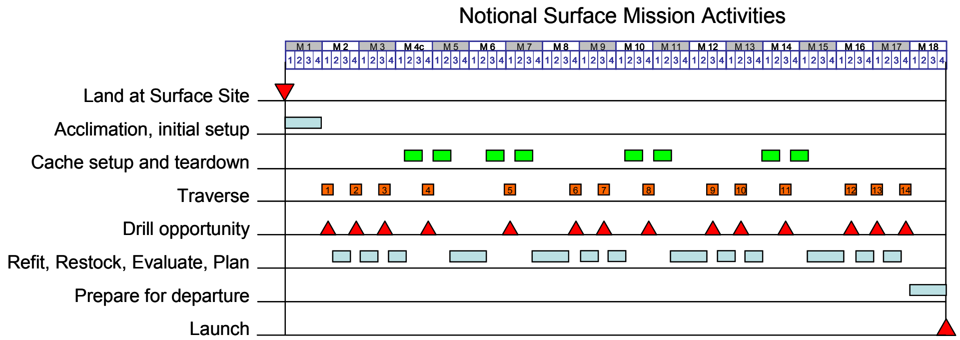



**Figure 4-9.** A notional surface exploration timeline.

With the limited resources available that are for this study, a very preliminary estimate was made of the mass for each of the surface system elements and their distribution between the two cargo elements that are used to deliver them to Mars. Table 4-5 provides a summary of these masses and their distribution.

**Table 4-5.** Mass summary for surface systems and the Distribution

| | | Hab Lander | DAV Lander |
|---|---|---|---|
| **Manifested Item** | **Quantity** | **CBE MASS (kg)** | **CBE Mass (kg)** |
| Crew Consumables | | 1,500 | 4500 |
| Science | | 0 | 1,000 |
| Robotic Rovers | 2 | 0 | 500 |
| Drill | 1 | 0 | 1,000 |
| Unpressurized Rover | 2 | 500 | 0 |
| Pressurized Rover | 2 | 8,000 | 0 |
| Pressurized Rover spares | | (included above) | 0 |
| Pressurized Rover growth | | 1,600 | 0 |
| Pressurized Rover power | 2 | 0 | 1,000 |
| Traverse Cache | | 0 | 1,000 |
| Habitat | 1 | 16,500 | 0 |
| Habitat growth | | 5,000 | 0 |
| Habitat spares | | (included above) | 0 |
| Stationary Power System | 2 | 7,300 | 7,300 |
| ISRU Plant | 1 | 0 | 1,305 |
| | | | |
| Ascent stage 1 (no $LO_2$) | | 0 | 12,156 |
| Ascent stage 2 (no $LO_2$) | | 0 | 9,330 |
| 30-day temporary habitat | | 0 | 0 |
| | | | |
| Descent Stage (wet) | | 23,300 | 23,300 |
| | | | |
| Aeroshell | | 437,000 | 437,000 |
| | | | |
| **Total IMLEO Mass** | | 107,400 | 106,100 |

## *4.3 Mission Risk and Risk Mitigation Strategy*

### 4.3.1 Summary

This initial risk and reliability analysis of candidate Mars architectures does not claim to quantify exact estimates of system reliability; its goal instead is to arrive at reasonable estimates that can be used to identify "differences that make a difference." Over the course of the study, risk analysts worked in conjunction with designers and technical experts to perform system and mission risk analyses. These analyses allowed the decision-makers to discriminate between various decision packages; but more importantly, they provided insights into the impacts that varying combinations of technology risk mitigation techniques would have on mission success.

The focus was to quantify relative comparisons between candidate technology and mission configurations by determining system risk drivers. Once these risk drivers were uncovered, analysts began initial talks with technology experts to determine methods that could be adopted to mitigate the risk impacts of the technologies. Due to the level of uncertainty that is included in many of the proposed technologies and mission configurations, a high degree of uncertainty is apparent in many of the risk estimates. This level of uncertainty is represented using a risk range that represents the potential risk an element would bring to the mission depending upon what investments are made in the way of inherent design reliability, precursor activities, or sparing/modularity capabilities that the technology may have.

The anticipated reliability range of Mars architecture risk driver ranges is shown in figure 4-10. The range of the potential probability of failure is represented below using bars for risk-driving elements. The tic marks on each bar represent the nominal failure probability that is assumed during trade studies. Investments and further analyses in the way of precursor activities, reliability growth implications, sparing/modularity capabilities, and ISS/lunar synergies will determine where the actual element reliability falls within the given range. While these results provide a high-level insight into the reliability story, further analyses are expected to directly tie cost and reliability improvement programs with their risk-mitigation impacts for elements and mission architectures.

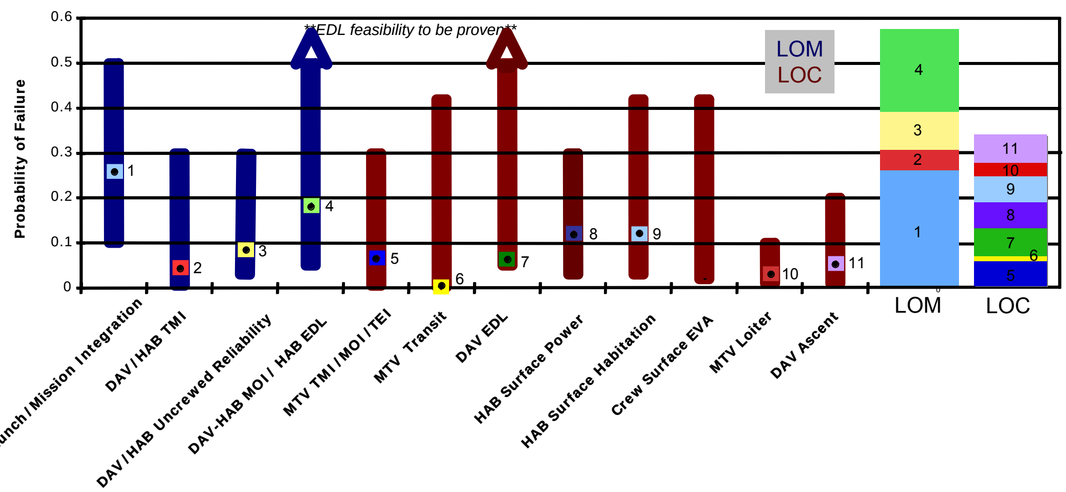



**Figure 4-10.** Mission risk driver ranges.

The top Mars architecture technology risk drivers are described below along with the risk ranges and potential risk mitigation/precursor strategies that are shown in table 4-6. These identified risks must be examined and tracked carefully as the architecture design and development progresses. To reach an acceptable level of risk for the overall Mars architecture, a thorough risk-reduction effort must be made across all technologies. Vigilance will be needed throughout the program to assure that other risks remain low. The bars are roughly arranged in the order of the mission events. A basic feature of the split mission causes the initial failures to be LOM rather than LOC since the crew would not launch if these initial failures occurred.

- ***Launch/mission integration***. The required level of mass to LEO in the necessary launch window makes the launch and integration stage of a Mars mission very difficult. With current ground processing and delay history, the required 10+ launches within the Mars launch window would require investments to lower the probability of failure for the mission. The number of launches and the reliability of the launch vehicle limits the improvement that can be achieved.

- ***TMI/MOI/TEI burns***. The lack of experience that the NTP system has would make it a risk driver for a mission to Mars. Extensive testing and potential lunar or other synergies need to be further analyzed to provide opportunities to mature the propulsion system to an acceptable level.

- ***Mars EDL***. Extreme uncertainty concerning how to design the Mars EDL system makes it a major risk driver for a human Mars mission. The United States has successfully landed five robotic systems on the surface of Mars, all of which had landed a mass that is below 600 kg (0.6 t). A human Mars mission requires a simultaneous two order-of-magnitude increase in landed mass capability, a four order-of-magnitude increase in landed accuracy, and an EDL operations sequence that may need to be completed in a lower-density (higher-surface-elevation) environment.

- ***Crewed/equipment reliability***. The duration of a mission to Mars makes both crewed and uncrewed time on systems a large risk driver. Current technology and design philosophies create an unacceptable level of risk when applied to a Mars mission. With no resupply capability, methodologies concerning sparing, levels of modularity, and scavenging need to be thoroughly explored to design systems that are capable of sustaining a crew for the duration of such an extreme mission. Mission phases will require dramatic improvements in equipment reliability since there is a limit to the mass that is available for redundancy.

**Table 4-6.** Mission Risk Driver Ranges with Potential Risk Mitigation Techniques

| Risk Element | No | Hig | Low | Basis | Mitigation/Precursors |
|---|---|---|---|---|---|
| TMI/MOI/TEI Burns | .07 | .3 | .01 | Existing Study, and Adapted ESAS Maturity Models | •Develop NTP engine and test on Earth before flight test (2–3 demo engines)<br>•Lunar NTP flight test (demo NTP engines for lunar transfer stage)<br>–Dress rehearsal of Mars-type mission with lunar cargo around moon<br>•Any use on high-energy science missions<br>•At least 1/10-scale Mars mission<br>•Full-scale Mars cargo mission |
| Crewed Element Reliability | .12 | .3 | .03 | ISS/Shuttle Equipment Reliability Redundancy Assumptions | •Develop repair concepts<br>•Earth-based technology development and field tests<br>•Operational experience on the lunar surface<br>•Robotic (partial-scale?) demonstration on Mars surface |
| EDL | .25 | .5 | .01 | Notional Concepts, Mars EDL Experience | •Flight tests of TPS entry at Earth<br>•At least 1/10-scale precursor flight at Mars<br>•Full-scale cargo mission at Mars may provide certification for human landing |
| DAV Ascent | .054 | .2 | .01 | ESAS Maturity Models | •Most of the development testing in vacuum and high-altitude chambers on Earth for engines and cryogenic fluid management<br>•Flight test of ascent system in LEO<br>•Common system with lunar lander |
| Surface Operations/EVA | - | .42 | .03 | Not Analyzed yet | |
| Habitat Surface Power | .12 | .2 | .01 | Notional Maturity Estimate | •Earth-based technology development and field tests<br>•Operational experience on the lunar surface<br>•Robotic (partial-scale?) demonstration on Mars surface |
| Elements (repair not an option) | .08 | .43 | .03 | ISS/STS Equipment Reliability Redundancy Assumptions, Notional Improvement Estimates | •Modularization<br>•Reliability improvement programs<br>•Operational experience on ISS<br>•Operational experience on lunar surface<br>•Robotic experience |

### 4.3.2 Methodology

A two-phased, risk modeling approach was used during the Mars Architecture Study. The purpose of this approach is to conduct initial, high-level analyses during which key risk drivers are identified; the second phase included a strategic refinement of risk-critical models. This will ensure that resources are allocated to the most risk-critical architecture elements, focus is maintained, and the study expectations of the stakeholders are verified prior to the bulk of program expenditures.

The first phase of this analysis was based on a variety of techniques that were developed over the past few years. The top-down, scenario-based risk assessment approach that was used by this study is a complex process that incorporates many sources of information to produce a representative analysis. This approach combines modules that represent risk drivers in a transparent fashion so that design teams can easily understand risks and analysts can quickly generate models. An intensive review of heritage information back to Apollo, past risk assessments, and interaction with vehicle designers and operations experts was performed by experienced analysts to identify risk drivers for the proposed Mars missions. The risk drivers of individual mission events were combined into models for the specifics of each mission implementation.

The second phase of the analysis focused on risk mitigation strategies for human exploration of Mars. This analysis included the refinement of leading architectural approaches based on the trade tree and the elimination of options that do not meet risk, cost, or performance specifications. In the event that the calculated risk exceeds the requirements, risk drivers were identified and special studies were initiated to focus on key aspects of leading options to improve the fundamental approach.

Figure 4-11 shows the risk analysis process as it was executed during the Mars Architecture Study.

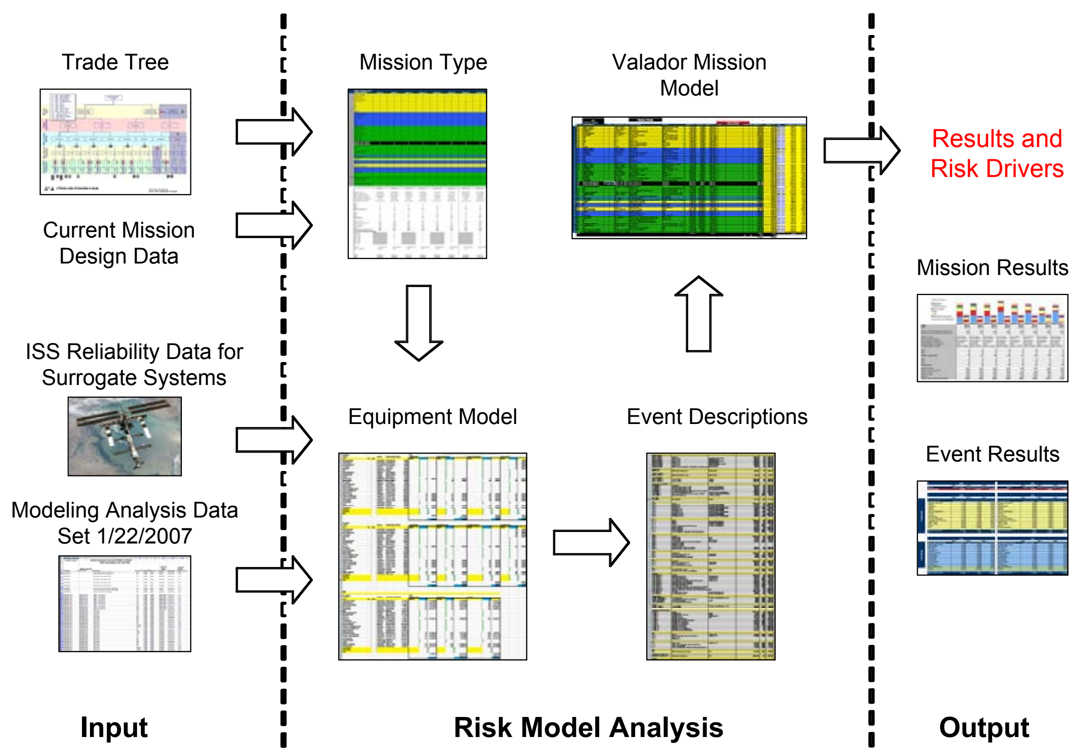



**Figure 4-11.** Risk analyses process flowchart.

### 4.3.3 Risk analyses

The analyses were conducted to identify risk drivers for each mission option as well as to discriminate between major architecture decision points on the trade tree. The associated probability of success of each mission under consideration was represented and quantified as a sequence of events. These events allowed for a modular analysis and were typically partitioned at critical points in the mission profile; i.e., burns, docking, etc. LOC and LOM were then estimated or calculated for each of these events in smaller models or calculations. This modular modeling technique allowed analysts to quickly determine how mission configuration changes would ripple through a given architecture.

Multiple classifications of data were used to quantify each of the mission events. Data classification of events that were used to form the model were quantified from the following data classes (listed by increasing certainty): expert estimates, simulations, PRA-supported calculations, ISS heritage, and shuttle heritage. Many of these data sources admittedly carry with them a high degree of uncertainty; therefore, future work is suggested to refine these values to help in the continued prioritization of a program-level reliability improvement investment portfolio.

#### *4.3.3.1 Expert estimates*

Discussions with subject matter experts allowed for trades to be made based on “order-of-magnitude” PLOM estimates. Many expert judgment estimates result in a range of potential values; for the sake of the trade studies, an anticipated nominal value was selected for the events. Further work will include detailed technology maturity modeling and potential precursor activity impacts.

#### *4.3.3.2 Simulations*

A Monte Carlo simulation that was based on historical shuttle delay data was used to average launch window delays, thereby enabling a PLOM calculation. An orbital flux calculation provided the micrometeoroid and orbital debris (MMOD) risk for various launch configurations. Future work includes discussions with subject matter experts to finalize launch configurations and ground operations procedures.

#### *4.3.3.3 Probabilistic-risk-assessment-supported calculations*

Calculations were made that were based off of existing PRAs and analyses. Some data calculations were made from existing datasets and current MAWG data (i.e., number of chemical propulsive modules). Median of the log-normal distribution was calculated for the NTR burns.

#### *4.3.3.4 Station heritage*

A reliability model was created from ISS equipment data. Future sparing, hardware reliability improvement programs, and precursor risk mitigation are to be considered.

#### *4.3.3.5 Shuttle heritage*

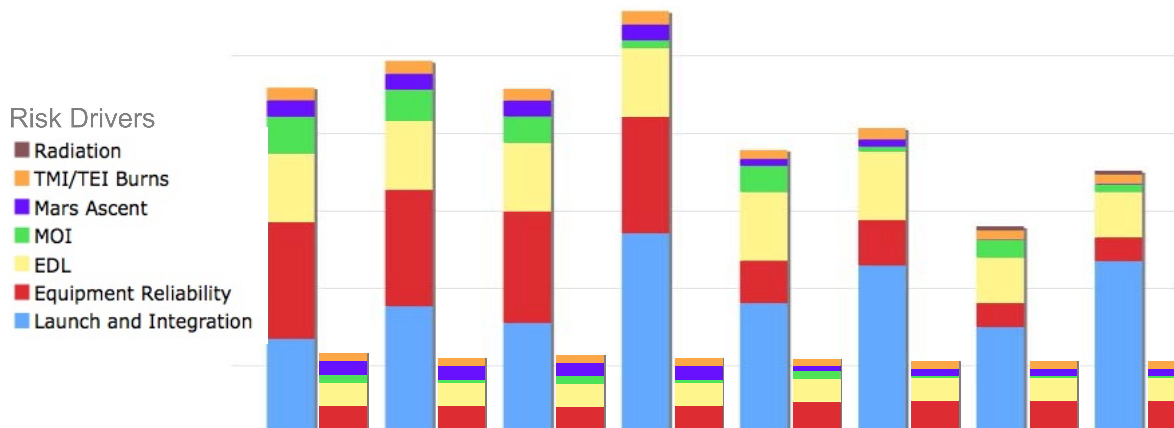
Launch data and test-stand based calculations constitute the portions of shuttle heritage that were used.

From these varying forms of data classification, models were created of various systems and time on those systems was input. The results of these mission analyses are shown in figure 4-12.

### 4.3.4 Conclusions

While the mission risk analyses did produce estimate LOM and LOC values, there were no clear cut decisions that could be made from a risk standpoint for the trade studies conducted. The general story that was constantly reiterated from the individual mission risk analyses was that current design philosophies and technologies would not provide an acceptable level of reliability for a Mars mission.

This insight led to further analysis of the risk driving elements in the design and sensitivities of those elements to risk mitigation techniques that could be applied. Many of the risk mitigation philosophies are somewhat under developed within NASA due to both the shorter duration of ISS and Shuttle manned missions as well as the logistics train available to these systems.




| | Mission 4 | Mission 6 | Mission 10 | Mission 12 | Mission 22 | Mission 24 | Mission 34 | Mission 36 |
|---|---|---|---|---|---|---|---|---|
| **LOM** | **44.89%** | **45.94%** | **45.01%** | **52.63%** | **40.28%** | **41.56%** | **36.04%** | **40.67%** |
| **LOC** | **12.89%** | **8.03%** | **12.57%** | **8.03%** | **12.00%** | **7.44%** | **13.37%** | **9.16%** |
| | | | | | | | | |
| Mission Type | Long Stay | Long Stay | Long Stay | Long Stay | Long Stay | Long Stay | Short Stay | Short Stay |
| Cargo Deployment | Pre-Deploy | Pre-Deploy | Pre-Deploy | Pre-Deploy | All-up | All-up | Pre-Deploy | Pre-Deploy |
| Mars Capture Method | Aerocapture | Aerocapture | Propulsive | Propulsive | Propulsive | Propulsive | Propulsive | Propulsive |
| Mars Ascent Propellant | No-ISRU | No-ISRU | No-ISRU | No-ISRU | No-ISRU | No-ISRU | No-ISRU | No-ISRU |
| Interplanetary Propulsion | NTR | Chemical | NTR | Chemical | NTR | Chemical | NTR | Chemical |
| | | | | | | | | |
| Interplanetary Power | Solar Arrays | Solar Arrays | Solar Arrays | Solar Arrays | Solar Arrays | Solar Arrays | Solar Arrays | Solar Arrays |
| | | | | | | | | |
| HAB | 3 | 3 | 3 | 5 | 7 | 8 | 0 | 0 |
| DAV | 3 | 3 | 3 | 5 | 0 | 0 | 2 | 5 |
| MTV | 3 | 5 | 4 | 5 | 4 | 5 | 8 | 9 |
| **AresV Launches** | **9** | **11** | **10** | **15** | **11** | **13** | **10** | **14** |
| | | | | | | | | |
| HAB | 2 | 2 | 2 | 4 | 6 | 7 | 0 | 0 |
| DAV | 2 | 2 | 2 | 4 | 0 | 0 | 1 | 4 |
| MTV | 2 | 4 | 3 | 4 | 3 | 4 | 7 | 8 |
| **Autodocks** | **6** | **8** | **7** | **12** | **9** | **11** | **8** | **12** |
| | | | | | | | | |
| Cargo Transit | 283 | 283 | 283 | 283 | 180 | 200 | 283 | 283 |
| Ops Pre Crew Launch | 330 | 330 | 330 | 330 | 0 | 0 | 330 | 330 |
| Crew Transit | 200 | 200 | 180 | 200 | 180 | 200 | 189 | 189 |
| Surface Ops | 540 | 540 | 540 | 540 | 540 | 540 | 30 | 30 |
| Return | 200 | 200 | 180 | 200 | 180 | 200 | 328 | 329 |
| **Total Crew Flight Duration** | **940** | **940** | **900** | **940** | **900** | **940** | **547** | **548** |

**Figure 4-12.** Mars architecture risk-driving elements.

### *4.3.4.1 Sparing/modularity/scavenging*

The duration of the Mars mission makes crew and equipment reliability of all systems a large risk contributor, particularly in terms of the criticality and time on the power, thermal, and environmental control life support systems.

An important question arises: namely, at what level do you introduce repair or replacement of the elements of a system? These levels range from raw materials on the low end – i.e., things such as silicon, solder, lubricants, sealing materials, and wire – to actual spacecraft on the high end. The lower the level of modularity, the broader the range of applicability across the design and the greater the ability to scavenge within a design and across elements of a design. What this means is that for a given mass, carrying the reliability or availability for systems that can sustain downtime across the design is improved more the lower the level of modularity. On the other hand, the lower the level of modularity, the more skill and, usually, time are necessary for repair.

The shuttle, and particularly the ISS, have a relatively high level of modularity. The line replaceable units (LRUs) on the shuttle and the orbital replacement units (ORUs) on the station were designed primarily with terrestrial-based maintenance of the lower level assemblies in mind; therefore, unit replacement is possible only on orbit.

Figure 4-13 graphs the sensitivity of hot/cold spares and level of modularity to system reliability. While additional hot spares decrease LOM probabilities, cold spares provide a much greater reliablity improvement. When varying levels of modularity are analyzed (seen here as various numbers of parts), reliability improvement is seen by orders of magnitude depending on the level of modularity and part reliability of the modules.

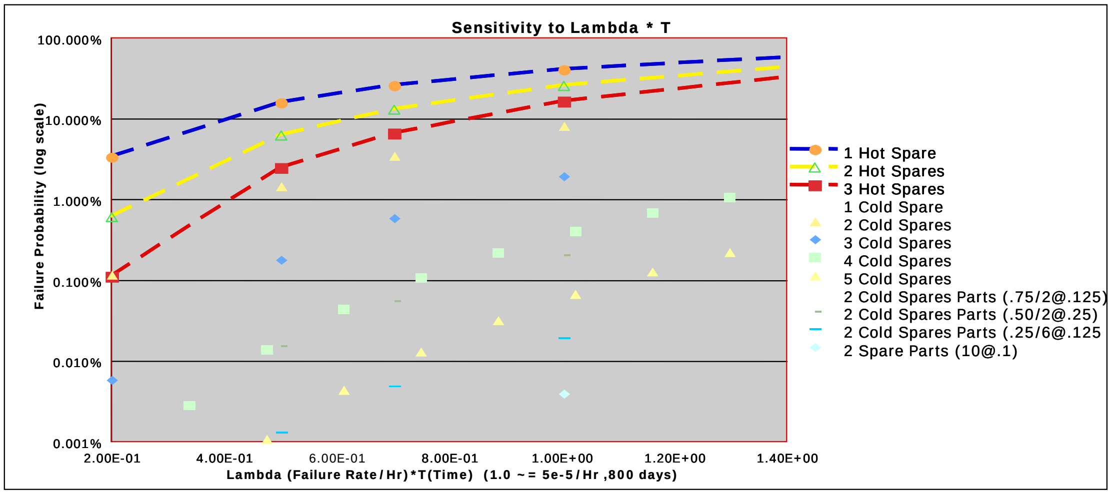



**Figure 4-13.** Sensitivity of sparing and levels of modularity on reliability.

### *4.3.4.2 Precursor activities/reliability growth implications*

The immaturity of the developmental technologies that are needed to complete a Mars mission will have to be addressed through testing or precursor missions. Through demonstrations and experience, a reliability growth will occur as flaws are tested out of the design. The high-risk developmental technologies that were identified for a Mars mission are:

- Mars EDL
- Nuclear propulsion in space
- Surface nuclear power
- Solar array deployment
- $LO_2$ $CH_4$ 40-klb thrust
- Hardware activities

Due to time constraints, a maturity model that was designed to model propulsion systems and developed during the ESAS was used to model all of these developmental technologies. The reliability curves that are in figure 4-14 show the experience that is necessary to mature these technologies across their reliability ranges. These curves correspond with the bars that were used to represent a range of anticipated reliabilities for each technology.

### *4.3.4.3 Maximize lunar base and International Space Station synergies*

Developing an architecture that can take advantage of all possible lunar architecture and ISS technology synergies is critical to achieving acceptable reliability metrics. Taking advantage of precursor tests that can be carried out as a part of the lunar mission architecture or that of the ISS will offer invaluable reliability growth for Mars technologies. Potential technologies that can benefit form a test program during lunar or ISS missions include but are not limited to:

- Nuclear surface power
- Nuclear propulsion
- Lander propulsion
- ECLSS
- Reliability programs

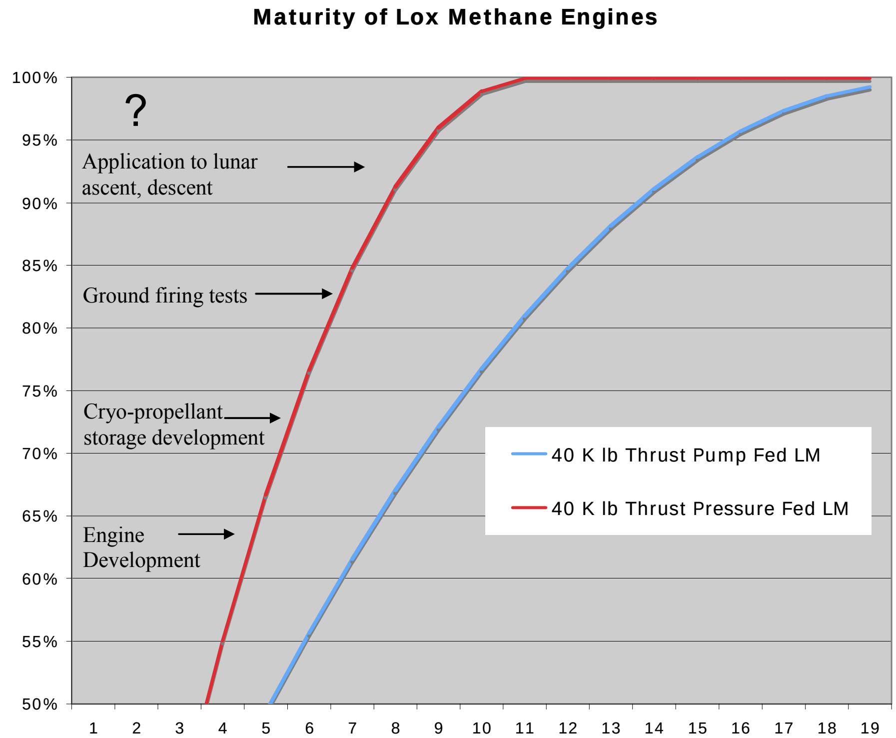



**Figure 4-14.** Developmental technology reliability growth curves.

### 4.3.5 Future work

This initial risk analysis has highlighted many areas that would benefit from additional more-detailed analyses. Continued refinement of the risk driver calculations can be made as systems details become more comprehensive. As these drivers are continually refined, decision-makers should maintain a prioritized reliability improvement investment portfolio.

The maturity model that was used during this analysis to model the reliability growth of developmental technology needs to be adjusted for each technology through discussions with experts. The initial model was developed to model propulsion systems and may differ in character for varying technologies. Once these reliability curves are modified for each technology, costing and precursors should be performed based on overall risk buy-down for the mission architecture. A detailed maturity growth model would allow potential risk buy-down to be quantified for each technology, and provide decision-makers with the information that is required to allocate resources to make the most impact on the probability of mission success.

Philosophies concerning sparing, degrees of modularity, and scavenging capability need to be continually studied and considered by integrated analysis teams. To achieve an acceptable level of reliability, these capabilities need to be designed into the systems from the beginning.

## *4.4 Flight Crew*

The following discussion on flight crew was generated as part of Mars Design Reference Mission 1.0 and is provided here for completeness.

### 4.4.1 The role of the flight crew

Humans are the most valuable mission asset for Mars exploration and must not become a weak link. The objective for humans to spend as many as 600 days on the martian surface places unprecedented requirements on the people who will be chosen for these crews and their supporting systems. Once committed to the mission on launch from LEO, the crew must be prepared to complete the full mission without further resupply from Earth. Unlimited resources cannot be provided within the constraints of budgets and mission performance. The resources for the crews would either be with them or would have already been delivered to or produced on Mars. Trade-offs must therefore be made between cost and comfort as well as performance and risk. Crew member self-sufficiency is required because the long duration of the mission and the fact that the crew's distance from Earth impedes or makes impossible the traditional level of communications and support by controllers on Earth. The crews will need their own skills and training and specialized support systems to meet the new challenges of the missions.

The number of crew members to be taken to Mars is an extremely important parameter for system design because the scale of the habitats, space transportation system, and other systems supporting the mission are directly related to the number of crew members. This, in turn, will have a direct relationship to the cost of the first missions. The size of the crew also is probably inversely proportional to the amount of new technology that must be developed to allow all of the tasks to be performed. Because of communication time delays between Earth and Mars, some functions that have previously been performed by people on Earth would be carried out autonomously or by crew members. Generally, a high degree of automation would be required for routine operations on the Mars journey to allow crew members to do specialized tasks.

The number of crew members needed to successfully complete a mission to Mars has been a topic of debate since the earliest serious studies of this endeavor. In 1948, 60 years prior to the current study and in a year when our knowledge of Mars was much more limited, Dr. Wernher von Braun and a group of scientists and engineers put together a plan that would have sent 70 people to Mars, 50 of whom would explore the surface, for the first human expedition. As our understanding of this planet has matured and as technology has advanced, the number of crew members that would be needed for a successful first mission has steadily decreased; and, at present, it is typically assumed that a crew size of fewer than 10 people is reasonable. How many fewer than 10 people is reasonable is still a matter of analysis and debate. Making a final determination on this topic will depend on the specific objectives that are set for the crew, the concept of operation that is defined for the crew to accomplish these objectives, and the type of contingencies with which the crew would be expected to deal. Two approaches to analyzing this crew size question, a top-down functional assessment and a bottom-up historical assessment, illustrate the range of approaches that is used to understand this issue sufficiently to reach a conclusion.

### 4.4.2 Top-down functional assessment of crew size

The objectives of the missions are to learn about Mars and its capability to support humans in the future; there therefore would be a minimum level of accomplishment below which a viable program is not possible. Survival of humans on the trip to Mars and back is not a sufficient program objective.

Crew members should be selected who would agree to conduct operational research willingly and openly, and who can relate their experiences back to Earth in an articulate and interesting manner. They should also be given enough free time to appreciate the experience and the opportunity to be the first explorers of another planet.

For the Reference Architecture study, it was assumed that crew health and safety are the first priority in successfully achieving mission objectives; and that the surface system design requirements for operability, self-monitoring, maintenance, and repair would be consistent with the identified minimum number of crew members. The crew size and composition was determined in a top-down manner (objectives → functions → skills → number of crew members + system requirements) as the systems have not been defined in a bottom-up manner based on an operational analysis of the system.

From previous studies, workload analysis assumed that the crew would spend available time in either scientific endeavors or habitation-related tasks. From that analysis, the lists of required skills were developed. Expertise is required in the following three principal areas:

- *Command, control, and vehicle and facility operations functions*. These functions include command, management, and routine and contingency operations (piloting and navigation, system operations, housekeeping, maintenance, and repair of systems). Maintenance must be accomplished for facility systems, human support systems (medical facilities, exercise equipment, etc.), EVA systems, and science equipment.

- *Scientific exploration and analysis*. This area includes field and laboratory tasks in geology, geochemistry, paleontology, or other disciplines that are associated with answering the principal scientific questions.
- *Habitability tasks*. These tasks include providing medical support; operating the bioregenerative life support system experiment; performing biological, botanical, agronomy, and ecology investigations; and conducting other experiments that are directed at the long-term viability of human settlements on Mars.

If each skill is represented by one crew member, the crew size would be too large. Personnel would have to be trained or provided the tools to perform tasks that are not their specialty. Special-skill requirements appear to be in the areas of medicine, engineering, and geoscience.

- *Medical treatment*. In a 3-year mission, it is very likely that an accident or disease would occur. At least one medically trained person would be required as well as a backup who is capable of conducting procedures under the direction of medical experts on Earth (through telemedicine).
- *Engineer or technician*. A person who is skilled in diagnosing, maintaining, and repairing mechanical and electrical equipment would be essential. A high degree of system autonomy, self-diagnosis, and self-repair is assumed for electronic systems; however, the skill to identify and fix problems, in conjunction with expert personnel on Earth, has been repeatedly demonstrated to be essential for space missions.
- *Geologist-biologist*. A skilled field observer-geologist-biologist is essential to manage the bioregenerative life support system experiment. All crew members should be trained observers, highly knowledgeable of the mission science objectives, and able to contribute to the mission science. Other factors would also contribute to the final determination of crew size: system autonomy, simultaneous operations, contingency situations, human factors, and international participation.

Electronic and mechanical equipment must be highly autonomous, self- or crew-maintained, and possibly self-repairing. The amount of time that is taken to do routine operations must be minimized through system design. In principle, the operation of supporting systems (e.g., power, life support, in-situ resource recovery) should be transparent to the crew. The best approach in this area is to define the requirement for technological development based on the mission requirements for a given crew size.

Simultaneous operations would be required during the nominal mission. All crew members would be fully occupied during their assigned working hours, and a minimum number of crew members would be required by the distribution of tasks. For example, EVAs are likely to require at least two people outside the habitat at any one time to assist each other. A third person is likely to be required inside to monitor the EVAs and assist if necessary. If other tasks (repair, science, bioregenerative life support system operation) are required to be done simultaneously, the number of crew members may need to be increased.

Specific contingency situations and mission rules have not been established for the Reference Architecture because it is too early in the design phase. However, the choice of what the crew would be allowed to do or not do can impact the size of the crew. For example, during exploration campaigns, mission rules may require that some portion of the crew be left in the main habitat while the remainder of the crew is exploring in the mobile unit. It would be necessary to have a backup crew to operate a rescue vehicle in the event the mobile unit has a problem. If the exploration crew requires three people, the requirement to have one driver for a backup unit and one person left at the outpost implies a crew of not less than five.

In terms of human factors considerations, the psychological adjustment is more favorable in larger crews of six to eight than in smaller crews of three to five. However, the psychological environment may be met by system and support provisions rather than by the crew size itself.

It is conceivable that each country that makes a major contribution to an international Mars exploration mission would demand representation on the crew. Currently, a Mars crew might be patterned after the ISS with representatives from the United States, Russia, ESA nations, and Japan. However, in an enterprise of this magnitude, Third World representatives might also be selected by the United Nations.

Taking these factors into account at a summary level, the five most relevant technical fields that are necessitated by exploration and habitation requirements include mechanical engineer, electrical and electronics engineer, geologist, life scientist, and physician-psychologist. These fields should be represented by a specialist, with at least one other crew member who is cross-trained as a backup. Crew members would also be cross-trained for the responsibilities of a wide variety of support tasks as well as tasks of command and communications. The result of the workload analysis indicates that the surface mission can be conducted with a minimum crew of five, based on the technical skills that are required. However, loss or incapacitation of one or more crew members could jeopardize mission success. Therefore, a larger crew may be required to address the risk issues.

### 4.4.3 Bottom-up historical assessment of crew size

The flight crew size for the Mars DRA is currently baselined as six crew members. As part of the continuing evolution of the DRA, and in an attempt to reduce the overall IMLEO and costs that are associated with the mission, consideration must be given to reviewing the number of crew members that are required for this mission. The base assumption for the current crew of six is to limit each person to two specialized tasks. The following will summarize relevant shuttle, *Mir*, and ISS experience, as well as propose a new DRA assumption that is based around a reduced crew size.

#### *4.4.3.1 Shuttle*

Although early shuttle flights were accomplished with two crew members, the standard space shuttle crew is five. This is generally expanded to seven during research flights. The crew includes a commander and a pilot, and three or more mission specialists. The commander and pilot support ascent, abort, and entry flying tasks, among other duties. The mission specialist-2 serves as a “flight engineer” during dynamic flight phases, assisting the commander and pilot in systems and procedure monitoring. Mission specialist-1 sits with mission specialist-2 on the flight deck during dynamic phases, but has a reduced workload as compared to the mission specialist-2. Other mission specialists sit on the mid-deck during these phases, and are responsible for assisting other crew members with egress or bailout in a contingency. During orbit phases, the general categories of crew responsibilities include: portable computers, Earth observation, EVA, medical, in-flight maintenance (IFM), crew equipment, photography, payloads, development test objectives/ detailed supplementary objectives (DTOs/DSOs), and phase support (rendezvous, prime payload deploys, etc.).

On several past missions when a series of EVAs was included in the flight plan, two pair of EVA crews were assigned. In all cases, whenever an EVA was executed, at least one crew member who is not going outside is assigned as an intravehicular (IV) crew member supporting the crew members who are outside. The IV crew member keeps track of the timeline, consumables, tools, and tasks. These are items that would be difficult and potentially dangerous for the extravehicular (EV) crew members to track on their own. Only in rare and extreme contingency situations would an EVA be performed with only one EV crew member; the “buddy system” is critical to success and safety in performing an EVA. In each grouping of tasks, a prime and backup EV is assigned and trained.

To support the mission payloads, a payload commander is assigned overall payload responsibility. Generally, each payload has a prime and a backup person to work them unless the number of experiment runs or subjects requires more than two people.

#### *4.4.3.2 Mir*

Experience from the Space Shuttle/*Mir* Phase One missions has shown that the standard *Mir* crew of three is roughly along the commander, pilot, and flight engineer arrangement that was described above. For a number of years, *Mir* was nominally flown with a crew of only two. EVAs were performed in a manner similar to the Space Shuttle Program, with two EV crew members and the third crew member assisting from inside the *Mir* or the Soyuz. Obviously during crew changeout, or during the docked phase of a space shuttle re-supply mission, the overall crew size jumped dramatically to as many as 10 people overall. Transferring equipment and supplies between the vehicles proved to be challenging with this number of people. However, this task was streamlined in later missions.

#### *4.4.3.3 International Space Station*

The needed ISS crew size will change as it continues to be assembled. Crew size has thus far been defined by the need for assured crew return capability, which has been provided by the Soyuz spacecraft and, thus, constrained to three people. With the recent addition of the ESA Columbus module and Japan Aerospace Exploration Agency (JAXA) Kibo module, the crew will be expanded at least six to make full use of the station and its research capabilities. Assured crew return will be expanded to accommodate this crew size.

EVAs are carried out on station by a crew of two with the third crew member functioning much as the shuttle IV crew member functions: operating the station robotic arm and keeping track of the timeline, consumables, tools, and tasks. These EVAs are staged from the station's own airlock.

In all of these cases, the missions have been controlled and supported by an extensive ground component. Details of this ground element will be discussed in the next section.

In summary, the standard shuttle crew is five. The standard *Mir* crew was initially two and grew to three; and the standard ISS crew will be three during most of the assembly, growing to six or possibly seven at assembly complete.

These near-Earth missions, which represent the bulk of human space flight experience, have, by their nature, simplified crew tasks (communication is nearly continuous and instantaneous, re-supply is generally not more than 1 month or so away, the ground can perform a greater degree of systems monitoring/trend analysis, an emergency de-orbit can have the crew back on the ground in relatively short order, etc.). To determine the size of the DRA crew, consideration must be given to the tasks that they would need to accomplish as well as to the support that they would be given by both Earth-based resources (a Mission Control Center (MCC)) and on-board resources (automation).

Two factors will drive technology development requirements for exploration missions beyond LEO: (1) the communications delay due to distance, and (2) the need for more capable on-board monitoring and analysis to augment the inability of the ground to provide real-time systems management and oversight. Both of these factors have the potential of reducing the steady-state crew workload while increasing the criticality of effective redundancy management. They also result in an increased need for the crew to be able to perform a variety of in-situ repairs and maintenance tasks.

Additionally, due to the extended micro-g cruise between planets, it is expected that the majority of the manual flying tasks for Mars and Earth entry would be automated, with the capability for manual controls as a backup only. All of these factors (higher overall levels of automation, independence due to communications delays, etc.) generally support having exploration missions with smaller, or at least comparable, crews that have been typically used in LEO.

Taking these considerations into account, the following proposed crew specialty and task distribution is provided. It goes without saying that each crew member, while specializing in one area of expertise, must be competent in the other major areas, so that each of the crew members can provide effective backup to the critical skills that would be required of the entire crew in general.

- *Mission commander and vehicle systems specialist/engineer*: Responsible for overall on-board operations, safety, and mission success. Expert in the vehicle systems, redundancy management, and crew support of mission requirements. Will provide backup to the technical specialist for IFM tasks and troubleshooting.
- *Medical doctor*: Self-explanatory for long-duration missions. At least emergency medical technician (EMT)-level training would be likely for all crew members as a backup. Biology background would also back up the science crew.
- *Geologist/biologist/meteorologist/planetary scientist*: While each crew member would be trained and proficient in geology, at least one crew member should be a professionally trained geologist who is experienced in "expedition" research. This capability would be leveraged to help ground planners choose exploration targets and priorities. Heavy emphasis on biology and meteorology would be necessary to assist with mission/science planning and as a backup to the medical doctor.
- *Technical specialist/assistant geologist*: This person would be an expert at IFM tasks, troubleshooting problems, and fabricating parts, while also having a substantial geology background. This would provide backup to both the systems expertise of the mission commander and the exploration background of the planetary scientist.

These four individuals would form two EVA teams for surface exploration. While one pair is outside, the second pair would remain inside, with one individual performing IV support and the other monitoring vehicle health/status. With respect to ground support, the concept of "real time" would actually be as much as 40 minutes old. Hence, all of the current oversight and monitoring tasks that the MCC traditionally performs during an EVA would have to be handled autonomously or done by the IV crew member. With sufficient time off to prevent exhaustion and burnout, these two pair of EV teams would perform exploration EVAs on alternating days within the crew scheduling constraints and as the mission schedule demands.

Reducing the crew size from six to four offers a substantial reduction in vehicle size, pressurized volume (at least 180 $m^3$ of volume using current planning), and life support infrastructure. This also translates to a reduced TMI propellant load, a reduced ISRU demand while on Mars, and a smaller vehicle for Earth return. While an even number of crew members does not strictly follow the "authority pyramid" concept of having an odd number of crew members with one obvious leader, the concept does clearly respect the need for a mission commander. The individuals who are selected for this mission would be highly screened against a variety of criteria, including compatibility. Therefore the smaller, albeit even, number of crew members would be sufficient from an operations perspective.

It is shown that this reduction is consistent with past LEO experience, and does not incur a substantial increase in exploration/EVA risk to the crews, whether they are assuming EV or IV duties. By carefully selecting crew members based on past training and/or professional experience that is consistent with the guidelines listed above, each individual can provide backup to the others while specializing in his/her own area(s) of expertise.

It should be noted that while a reduction to four crew members appears operationally sufficient, it may not be optimal for mission success. For instance, the assumptions that are outlined above do not presuppose the ability to sustain the incapacitation of a single crew member for a prolonged period of time. PRA has shown that for the duration of the proposed mission, at least one crew member would sustain a serious injury or illness. Despite an attempt at redundant crew training, depending on the affected crew member the level of either scientific return or vehicle/ systems capability may be reduced; thus affecting overall mission success.

Additionally, the areas of expertise that are outlined above are fairly ambitious. For example, a high level of in-depth knowledge, spread across several, unrelated scientific fields, would likely be required of the planetary scientist on a mission. While it is possible for one crew member to have a working-level knowledge of several scientific disciplines, the in-depth knowledge that is required for successful planetary exploration may make it more prudent to have an additional, dedicated crew member for this task.

The above proposal also identifies two EVA teams of two crew members each. For scenarios where a remote EVA is in progress during the same timeframe in which a local EVA is required to maintain a vehicle or system, no crew members are left to perform the required IV tasks for either EVA team. The same situation results for contingency EVA rescue operations.

In summary, the proposed crew size in the DRA is well within human space flight experience. A reduction from six crew members can offer substantial savings in, among other areas, mass and volume, thus reducing overall mission cost. A crew size of four is considered operationally sufficient. However, to realistically determine the crew size, a trade must be made between the cost that is associated with one additional crew member and the level of risk that is considered acceptable to achieve mission success.

#### 4.4.4 Design Reference Architecture assumed flight crew

Past studies such as these have examined the size and makeup of the crew that would be needed to meet both operational needs and mission objectives of the first human Mars missions. The results of these studies arrived at the following general conclusions:

- Skill mix requirements indicate the need for a crew of five
- Peak workload indicates the need for a crew of six (three at the base and three in the field)
- Requirements for margin suggests the need for a crew of seven or eight

While no final conclusion has been reached regarding the required number of crew, recent studies have tended to assume a crew of six. The specific skill mix for this crew also continues to be analyzed and will be dependent on needs driven by the objectives that are set for this crew. Since we are lacking a specific set of objectives, the following functional breakdown of crew specialization (although cross training is assumed) is representative of mission needs:

- Pilot
- Physician
- Geological scientist
- Biological scientist
- Mechanical engineer
- Electrical/electronics engineer

This crew is also assumed to be of mixed gender.

## *4.5 Mission Operations*

Mission operations for Mars missions will continue to evolve as it has done from the Apollo Program to the Space Shuttle Program to the International Space Station Program: retaining what has proven successful for missions in the space environment, and adding or modifying the capabilities that are needed to operate on this planetary surface. This section will describe key features of mission operations that have been developed and accepted as being representative given the current level of definition for Mars missions.

### 4.5.1 Mission Control Center

An MCC would be established to support all phases of the Mars mission. This MCC would be fully staffed only during those periods of high activity that would be specified in the mission plan. During quiescent periods, which would include most of the transit to Mars, the MCC would transition to providing planning and information support, and would operate with reduced staffing.

The long duration and communications delay of human exploration missions renders continuous MCC staffing both impractical and unnecessary. Fully staffed periods would include activities such as final planning and execution of mid-course maneuvers, major changes to the spacecraft configuration, contingency EVAs, etc. Routine systems management actions – e.g., cabin atmosphere adjustments, thermal load monitoring, thermal control of temperature sensitive systems, radiation, solar event monitoring, consumables management and tracking, etc. – would be completely automated with the capability provided for crew monitoring and intervention, if necessary. Skill distribution and requirements for Flight Control Team (FCT) members are still TBD at this time

### 4.5.2 Automation and maintenance

Automating routine habitability tasks, while still allowing for crew intervention, will be a high-priority development capability for all of these systems to allow a reduced crew workload. This would free up crew time for higher-priority tasks while yet retaining the ability to control systems as needed in the event of problems.

### 4.5.3 Redundancy management

Redundancy management (RM) will be employed in the selection of backups to replace failed or degraded systems, or to manage the rotation of redundant systems to equalize hours of operation. Some systems will have one or more identical backup units, ensuring physical redundancy. Other systems, for which there are no physically identical replacements, may have their functions assumed by nonidentical systems, ensuring functional redundancy.

Selection of a backup system upon failure of the designated primary system will be automated and controlled using rules-based logic to avoid failure propagation. This will minimize interruptions to normal operations and reduce risk to the crew. All such automatic reconfiguration actions will be annunciated to the crew members so they may maintain situational awareness.

Scheduled rotation of redundant systems will be automated to reduce crew workload, and all such reconfigurations will be reported to crew members for their situational awareness. Automating scheduled rotation of redundant systems prevents interrupting other work. The crew will have the capability to select between redundant systems that are operating normally based on engineering judgment. For example, a system that has not failed but that is displaying signs of a possible incipient failure or other indications of degraded performance may be taken offline for maintenance. This may prevent possible damage to an otherwise salvageable component, or it may preclude a more serious service interruption. The RM requirements and scenarios will be elaborated on as the vehicle designs mature.

### 4.5.4 Mission planning

During human exploration missions, it will be important for mission planners to achieve the right balance between disciplined scheduling and necessary flexibility. People work best within some level of established routine, but the length of Mars missions precludes planning the activities of each crew member to the level of detail that is seen with short-duration missions in LEO. The length of a nominal crew day, and the amount of time that is allocated to broad categories of activities (i.e., sleep, post sleep, mission support operations, exercise, rest and relaxation (R&R), etc.) can be established. Exactly how each activity fits into that template and, to some extent, who performs which actions and at what time will largely be left to the crew, however.

Some activities, such as orbital maneuvers or solar array deployments, are dictated by physics and must be executed within a short window of time. Other activities, such as crew exercise periods, etc., must be done faithfully every day, but can be placed in the daily schedule at the discretion of the crew. Small science experiments might allow for even more flexible scheduling; e.g., the crew could place them in a weekly or monthly timeline. The crew will require the software to make complex scheduling possible. Given certain constraints, such as the timing of inflexible tasks, priorities of tasks, typical time it takes to do a task, mission rules, skills of the crew, and a list of tasks remaining, the software must fit the activities into the crew day, week, and month without violating mission rules or scheduling incompatible tasks over one another.

A representative example of a "standard work day" for the crew would be similar to the following:

1.0 hour of post-sleep activities
3.0 hours for meals
0.5 hour for uplink message review
6.5 hours for mission operations support
2.0 hours for exercise (includes setup and teardown)
1.0 hour for report preparation and planning
2.0 hours pre-sleep activities
8.0 hours sleep

In this example, the pre-sleep, sleep, and post-sleep periods are considered off-duty time for the crew with no scheduled activities. The sleep period may be shortened or the work/rest cycle shifted to accommodate mission-critical events. Not covered above is time for crew members to prepare and review personal messages, which will usually occur during their daily or weekly off-duty periods.

For Mars surface operations, the daily time allocations for crew activities during the workday will generally be the same as listed in section 6.1.2. The total workday length will be adjusted by adding further off-duty time to accommodate a martian day, or sol, which is 24 hours 39 minutes long. Surface exercise periods will be equivalent to 2 hours a day, 6 days per week, minus the exercise equivalent of the activities that are performed during EVA by that individual. **Note:** Unless otherwise specified, all subsequent references to a martian day in the context of surface operations will be referred to as a sol.

A standard crew work week will be used for all mission operations and will consist of the following:

- Five days for mission-support activities
- One-half day for habitability activity and maintenance
- One-and-a half-day off-duty time

One-half day may be dedicated to habitability operations and/or routine maintenance. Crew exercise will be planned for 6 days a week (Monday through Saturday). Occasional mission-critical activities may be scheduled for off-duty days.
The nominal short-term (less than 7 sols in the future) planning will be done by the crew, consistent with recommendations and guidelines provided from the MCC. Longer-term planning will be primarily performed by the MCC.

### 4.5.5 Integrated vehicle health monitoring

Due to the communications delays and bandwidth limitations, the mission elements will need to be as autonomous as possible. For this reason, it is assumed that fully Integrated Vehicle Health Monitoring will be designed into all of these elements.

The objective of automated integrated vehicle health monitoring (IVHM) is to maintain a vehicle as nearly as possible in a nominal state. When an event occurs that perturbs the nominal state, IVHM has the intelligence to recognize the change and take action to reconfigure the system to a state that is functionally nominal, or as close to the ideal, as possible. On-board systems will thus be self-diagnostic. Health monitoring and fault detection of on-board systems will be automated, and systems data will be available to the crew via interactive displays.

Complete systems status information must also be available to the crew members, because they will be responsible for all near-term systems management decisions. Failures or deviations from acceptable tolerances in critical systems must be annunciated via audible and visible alarms to alert the crew to the need for immediate action. Failures in critical systems will automatically queue displays of the problem description and corrective action instructions, and will initiate those systems configuration and/or safing actions that are deemed time-critical.

### 4.5.6 Information systems

By current spacecraft standards, the on-board computing requirements for human Mars missions will be enormous. These missions will rely heavily on automation for vehicle systems monitoring, redundancy management, and even some level of corrective action. Advanced command and control systems and information display technologies, such as portable/wearable terminals or virtual-reality helmets, glass cockpits, or projection dome command centers, will be required for human Mars missions. Crew training, especially (Earth- and space-based) simulator requirements, will place a load on computer systems. Extensive flight documentation and procedures will be stored electronically. Large volumes of science and engineering data will be stored before being transmitted to the MCC. With autonomous navigation and maneuver planning, the purely computational demands for on-board computer systems will be greater than ever before. The new role of on-board real-time mission planning and re-planning will rely on on-board computing systems. Information systems must also be physically robust to endure the rigors of space flight (long-term, nearly continuous operation, high-Gs and vibration, sudden accelerations, micro-gs, high-energy-charged particles, cosmic rays, etc.). Finally, for crew safety, a high level of redundancy and software/data backups will be required.

The software for detailed analysis and sophisticated graphical display of the behavior of system parameters for long-term trend analysis will reside with the MCC, not on board. Long-term trend analysis and its associated tools remain a primary responsibility of the MCC for exploration missions.

## *4.6 Communication and Navigation*

### 4.6.1 Introduction

The maximum distance between Earth and Mars is roughly 400,000,000 km, or 1,000 times the distance between Earth and the moon. Because received signal strength scales as the inverse of range squared, communications from Mars are effectively 1,000,000 times more challenging than communications from the moon. The large distance to Mars also implies long signal transit times, with round-trip light times of up to 44 minutes. This, too, has a profound effect on the basic operations concepts for Mars exploration relative to lunar exploration.

So, while the lunar communications architecture provides important feed-forward concepts and capabilities, the strategies for Mars will require significant tailoring to address the much larger distances and light times that are involved. Like the lunar architecture, the Mars architecture includes a combination of ground stations on Earth and relay orbiters at Mars that will provide communications and navigation services to exploration users. Details of the design, however, particularly for the long-haul links between Earth and Mars, must be significantly modified. Nevertheless, common solutions will be sought wherever possible, including aspects of the short-range relay links as well as the upper layers of the communications protocol stack, above the physical and link layers that are driven by distance, which can remain essentially the same as in the lunar architecture.

#### 4.6.2 Concept of operations

The overall NASA communications and navigation architecture must support the full scope of Mars exploration, including launch, Earth orbital operations, TMI, Earth-Mars cruise, MOI, Mars orbital operations, EDL, surface operations, Mars ascent, on-orbit rendezvous, TMI, and Earth arrival. Meeting this range of mission phases will require the combined capabilities of the Space Network (for initial near-Earth support), the Deep Space Network (DSN), and a dedicated Mars Network (MN) assets. We summarize here some of the driving operations concepts across these mission phases, which establish a basis for the overall Communications and Navigation System requirements.

From launch through TMI, the NASA Space Network will provide continuous communications and navigation services via Tracking and Data Relay Satellite System (TDRSS) S-band and Ka-band links. After TMI, support during Earth-Mars cruise will transition to the NASA DSN, with basic telemetry, tracking, and communications (TT&C) functions provided at X-band and high-rate links supported at Ka-band. A particular challenge during cruise will be ensuring adequate safe mode communications in the event of an anomaly.[2]

On approach to Mars, DSN-based tracking, including range, Doppler, and radio interferometric techniques, provides an accurate trajectory, one that is potentially augmented with on-board optical imaging and/or radio tracking on links to previously deployed Mars orbital or surface assets for improved Mars-relative state knowledge.

Once in the Mars environment (on final approach, in orbit, in the atmosphere, or on the surface), a user spacecraft will be able to obtain efficient, high-rate communications and tracking services from on-orbit Mars relay satellites (MRSs). The MRSs will be outfitted with highly capable, direct-to-Earth communications payloads to support high rates on the trunk line back to Earth, allowing individual users to use much smaller, lighter, and lower-power communications systems on the relatively short-range links to the MRSs. In addition to providing an energy-efficient means for communications between a Mars user and the Earth, the MRSs will also play a key role in supporting communications between spatially separated users at Mars (e.g., between a Mars habitat and an astronaut on a long-range, over-the-horizon excursion). For users in the immediate line-of-sight vicinity of the Mars habitat, a Mars communications terminal on the surface will also provide even more efficient wired and wireless communications options over short-range links.

The MRSs will provide important telemetry coverage during critical mission events such as MOI or EDL. In addition, the orbital relay will provide tracking and navigation to support on-orbit rendezvous operations after launch of the crew from the martian surface.

#### 4.6.3 Strawman communication and navigation requirements

While detailed communications requirements for Mars exploration are not yet well-understood, the feed-forward nature of lunar exploration as a precursor to Mars exploration, within the Vision for Space Exploration, offers strong motivation for providing communications and navigation capabilities at Mars that are comparable to those that will be used at the moon, thereby supporting similar exploration operations concepts. Thus, as a starting point

[2] Human missions have typically counted on the ability to support voice links from any attitude to recover from spacecraft anomalies, such as the uncontrolled spin that was experienced during the Gemini 8 mission. For that reason, the CxP currently requires analog single side band voice communications capability from any attitude, which is accomplished through an omnidirectional antenna. While this suffices for Earth orbit or lunar missions, such an approach falls more than 40 dB short for the long distances that are experienced during an Earth-Mars cruise. A more appropriate design would use efficient digital modulation with a 2-kbps digitized voice signal, high spacecraft effective isotropic radiated power (EIRP) that is based on a 1-kW omni transmitter, and large ground aperture equivalent to four 70-m antennas. Alternatively, human missions may use “smart” antenna concepts to achieve increased spacecraft antenna gain, based on inertial sensors and/or an uplink beacon from Earth, to obtain voice-capable safe mode downlinks. In any event, safe mode communications during cruise will represent a potential driving telecommunications scenario.

for this study, we have adopted the data rate requirements emerging from the recent LAT study. (Schier, et al., 2007[3]).

The LAT generated a telecommunications data traffic model that was based on the presumed data requirements of individual exploration elements. Aggregate data rates for high-rate data are dominated by assumptions about high-definition video and high-resolution instruments. Table 4-7 shows aggregate peak rates without margin from the LAT model. A more detailed breakdown of surface element links for a lunar mission is shown in table 4-8.

**Table 4-7.** Aggregate Lunar Mission Data Rates

| Description | Applicable System(s) | Data Rates (Mbps) |
|---|---|---|
| Aggregate Peak Rate to Earth | LRS* and Earth Ground System | 154.9 |
| Aggregate Peak Rate from Earth | LRS and Earth Ground System | 67.1 |
| Aggregate Relay from Lunar Surface to Lunar Relay Satellite | LRS and LCT** | 222.4 |
| Aggregate Peak Rate from Lunar Relay Satellite to Lunar Surface | LRS and LCT | 147.1 |
| Aggregate Peak Rate Across Lunar Surface | LCT | 151.7 |

*LRS = lunar relay satellite; LCT = lunar communications terminal

**Table 4-8.** Lunar Mission Surface Element Links

| Transmission Source | Low Rate | | High Rate | |
|---|---|---|---|---|
| | Sources | Data Rate | Sources | Data Rate |
| LSAM | 1 | 0.592 | 1 | 22 |
| EVA suit | 4 | 0.002 | | |
| Rover | 2 | 1.75 | 2 | 20.5 |
| Surface Mobility Carrier | 2 | 0.25 | 2 | 4.5 |
| $O_2$ Excavator | 1 | 0.25 | 1 | 9.5 |
| $O_2$ Mobile Servicer | 1 | 0.25 | 1 | 11 |
| Habitat | 1 | 2.824 | 1 | 135 |
| **TOTAL** | **12** | **7.9** | **8** | **203** |
| **Ranging links** | **13** | | | |

These links would be made at the S-band and Ka-band on the moon (for operational and high-rate links, respectively); similar links would be at the X-band and Ka-band at Mars. Also, for the moon these links could be either direct to Earth (DTE) or relayed through a relay orbiter; at Mars, these data rates would only be achievable on the relay links, not on DTE links, due to the much larger Earth-Mars distance.

The lunar architecture envisions a fixed LCT that would aggregate communications from many local users into a single trunk line. We contemplate the use of a similar fixed Mars communications terminal (MCT), but details of the MCT depend very much on the specifics of a surface mission. If there is to be a major static outpost with a large concentration of elements, for example, a highly capable MCT would make sense, potentially one with a high-elevation tower to maximize local coverage. On the other hand, if there is to be a mobile base, a more modest MCT would be called for that would be integrated into the mobile vehicle.

Based on the concept of operations as described in section 6.6.2 and the feed-forward communications and navigation capabilities that are envisioned for lunar exploration, we have established a strawman set of requirements for the Mars Communications and Navigations System (CNS). A summary of high-level requirements is provided in table 4-9.

[3]Schier, James, et.al., "Lunar Architecture Team Phase 2 Final Report: Communications and Navigation Focus Element Team", September 2007.

**Table 4-9.** Summary of High-level Mars Communications and Navigation System Requirements

| **Requirement** |
|---|
| The CNS shall provide telecommunications, radio navigation, and timing services to missions from Earth to Mars during launch, in LEO, on departure from Earth, in transit between Earth and Mars, on final approach to Mars, in Mars orbit, during EDL on Mars, in the martian atmosphere, and on the martian surface. |
| The CNS shall provide telecommunications, radio navigation, and timing services to missions from Mars to Earth during launch, in Mars orbit, on departure from Mars, in transit to Earth, on final approach to Earth, and during recovery on Earth. |
| CNS communication links shall be continuously available for all human missions to Mars from Earth launch to Earth recovery.[4] |
| The CNS shall provide forward and return store-and-forward relay telecommunications services between users in the vicinity of Mars and Earth. |
| The CNS shall support real-time telecommunications services (voice, video, data) between two or more user assets at Mars. |
| The CNS shall provide radio navigation services to users in the vicinity of Mars. |
| The CNS shall provide communications to support educational outreach activities and public awareness campaigns. |
| The CNS shall provide navigation continuously to a crew MTV, which includes pre-launch, launch, normal and anomalous LEO operations, TMI, Mars transit (both to and from Mars), MOI, Mars orbit, Mars orbit rendezvous, TEI, low Mars orbit, and Earth entry and landing coverage, and also includes all Earth surface landing locations, planned or unplanned. |
| The CNS shall provide low data rate TT&C and voice communication services between the crew MTV at Mars and Earth-based MCC. |
| The CNS shall be capable of supporting human voice transmissions and ancillary data between Earth and the crew MTV in an emergency at a minimum data rate of 2 kbps with the MTV in any orientation (including tumbling). |
| The CNS shall provide navigation support of flight path angle (FPA) at B-plane with a precision of < 0.1 deg. (1 sigma). |
| The CNS shall support radiometric tracking on the access link to a user spacecraft on approach to Mars, with the capability to provide a root-sum-square (RSS) position knowledge uncertainty (3-sigma) of < 300 m at the Mars entry interface (125-km altitude). |
| The CNS shall provide navigation support of relative position with a precision of < 3 m (on-orbit-rendezvous) (1 sigma). |
| The CNS shall provide navigation support of landed position with a precision of 1 m (1 sigma) at sites that may have surface navigation aids. |
| The CNS system shall provide navigation support of landed position with a precision of 100 m (1 sigma) at sites without surface navigation aids. |
| The CNS shall support radiometric tracking on the access link to a user spacecraft on the surface of Mars, with the capability to provide < 30 m RSS (3-sigma) user position knowledge within 1 sol. |
| The CNS shall provide navigation signals on the access link with the capability to continuously maintain < 30 m RSS (3-sigma) user position knowledge. |
| The CNS shall support access links compliant with the CxP command, control, communications, and intelligence (C3I) Interoperability Specification (to be resolved (TBR)). |
| The CNS shall be able to communicate simultaneously with five separate elements in at least two regions in the vicinity of Mars. |
| The CNS shall be able to communicate simultaneously with five separate elements in the vicinity of Mars through direct Earth links. |
| The CNS shall be able to communicate simultaneously with 10 separate elements in a local network that is linked to Earth via a router on the martian surface either through direct links or through one or more relay orbiters. |
| The CNS shall communicate with a reliability of 99% for operational links with interruptions of scheduled communications no greater than one-half hour. |
| The CNS shall support operational TT&C links at Mars with an aggregate data rate up to 6 Mbps. |
| The CNS shall support high-rate communication links from Mars with an aggregate data rate of up to 250 Mbps. |
| The CNS shall support high-rate communication links from Earth with an aggregate data rate of up to 100 Mbps. |

### 4.6.4 Communications trade options

A variety of architectural options have been considered for support of Mars human exploration. Direct-to-Earth communications provides a baseline capability, but has significant limitations. Surface coverage obviously is only

[4] This requirement for continuous link availability reflects the programmatic goal of ensuring a robust communications infrastructure. However, it is likely that this requirement will ultimately be descoped and/or waived for certain mission segments to allow less than 100% availability, based on future studies assessing mission and crew risk, along with cost, as a function of this parameter. (For the LAT study, a 60% availability was ultimately specified, largely driven by the achievable capability of a single lunar relay satellite.) In the event of a descope, it will also be critical to specify the maximum allowable gap time to characterize the risk of telecommunications outages. It is worth noting that the baseline Mars architecture that is described in section 6.6.5 below does in fact meet this requirement of continuous link availability.

possible when Earth is in view; this typically implies nighttime gaps at low and mid latitudes. In addition, at polar latitudes long seasonal DTE outages occur during the martian winter. The large Earth-Mars distance also implies limited communications capability for typical surface-constrained radio systems. Figure 4-15 summarizes X-band

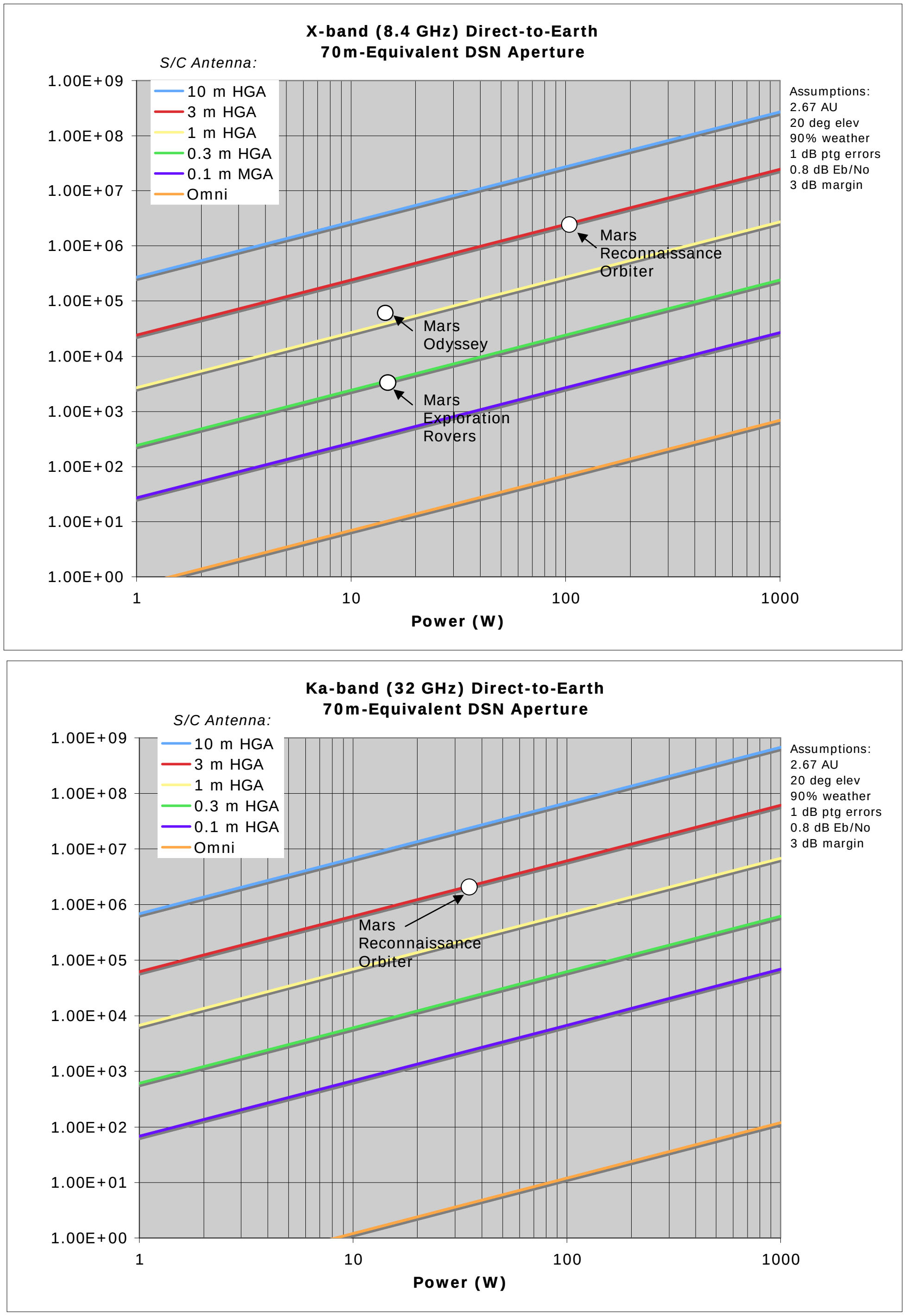



**Figure 4-15.** Direct-to-Earth link performance to a 70-m-equivalent Deep Space Network aperture

and Ka-band Mars-to-Earth communications capabilities as a function of spacecraft transmit power and antenna aperture. Representative performance of current state-of-the-art robotic spacecraft is indicated. The 2005 Mars Reconnaissance Orbiter achieves downlink data rates of several Mbps with its 3-m, 100-W X-band radio system, while the much more resource-constrained MERs, which have 15 W of radio frequency (RF) power radiated through a 28-cm antenna, achieve only a few kbps on the downlink to a 70-m DSN antenna.

For more capable telecommunications services, orbiting MRSs are called for. Options for relay satellites include dedicated telecommunications orbiters as well as the use of other orbital assets with added telecommunications functionality. In the latter category, the MTV itself, as well as potential cruise stages associated with surface-deployed elements (e.g., habitat, cargo vehicles, assuming deployment from orbit) could provide relay functionality at low cost by incorporating a relay payload. (This strategy has been used successfully for Mars robotic exploration, with remote sensing orbiters in low circular orbit, such as the Mars Global Surveyor, Mars Odyssey, and MRO, providing valuable relay services to robotic landers such as the MERs, Phoenix Lander, and the Mars Science Laboratory (Edwards, 2007[5])) However, telecommunications functionality will typically be limited based on competing demands on spacecraft orbit. For instance, the MTV orbit strategy will be driven by the demands of MOI, Mars landing site targeting, on-orbit rendezvous, and TEI; these requirements will likely result in an orbit that is not optimal from a telecommunications perspective. In contrast, a dedicated relay spacecraft can have its orbit optimized for communications and navigation functions. A variety of orbits have been considered for dedicated relay assets, including low-altitude circular orbits, mid-altitude circular orbits, elliptical orbits, and aerostationary orbits (Mars equivalent of Earth geostationary orbits). FOMs were defined, including instantaneous footprint, global coverage, contact time per sol, and maximum gap time. Table 4-10 summarizes these FOMs for several of the considered orbits. Based on these FOMs, the aerostationary orbit option is selected as the most desirable dedicated orbit option, in particular based on its continuous coverage capability. (Nevertheless, future trade studies may wish to explore the telecommunications potential of other planned orbiter vehicles, e.g., MTV, as a low-cost alternative and/or backup to dedicated aerostationary orbiter(s).)

Within the vicinity of the Mars SHAB, an MCT is envisioned to provide high-rate, energy-efficient services to users in the immediate surface environment. The MCT provides the necessary EIRP and gain/temperature (G/T) to achieve high-rate links to MRSs, as well as backup links to the DSN in the event of MRS anomalies.

**Table 4-10.** Comparison of Candidate Mars Relay Satellite Orbit Characteristics

| Orbit Type | Orbit Characteristics | Instantaneous Footprint (>10 deg elev) | Global Coverage | Contact Time per Sol | Max Gap Time |
|---|---|---|---|---|---|
| DSN (for reference) | N/A | 41 % | Seasonal coverage of pole | 9 ± 2.6 hrs | 16 hrs |
| 400-km Sun-Sync Circular | 400-km circular; i=93 deg | 2.5 % | All | 0.36 hrs | 13.9 hrs |
| 4450-km Sun-Sync Circular | 4450-km circular; i=130 deg | 21% | All | 5.2 hrs | 10 hrs |
| Critically-Inclined 0.25-Sol Elliptical Orbit | 950 x 8500 km; i = 63 or 117 deg | 7-28% | All | 5.2 hrs | 12 hrs |
| Circular Low-Inclination Orbit | 1,000-km circular, i=30 deg | 7% | ±60 deg latitude | 2.1 hrs | 14 hrs |
| Aerostationary | 17,303-km altitude; circular, equatorial | 33% (continuous view of region of interest) | ±70 deg latitude | Continuous | 0 (within coverage zone) |

[5]Edwards, Charles D., "Relay Communications for Mars Exploration," *International Journal of Satellite Communications and Networking,* **25**, 111-145, 2007.

### 4.6.5 Baseline communications architecture

#### *4.6.5.1 Overview*

As shown in figure 4-16, we envision a network consisting of an MCT on the martian surface that provides wired and wireless communications in the immediate vicinity of the Mars habitat, one or more MRSs that offer communications and navigation services over the full range of surface exploration as well as support to orbiting assets, and the DSN that provides Earth-based transmit and receive functions.

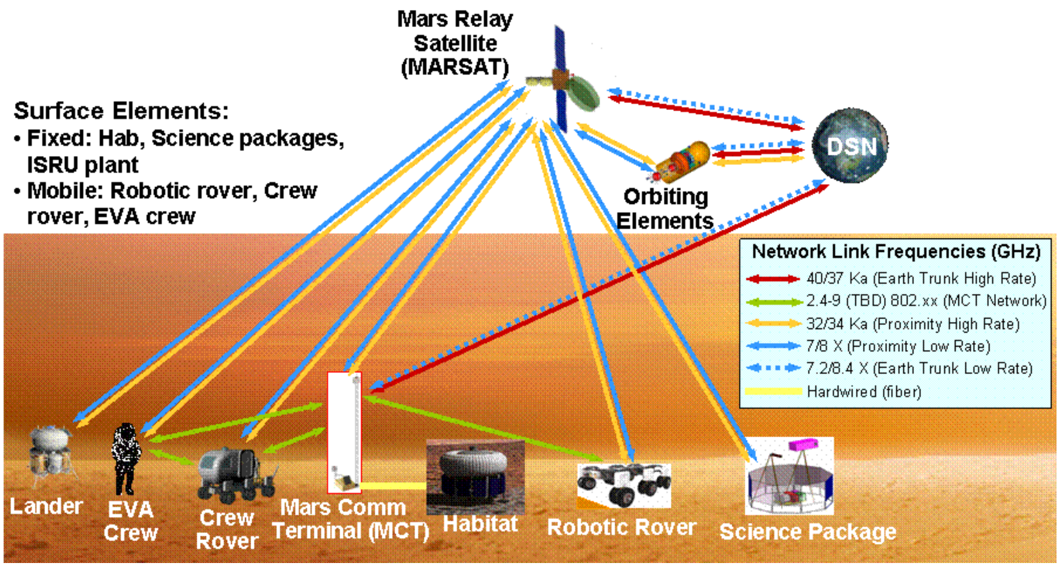



**Figure 4-16.** Overview of Mars communications architecture.

#### *4.6.5.2 Spectrum utilization*

Figure 4-17 shows the spectrum that is to be used for human exploration of Mars as specified in the NASA Space Communications Architecture (Space Communications Architecture Working Group (SCAWG), 2006[6]). Briefly, the 7,145–7,190-MHz (Earth-to-space) and 8,400–8,450-MHz (space-to-Earth) bands are currently allocated and used for deep space communications and would continue to be used for operational links for missions en route to Mars and for direct-from-Earth (DFE) and direct-to-Earth (DTE) links. The Ka-band allocation that is currently used by the DSN, 31.8–32.3-GHz (Earth-to-space) and 34.2–34.7-GHz (space-to-Earth), would be used for high-rate links DTE/DFE by user spacecraft in Mars orbit and by surface elements, and for high-rate relay links with an MRS.

Relay users would use X-band spectrum for operational relay links, with the specific frequencies that are to be chosen to be far enough away from the X-band DTE/DFE bands to prevent interference on the MRS while being close enough to enable easy implementation of user radios that are capable of supporting either DTE/DFE or relay links. Further analysis is needed to specify the frequencies that should be used for this purpose. (Ultra-high frequency (UHF) links are provided for in the NASA Space Communications Architecture spectrum plan, but whether to use them or not should be decided following a trade study.)

[6]Space Communication Architecture Working Group (SCAWG) NASA Space Communication and Navigation Architecture Recommendations for 2005-2030, Final Report, May 15, 2006.

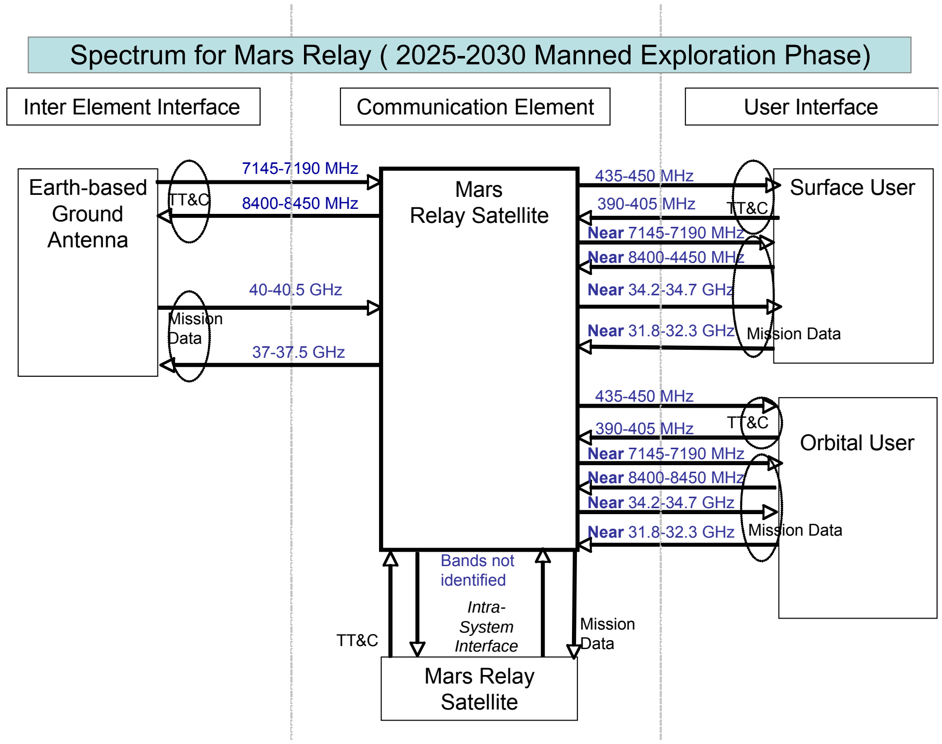



**Figure 4-17.** Mars spectrum from NASA Space Communications Architecture.

### *4.6.5.3 Mars communications terminal*

The MCT establishes a central communications node in the vicinity of the Mars habitat. The MCT will have wired links with fixed elements such as the habitat itself, as well as Wireless Local Area Network (WLAN) links such as IEEE 802.16 for mobile human and robotic assets in the immediate vicinity of the MCT. The MCT will provide routing functions, with data traffic destined for Earth and/or for other Mars regions routed up to MARSAT via X- and Ka-band links for operational and high-rate data , respectively. For a fixed habitat, the MCT may be a stand-alone element, incorporating up to a 10-m tower to increase the coverage zone around the habitat area. For a mobile habitat option, a more modest MCT will need to be integrated into the habitat itself. The MCT will also support DTE/DFE links to the DSN as a contingency in the event that orbital relay assets are not available.

### *4.6.5.4 Mars relay satellite*

Based on its ability to provide continuous communications services to Mars surface assets within a very large footprint around a low- or mid-latitude landing site, along with a simple operations concept resulting from a constant orbiter position in the surface user's reference frame, a Mars aerostationary orbit is selected as an initial relay baseline. We have established a point design for a Mars aerostationary relay satellite (MARSAT) to understand the mass, performance, and cost of such a relay asset. In keeping with the philosophy of leveraging the lunar communications architecture, our baseline MARSAT design provides similar functional capabilities as the LRS in terms of bandwidth on the surface-to-orbiter and orbiter-to-Earth links. However, the much greater orbiter-to-Earth communications distance for a Mars relay orbiter and the choice of an aerostationary orbit (for which no similar option exists at the moon) leads to some significant differences in the resulting spacecraft design.

Figure 4-18 illustrates the basic functional capabilities of the baseline MARSAT design. Based on the spectrum plan described in section 6.6.5.2, MARSAT uses X-band links for operational TT&C data, and Ka-band links for high-rate mission data. The eight corresponding MARSAT links (operational vs. high-rate, link-to-Earth vs. link-to- Mars surface, and forward vs. return) are listed in table 4-11. The driving functional requirement is the support of a high-rate downlink to Earth with a capability of 250 Mbps at a maximum Earth-Mars distance of 2.7 AU. Achieving this level of performance, which is roughly two orders of magnitude beyond the current state of the practice as

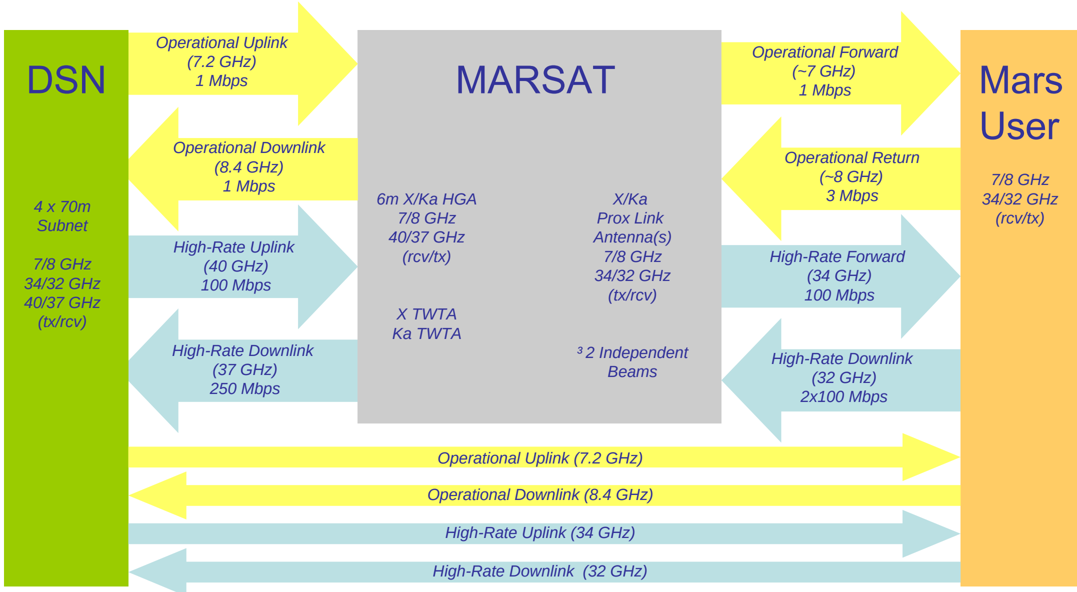



**Figure 4-18.** Baseline Mars aerostationary communication relay satellite.

**Table 4-11.** Mars Aerostationary Relay Satellite Data Rate Capabilities

| Domain | Class | Direction | Freq | Data Rate (LAT-derived)[1] | Data Rate (Descope)[2] |
|---|---|---|---|---|---|
| Proximity | Operational TT&C | Forward (MARSAT-to-User) | 7 GHz | 1 Mbps | 1 Mbps |
| Proximity | Operational TT&C | Return (User-to-MARSAT) | 8 GHz | 3 Mbps | 3 Mbps |
| Proximity | High-rate Mission Data | Forward (MARSAT-to-User) | 34 GHz | 100 Mbps | 10 Mbps |
| Proximity | High-rate Mission Data | Return (User-to-MARSAT) | 32 GHz | 2x100 Mbps | 2x10 Mbps |
| Deep Space (@ 2.7 AU) | Operational TT&C | Uplink (DSN-to-MARSAT) | 7.2 GHz | 1 Mbps | 1 Mbps |
| Deep Space (@ 2.7 AU) | Operational TT&C | Return (MARSAT-to-DSN) | 8.4 GHz | 1 Mbps | 1 Mbps |
| Deep Space (@ 2.7 AU) | High-rate Mission Data | Uplink (DSN-to-MARSAT) | 40 GHz | 100 Mbps | 10 Mbps |
| Deep Space (@ 2.7 AU) | High-rate Mission Data | Return (MARSAT-to-DSN) | 37 GHz | 250 Mbps | 25 Mbps |

[1]LAT-derived baseline assumes 4x70-m-equivalent DSN aperture
[2]Descope case assumes 1x70-m-equivalent DSN aperture

embodied by the 2005 Mars Reconnaissance Orbiter, requires growth in the spacecraft EIRP, which is the product of spacecraft transmit power and antenna gain, and/or the effective receive aperture of the DSN. For the purpose of this baseline point design, we assume an expanded DSN capability that corresponds to the equivalent of four 70-m antennas at each DSN complex (potentially achieved via arraying a large number of smaller antennas) that are equipped with X- and Ka-band transmit/receive capability.

The resulting MARSAT design, which is illustrated in figure 4-19, uses a deployable, body-fixed, 6-m X-/Ka-band high-gain antenna that is fed by traveling wave tube amplifiers that have an output power of 500 W at Ka-band and 100 W at X-band for the downlink to Earth. The same antenna is used for receipt of uplink from Earth. Additional low-gain antennas on MARSAT support early cruise and safe mode communications needs.

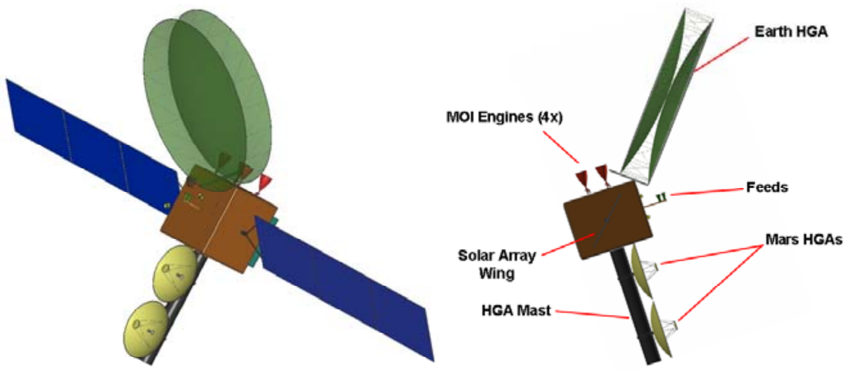



**Figure 4-19.** Strawman Mars aerostationary relay satellite design.

Proximity links to users on the martian surface are supported with a pair of gimbaled 2-m X-/Ka-band antennas. Each antenna can be independently pointed, thereby allowing link establishment to two sets of users anywhere on the disk of Mars or in Mars orbit within view of MARSAT. From its 17,030-km altitude above the martian equator, MARSAT can access latitudes up to ±70 degrees for a 10-degree surface elevation mask. As shown in figure 4-20, each proximity link antenna will establish a footprint on the surface within which users can access MARSAT services; for the X-band TT&C links, the footprint diameter (calculated out to 3 dB pointing loss) is 300 km; for the more directional high-rate Ka-band mission data links, the surface footprint diameter is 75 km.

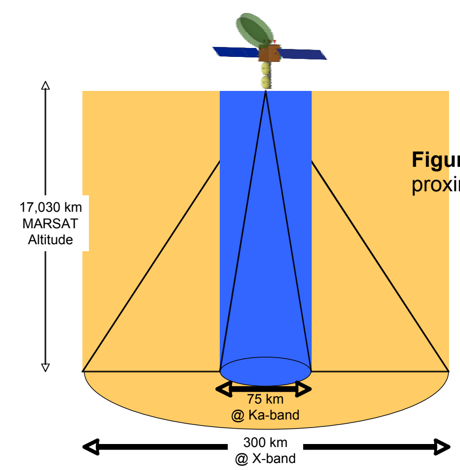



**Figure 4-20.** Mars aerostationary relay satellite proximity link footprint.

MARSAT could launch on an Atlas V 531 launch vehicle that has a launch mass of 4,212 kg, including a 30% allocation to contingency mass margin. The spacecraft dry mass (again, including contingency) is 1,881 kg with a propellant load of 2,331 kg. The large propellant load supports a simple chemical propulsive MOI into a 300×17,030-km capture orbit, with an inclination that is dictated by the declination of the arrival asymptote. The MARSAT would remain in this orbit for as many as 7 months, or until the line of apsides precesses into the martian equatorial plane, and then execute a combined inclination change and periapse raise maneuver to achieve the final circular, equatorial, aerostationary orbit. A total delta-V of 2,175 m/s is required to achieve the final service orbit in this scenario. Several options could be considered to reduce the delta-V requirements and, hence, the overall spacecraft wet mass. For instance, MARSAT could insert into a higher-apoapse 2-day capture orbit (300 × 56,675 km) and then aerobrake down to the 300×17,030-km preliminary phasing orbit, thus reducing the total delta-V requirements by roughly 10%. Alternatively, MARSAT could be carried into orbit as part of the crewless precursor cargo mission, which plans to use aerocapture to minimize propulsive requirements, and then be released into this capture orbit prior to targeting of the rest of the cargo spacecraft to landing.

An individual MARSAT spacecraft will experience occultations on its link to Earth of up to 1 hour 20 minutes each day as the orbiter passes behind Mars (as viewed from Earth). However, with a pair of MARSAT orbiters, which are separated by 20 degree in longitude, one orbiter will always be in view of Earth. Thus, the two-element MARSAT constellation that is depicted in figure 4-21 offers continuous connectivity with Earth as well as functional redundancy.

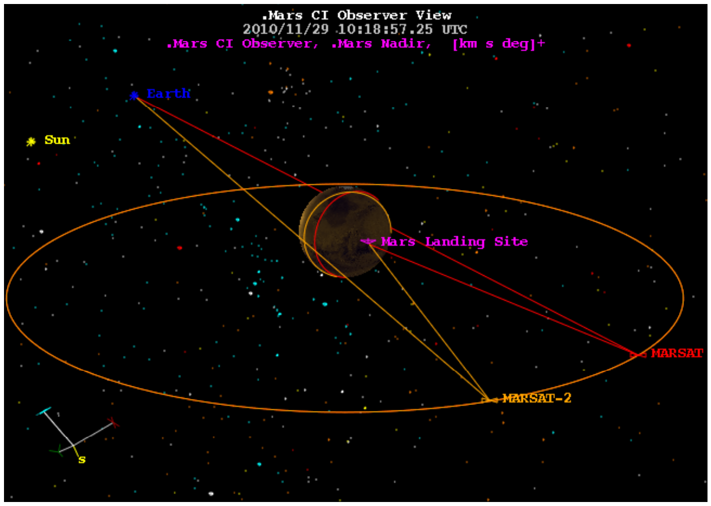



**Figure 4-21.** Aaerostationary relay satellite orbiters provide continuous end-to-end connectivity.

At superior conjunction (when Earth and Mars are on opposite sides of the sun), the MARSAT-Earth line-of-sight will pass close to the sun, with potential link interruption due to solar occultation or effects of solar plasma. Due to the relative inclinations of the Earth and Mars orbits, it is often the case that superior conjunction does not result in an actual solar occultation of the MARSAT-Earth link. However, the X-band link is likely to suffer degradation for sun-Earth-probe (SEP) angles of up to 3 degrees. At the shorter wavelength of the Ka-band link, plasma effects are reduced, and periods of link degradation are likely limited to SEP angles of only up to 1 degree. Table 4-12 identifies periods of link degradation for all superior conjunctions occurring during the 2030–2040 timeframe.

**Table 4-12.** Superior Conjunctions and Likely Duration of DegradedMars-Earth Communications

| Epoch of Superior Conjunction | Duration of Link Degradation (days) | |
|---|---|---|
| | X-band (SEP < 3 deg) | Ka-band (SEP < 1 deg) |
| 25-May-2030 | 24 | 9 |
| 11-Jul-2032 | 19 | 3 |
| 19-Aug-2034 | 17 | 1 |
| 24-Sep-2036 | 17 | 5 |
| 1-Nov-2038 | 19 | 8 |
| 17-Dec-2040 | 22 | 8 |

**Table 4-13.** Comparison of Key Parameters for Mars Aerostationary Relay Satellite

| | MARSAT | |
|---|---|---|
| | **Baseline Option** | **Descope Option** |
| Maximum Downlink Rate | 250 Mbps | 25 Mbps |
| Spacecraft (S/C) Dry Mass | 1,315 kg | 1,112 kg |
| S/C Dry Mass w/30% contingency | 1,881 kg | 1,589 kg |
| Propellant and Pressurant | 2,330 kg | 2,024 kg |
| Propulsive Delta-V | 23,62 m/s | |
| S/C Wet Mass | 4,212 kg | 3613 kg |
| Launch Vehicle | Atlas V 531 | Atlas V 521 |
| S/C Power | 3,161 W | 1886 W |
| Mission Duration | 10.8 yrs | |
| Radiation Total Dose | 27 krad | |
| Atmospheric control and supply (ACS) | 3-axis | |
| Pointing Control | 80 arcsec | |
| Development Cost[1] (fiscal year 2007 (FY07) $M) | $524M | $491M |
| Launch Vehicle Cost (FY07 $M) | $180M | $160M |
| Operations Cost[2] (FY07 $M) | $87M | $87M |
| Total Mission Cost[3] (FY07 $M) | $791M | $738M |

[1] Development cost includes 30% reserves.

[2] Operations cost includes 15% reserve.

[3] Total mission cost is exclusive of DSN tracking costs.

To weigh the cost-sensitivity of the communications architecture to the relatively immature communications data rate requirements, a descoped MARSAT option was considered that had an order-of-magnitude reduction in all the high-rate mission data Ka-band links. For this case, the spacecraft and ground are assumed to share in the Performance descope; in particular, the DSN capability is assumed to be descoped to the equivalent of a single 70-m antenna at each complex (again, potentially implemented via arraying of smaller antennas); the remaining descope is achieved by reducing the spacecraft EIRP. For the purpose of this point design, the EIRP reduction was achieved by reducing the Ka-band transmit power to 200 W on the downlink to Earth, leading to a significant reduction in the spacecraft bus power requirements, which results in a 200-kg reduction in the spacecraft dry mass and a roughly 600-kg reduction in wet mass.

### 4.6.6 Position, navigation, and time considerations

The position, navigation, and time (PNT) approach is summarized by mission phase in table 4-14. Figure 4-22 depicts key features of the EDL approach showing the best-case aided situation with DSN, MRSs, and MCTs in view. The worst-case situation occurs if these elements are not in view and the lander has to rely entirely on on-board capabilities.

**Table 4-14.** Position, Navigation, and Time Approach by Mission Phase

| Phase | PNT Approach | Issues |
|---|---|---|
| Cruise to and from Mars | Existing DSN tracking, position determination, and trajectory maneuver capabilities handle human Mars missions during the cruise phase. | None. |
| EDL – Autonomous | The ETDP is developing the Autonomous Landing and Hazard Avoidance Technology (ALHAT) project to advance technologies and an integrated capability for space vehicle landing using entirely vehicle-based sensors, effectors, and control. Terrain-relative navigation (TRN) is used from entry interface to final approach point. Hazards at the landing site are detected and avoided by combining detailed terrain map data with real-time sensor data. Landing accuracy should be < 100m (3σ). | Unaided approach does not provide telemetry coverage of critical events in case of failures. ALHAT does not have a planned next phase for Mars EDL. |
| EDL – Aided | DSN, MARSAT, and MCT provide measurements of one and two-way range and Doppler tracking when in view of the lander. MARSAT and MCT provide time synchronization to drive timing errors to nearly zero. Landing accuracy should be < 10m (3σ) without crew intervention. | Detailed analysis of navigation performance during EDL was not performed for the MAT study. |
| Surface Navigation | Mobile systems will have on-board odometers, star trackers, and inertial measurement units (IMUs) providing accuracy that accumulates errors up to ~1.2 km (Galor, 2005[7]). Ranging to two longitudinally separated MARSATs, augmented with on-board digital elevation model (DEM) data, can provide radio-based surface positioning at accuracies below 100 m. Terrain feature database combined with optical feature extraction and feature recognition provides an independent technique for surface-relative positioning, with performance dependant on surface morphology. | None. |
| Ascent | Similar to EDL. | Similar to EDL. |

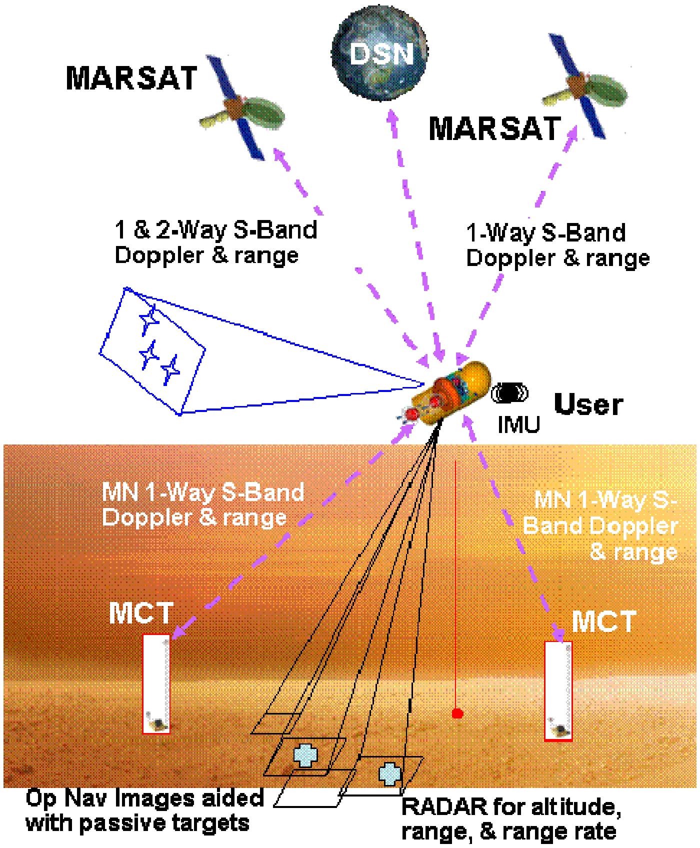



**Figure 4-22.** Navigation approach for aided Mars entry, descent, and landing

[7]Gaylor, David, Benjamin Malay, and George Davis, "Stellar-Aided Inertial Navigation Systems for Lunar and Mars Exploration," 2005 Flight Mechanics Symposium, NASA Goddard Space Flight Center, Greenbelt, MD, October 18-20, 2005.

### 4.6.7 Heritage and key differences from the LAT communication/navigation architecture

Key differences between the LAT's communication and navigation architecture and the reference Mars architecture are shown in table 4-15. The Mars architecture was assessed for two cases of forward and return data rates: the LAT data rates on the high end, and 1/10$^{th}$ of the LAT data rates on the low end.

**Table 4-15.** Comparison of Lunar and Mars Reference Architectures

| Characteristic | Lunar Architecture | Mars Architecture | | Design & Technology Implications |
|---|---|---|---|---|
| High-rate data band (trunk line) | Ka-band, 40/37 GHz | Ka-band, 40/37 GHz | | No change |
| Proximity data band | Ka-band, 23/26 GHz | Ka-band, 34/32 GHz | | Allows user radio to access relay orbiter or DSN for contingency |
| Low-rate data band | S-band, 2.0/2.1 GHz | X-band, 7/8 GHz | | X-band is standard for other deep space missions |
| Space loss on Earth links | –236 dB | –283 dB | | New, high-power traveling wave tube amplifier (TWTA) required; improved coding and modulation required |
| Orbit | 12-hour, frozen, highly elliptical, and eccentric | 1-sol aerostationary | | Mars surface antennas can be simpler (fixed pointing vs. gimbaled on moon); MARSAT not as useful for surface navigation |
| Maximum slant range | 11,000 km | 20,000 km | | Higher-power proximity transceivers required |
| Maximum forward data rate | 100 Mbps | 100 Mbps | 10 Mbps | Advanced antenna technologies required, especially for high-end case |
| Maximum return data rate | 250 Mbps | 250 Mbps | 25 Mbps | Advanced antenna technologies required, especially for high-end case |
| DSN ground terminals | 4x18-m antennas (Goldstone, Madrid, Canberra, and White Sands) | 12x70-m antennas (four each at Goldstone, Madrid, Canberra) | 3x70-m antennas (Goldstone, Madrid, Canberra) | Current 70-m antennas are near end of life. Replacement by new 70-m antennas or a large array of smaller-diameter antennas is being studied. The large array is more scalable and flexible, allowing deferral of cost until decision is made to add capacity |

### 4.6.8 Recommendations for future work

This relatively short study focused on establishing a reference Mars communications and navigation architecture, which is derived in part from capabilities baselined for lunar exploration. Topics for future study include:

- Refinement of the fundamental communications bandwidth requirements, which are based on improved understanding of the needs of individual exploration elements
- Evaluation of the potential benefits of optical communications for relay and Earth-Mars links
- Low-cost relay options using non-dedicated relay assets (e.g., MTV) as alternatives to dedicated MARSAT(s)
- Comprehensive study of navigation tools, including radio-based, inertial, and image-based techniques

# 5 TRANSPORTATION SYSTEMS

## 5.1 Overview

For DRA 5.0, the transportation systems work that is described in this chapter focused primarily on updating previous DRMs using new CxP-derived launch vehicles (see section 5.2). The impacts of using these vehicles for a human mission to Mars were examined both in the context of required performance (e.g., IMLEO, number of launches, etc.) and in the context of their impacts to existing ground infrastructure at KSC (see section 5.3). A final decision was not made as to whether the chemical Earth departure stage that was used for lunar missions would be slightly augmented for trips to Mars or whether a new nuclear stage would be developed, but the impact of both cases is assessed (see section 5.4). In previous DRMs, a small capsule was envisioned for the Earth return vehicle (ERV), but with the design of the Orion CEV a block-upgrade path now exists that will seek to augment the capsule that is currently being designed to go to the moon for use on a round-trip Mars mission. This will primarily involve augmenting the TPS on the current Orion and certifying the vehicle for extended times in a space environment (see section 5.5). Perhaps the most important advancement in knowledge since the last DRM comes with respect to the EDL systems that are to be employed at Mars to land payloads on the order of 30–50 t. Previous estimates of EDL system mass were not conservative enough given the great unknowns that are still associated with landing payloads > 1 t on Mars. The new assessment (see section 5.6) details a more conservative estimate of EDL system mass that has Substantially increased, even in spite of the advantage that was gained from using a common Ares V launch shroud/ payload entry shield. Mass increases in this subsystem are a prime contributor to the overall increase in the initial mass-to-LEO estimates that are given in this DRA as compared to previous DRMs.

Despite the fact that detailed analysis work was not performed during DRA 5.0 on the MTV, MAV, or SHAB (sections 5.7 through 5.9), past analysis of all three vehicles was updated with current assumptions. This especially applies to the case of the MAV, in which ascent stages using ISRU were parametrically sized in comparison to ascent stages that are fully fueled from the beginning. The impact of using ISRU on the MAV was traced back all the way to LEO to make a recommendation with regard to the use of ISRU (see section 3.5). To close out the transportation systems analysis, an assessment of landing plume effects was completed (see section 5.10)

## 5.2 Launch Systems: Reference Vehicle

The reference vehicle for the currently envisioned lunar campaign (as described by the CxP in autumn 2007 and ever-evolving) served as the point of departure for the Mars DRA 5.0 study; it consists of two 5-segment reusable solid rocket booster (RSRBs), a core stage that is powered by five Pratt & Whitney Rocketdyne RS-68B engines, an Earth departure stage (EDS) that is powered by one Pratt & Whitney Rocketdyne J-2X, and a payload shroud. This vehicle has a gross liftoff mass of approximately 3,323,000 kg (7,326,000 $lb_m$) and a height of 110.3 m (361.9 ft). This vehicle can be seen in figure 5-1. Because a new heavy-lift launch vehicle that will be designed specifically for Mars would be too expensive, maximum emphasis was placed on analyzing how well the vehicles that are currently being designed for the lunar mission could be adapted to meet objectives for Mars. As the Ares V that is being designed for lunar missions evolves, it must be continually assessed in reference to Mars suitability.

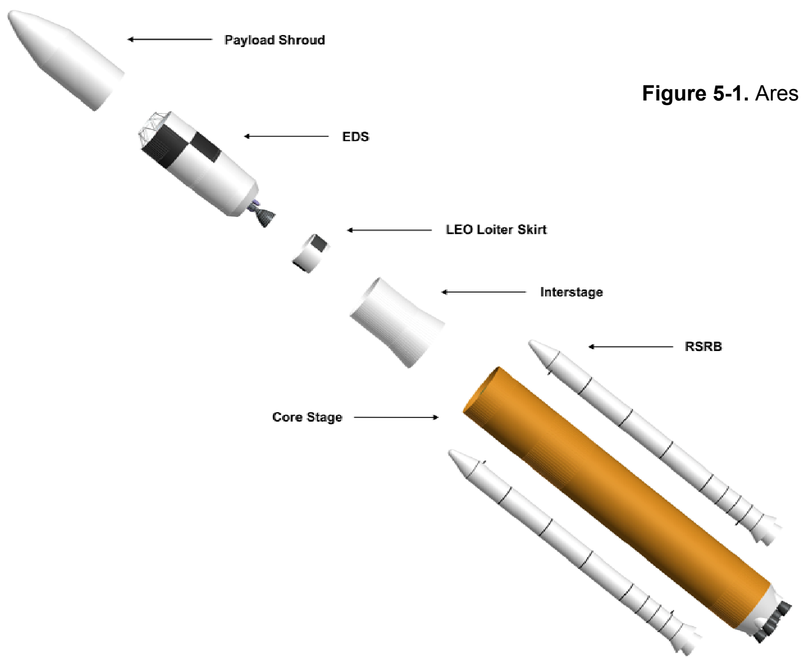



**Figure 5-1.** Ares V expanded view.

**REUSABLE SOLID ROCKET BOOSTERS**

The RSRBs (figure 5-2)are comprised of five segments that are similar to those used on Ares I. These boosters have a launch mass of 732,000 kg (1,613,000 $lb_m$) each, which includes approximately 626,000 kg (1,381,000 $lb_m$) of expended mass. Jettisoned at 126.6 seconds, the RSRBs provide 67% of the total liftoff thrust, or approximately 15,500 kN (3,480,000 $lb_f$) each. They use Al powder as a fuel, ammonium perchlorate as an oxidizer, iron oxidizer powder as a burning rate catalyst, polybutadiene acrylic acid acrylonitril terpolymer (PBAN) as a binder, and an epoxy curing agent. The five-segment RSRBs are based on the four-segment RSRBs that are used for the space shuttle. It is expected that many technologies from those boosters will also be used for the five-segment RSRBs. This will include non-pressurized skirt sections, the booster separation motors (BSMs), the attach hardware, forward frustum, and a nosecone.

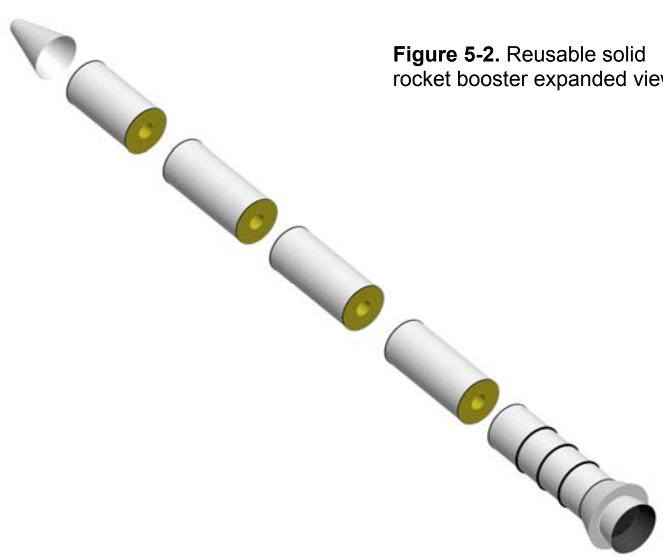


**Figure 5-2.** Reusable solid rocket booster expanded view.

**CORE STAGE**

The core stage (figure 5-3) is a 10.06-m (33-ft) outer diameter stage that is powered by five RS-68B engines. These engines operate at a 106% thrust level (with an associated $I_{sp}$ of 414.2 s), which provides approximately 3,500 kN (784,000 $lb_f$) vacuum thrust each. The core stage is fueled by 200,000 kg (442,000 $lb_m$) of $LH_2$, and 1,196,000 kg (2,637,000 $lb_m$) of $LO_2$ is used as an oxidizer. The launch mass of the core stage (including the interstage) is approximately 1,557,000 kg (3,432,000 $lb_m$), of which 165,000 kg (354,000 $lb_m$) are jettisoned after burning for about 325.3 seconds. The total height of the core stage is 65.0 m (213.3 ft) without the interstage, or 79.8 m (262.0 ft) with the interstage.

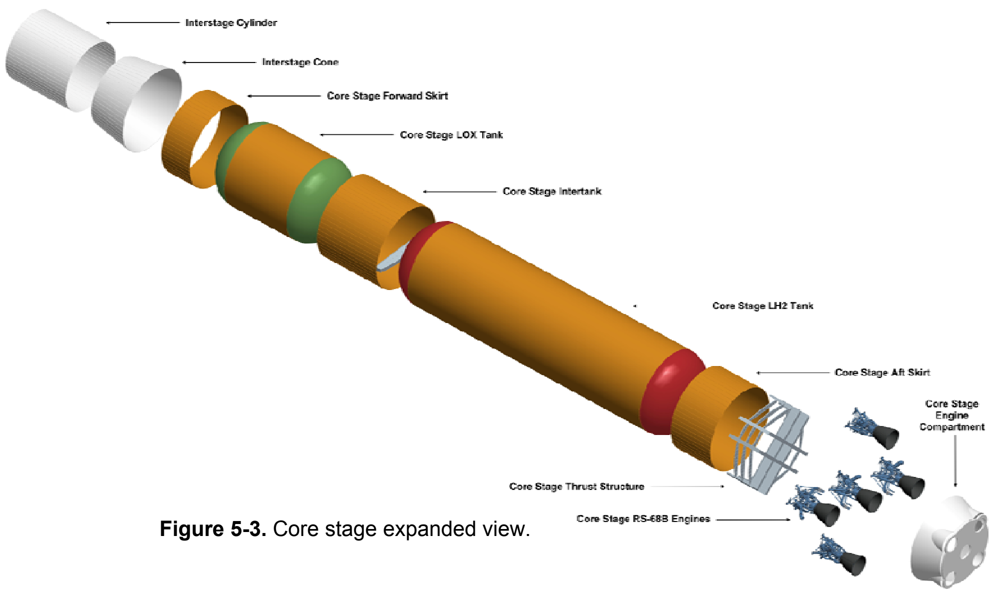



**Figure 5-3.** Core stage expanded view.

**EARTH DEPARTURE STAGE**

The EDS (figure 5-4) is an 8.4-m (27.6-ft) outer diameter stage that is powered by one J-2X engine. This engine operates at a 100% thrust level (with an associated $I_{sp}$ of 448 s), providing 1,300 kN (294,000 $lb_f$) vacuum thrust. The EDS is a dual-burn stage. The first burn provides a portion of the suborbital burn, which has a duration of 441.9 seconds and consumes 132,000 kg (290,000 $lb_m$) of propellant (~58.9% of the total EDS propellant loading). For lunar missions the second burn will be the trans-lunar injection (TLI) maneuver. This burn has a duration of 309.8 seconds, and consumes 92,000 kg (203,000 lbm) of propellant (~41.1% of the total EDS propellant loading). The launch mass of the EDS and Altair lander is approximately 293,000 kg (646,000 $lb_m$), of which 23,000 kg (51,000 $lb_m$) is jettisoned after the TLI maneuver. The total usable propellant loading of 224,000 kg (493,000 $lb_m$) includes approximately 34,000 kg (76,000 $lb_m$) of $LH_2$ and 189,000 kg (417,000 $lb_m$) of $LO_2$. The total height of the EDS (not including the Altair or payload shroud) is 23.3 m (76.4 ft); however, approximately 14.8 m (48.7 ft) are suspended in the intertank at liftoff.

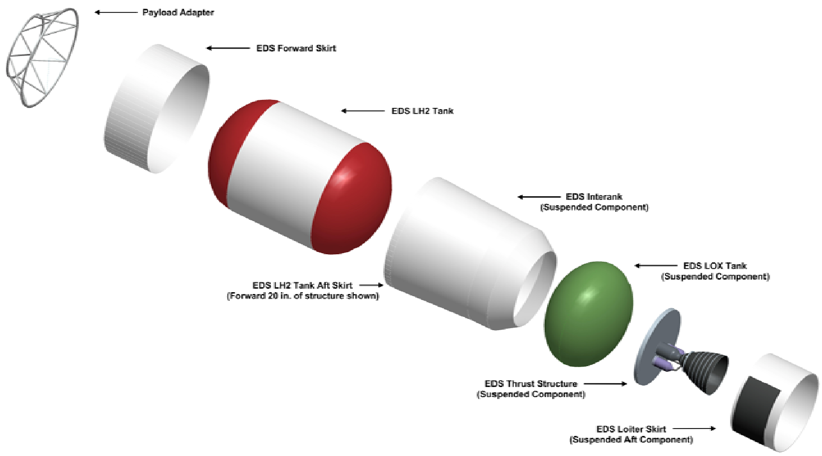



**Figure 5-4.** Earth departure stage expanded view.

### 5.2.1 Reference vehicle: lunar mission reference shroud

The shroud that will be used on the lunar mission reference vehicle is a structure that is built of IM7/8552 composite. It has an internal diameter (ID) of 7.5 m (24.6 ft) and an outer diameter (OD) of 8.4 m (27.6 ft). The total height of the shroud is 22.0 m (72.2 ft), which includes a barrel section of 12.0 m (39.4 ft) in length. These dimensions can be seen in figure 5-5. It is assumed that the shroud that will be used for the Mars campaign will be a variant of this shroud or perhaps a "block upgrade."

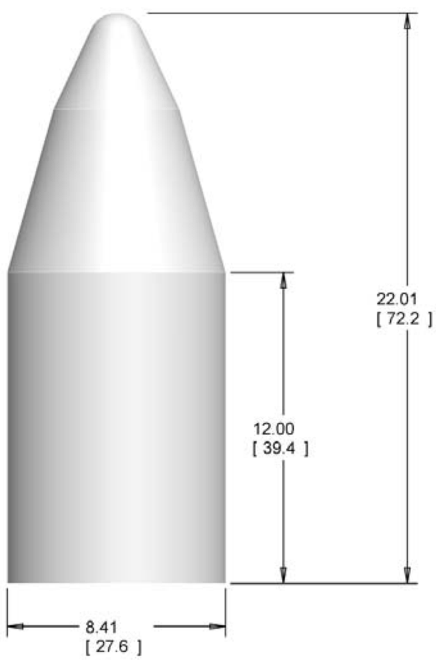



**Figure 5-5.** Option A shroud.

The structural components of the shroud are calculated by the launch vehicle analysis (LVA) tool at the NASA Marshall Space Flight Center (MSFC). Main drivers of this calculation are shroud type, material inputs for the IM7/8552 composite, maximum dynamic pressure that is experienced during flight, maximum g-loads that are experienced during flight, and dimensions. The shroud type that was analyzed is based on the Titan IV bi-conic design, and the material that was used for the structure is a pseudo-isotropic lay-up of IM7/8552 fiber that is placed in an isogrid stiffened pattern.

The maximum dynamic pressure of the reference vehicle is found to be 30.5 kPa (637 $lb_f/ft^2$), while the flight loads (g's) are approximately 3.9. Analysis showed that the nosecone of the shroud has a mass of about 1,800 kg (3,960 $lb_m$) while the barrel section has a mass of 2,600 kg (5,760 $lb_m$). In addition, a TPS for the nosecone and acoustic blankets is included in the total shroud mass. Overall, the total value of the payload shroud is approximately 5,840 kg (12,900 $lb_m$), as can be seen in table 5-1.

**Table 5-1.** Reference Shroud Mass Summary

| Reference Shroud | | |
|---|---|---|
| **Component** | **kg** | **$lb_m$** |
| Nosecone | 1,796.5 | 3,960.7 |
| Barrel Section | 2,614.1 | 5,763.1 |
| Acoustic Blankets | 1,344.1 | 2,963.3 |
| TPS | 82.1 | 180.9 |
| Total | 5,836.8 | 12,868.0 |

### 5.2.2 Mars shroud options

For DRA 5.0, the MAWG identified three shroud options/variants, which were based on the option that was described above, that required further study as part of DRA 5.0. These three shroud options are given the identifiers of Option A, Option B, and Option C. The shroud options are flown on the reference vehicle to see the performance impact of larger shrouds.

The MAWG identified the shrouds as having an ID of 7.5 m (24.6 ft) with a barrel length (BL) of 12 m (39.4 ft) (Option A), an ID of 10 m (32.8 ft) with a BL of 25 m (82 ft) (Option B), and an ID of 12 m (39.4 ft) with a BL of 35 m (114.8 ft) (Option C). The OD is estimated using a 0.9-m (3-ft) ID-to-OD conversion factor for all three shroud options. The shroud options that have a larger OD than the current OD of the EDS will require a transition section to integrate with the reference vehicle. This transition section will be a conic section that will taper from the OD of the shroud to the 8.4-m (27.6-ft) OD of the EDS. In addition, these three shroud options are analyzed using an IM7/ 8552 composite material.

Option A has the same dimensions as the lunar mission reference shroud, as described in section 5.2.1 and shown in figure 5-5. This shroud has an ID of 7.5 m (24.6 ft) and an OD of 8.4 m (27.6 ft). The total height of the shroud is 22.0 m (72.2 ft), which includes a barrel section of 12.0 m (39.4 ft) in length. For the MAWG study, the total mass of the Option A shroud is found to be 5,988 kg (13,200 $lb_m$). This can be seen in table 5-2.

**Table 5-2.** Option A Shroud Mass Summary

| Option A Shroud | | |
|---|---|---|
| **Component** | **kg** | **$lb_m$** |
| Nosecone | 1,874.9 | 4,133.5 |
| Barrel Section | 2,686.4 | 5,922.4 |
| Acoustic Blankets | 1,344.1 | 2,963.3 |
| TPS | 82.1 | 180.9 |
| Total | 5,987.5 | 13,200.1 |

The Option B shroud, which is shown in figure 5-6, is larger than the Option A shroud. It has an ID of 10.0 m (32.8 ft) and an OD of 10.9 m (35.8 ft). An adapter cone is needed for this shroud to reduce the OD of 10.9 m (35.8 ft) to the EDS OD of 8.4 m (27.6 ft). This adapter cone is found to be 2.17 m (7.1 ft) in length. The BL of this shroud is defined as 25 m (82.0 ft), and the total length of the Option B shroud is found to be 40.2 m (131.7 ft). The total mass of this shroud is found to be 19,022 kg (41,936 $lb_m$) (table 2-3).

**Table 5-3.** Option B Shroud Mass Summary

| Option B Shroud | | |
|---|---|---|
| **Component** | **kg** | **$lb_m$** |
| Nosecone | 3,664.8 | 8,079.6 |
| Barrel Section | 10,070.0 | 22,200.5 |
| Adapter Cone | 2,097.1 | 4,623.4 |
| Acoustic Blankets | 3,051.5 | 6,727.4 |
| TPS | 138.3 | 304.8 |
| Total | 19,021.7 | 41,935.7 |

The Option C shroud, which is seen in figure 5-7, is larger than both the Option A and Option B shrouds. It has an ID of 12.0 m (39.4 ft) and an OD of 12.9 m (42.3 ft). An adapter cone is needed for this shroud as well to reduce the OD of 12.9 m (39.4 ft) to the EDS OD of 8.4 m (27.6 ft). This adapter cone is found to be 3.90 m (12.8 ft) in length. The BL of this shroud is defined as 35 m (114.8 ft), and the total length of the Option C shroud is found to be 54.3 m (178.0 ft). The total mass of this shroud is found to be 33,361 kg (74,144 $lb_m$), which is reduced to its component level in table 5-4.

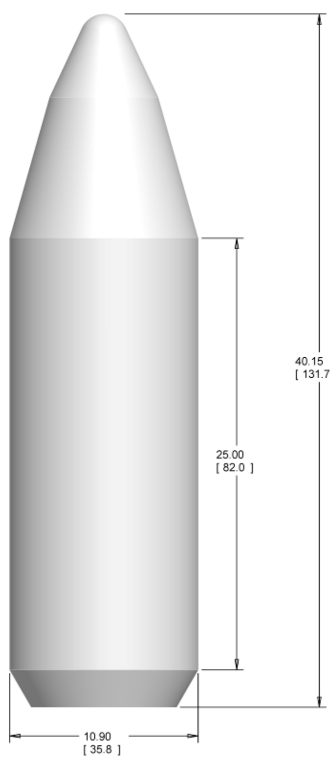



**Figure 5-6.** Option B Shroud

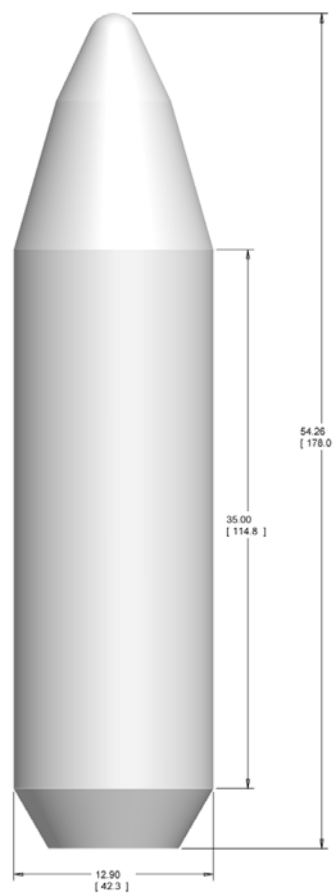



**Figure 5-7.** Option C Shroud

**Table 5-4.** Option C Shroud Mass Summary

| Option C Shroud | | |
|---|---|---|
| **Component** | **kg** | **$lb_m$** |
| Nosecone | 5,395.6 | 11,895.3 |
| Barrel Section | 17,858.6 | 39,371.5 |
| Adapter Cone | 5,374.2 | 11,848.1 |
| Acoustic Blankets | 4,809.5 | 10,603.2 |
| TPS | 193.0 | 425.5 |
| Total | 33,631.0 | 74,143.7 |

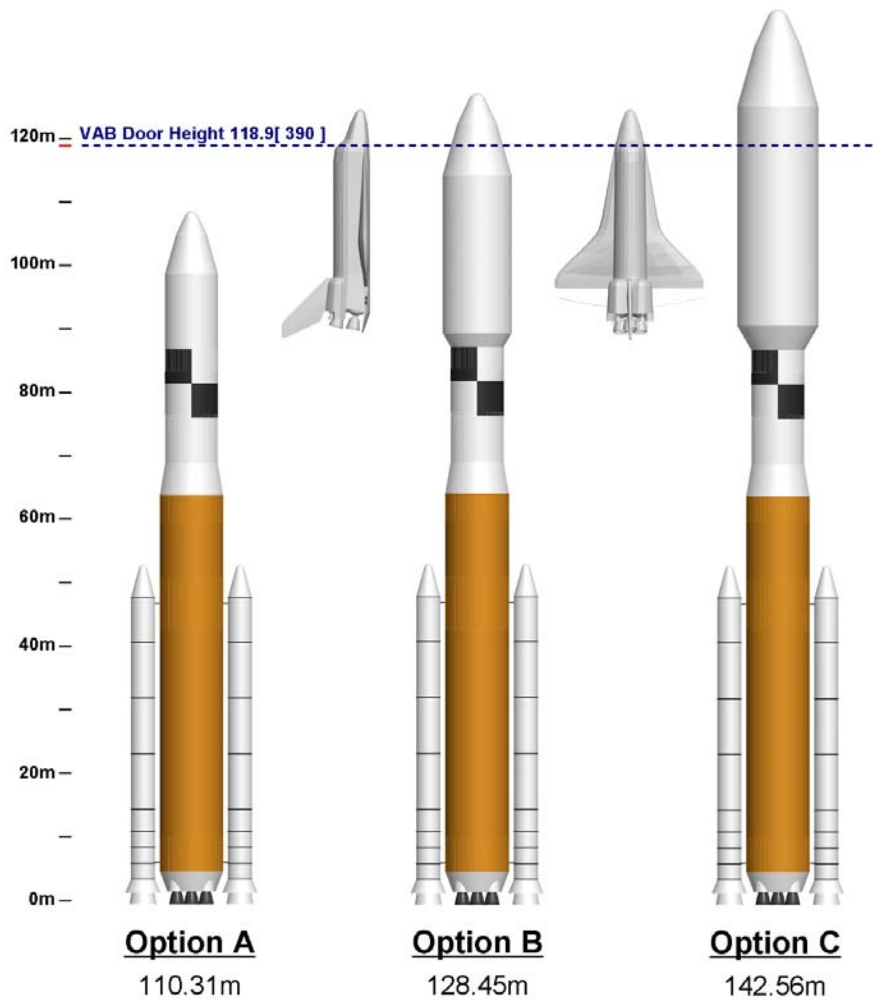



**Figure 5-8.** Shroud options A, B, and C on modified reference vehicle.

In addition to shroud Options A, B, and C (figure 5-8, above), the reference vehicle was analyzed in the scenario that is used as a Mars fuel stage. In this scenario, the "TMI module" would only require a nosecone because no payload would actually be launched with the vehicle. The only payload would be the fuel that was remaining in the EDS tanks. This nosecone would have the same OD as the EDS stage, 8.4 m (27.6 ft). No acoustic blankets would be required, but a TPS for the nosecone would be included. The total mass of this nosecone was found to be approximately 1,960 kg (4,310 $lb_m$), which is reduced to its component level in table 5-5.

**Table 5-5.** Fuel Stage Shroud Mass Summary

| Fuel Stage Shroud | | |
|---|---|---|
| **Component** | **kg** | **lb$_m$** |
| Nosecone | 1,874.6 | 4,132.9 |
| TPS | 82.1 | 180.9 |
| Total | 1,956.7 | 4,313.8 |

As seen in figure 5-9, the Mars fuel stage shroud has a total height of 10.0 m (32.9 ft), and an OD of 8.4 m (27.6 m). It is basically the reference shroud with no barrel section.

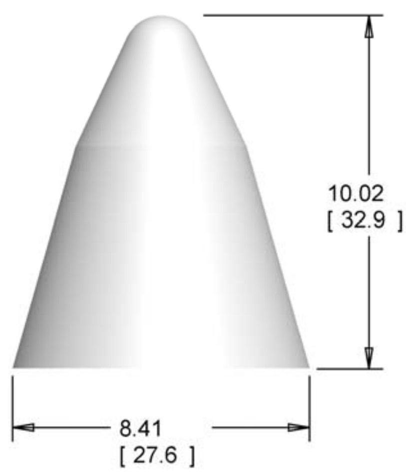



**Figure 5-9.** Mars fuel stage "Nosecone only" shroud.

The final shroud that was analyzed for the MAWG is a dual-purpose shroud that would be used for both launch to LEO and Mars atmospheric entry (i.e., reinforced with TPS). The total length of this shroud is defined as 30 m (98.4 ft), including the transition cone. When used on the reference vehicle, the transition cone is approximately 1.4 m (4.69 ft) while the BL is 16.6 m (54.4 ft) and the nosecone is 12.0 m (39.3 ft). In addition, the OD is defined to be 10 m (32.8 ft), and the total mass is defined as 50,000 kg (110,000 lb$_m$). Any subsequent subdivision of mass is considered the responsibility of the shroud designer, as is the definition of the ID. This dual-purpose shroud can be seen in figure 5-10.

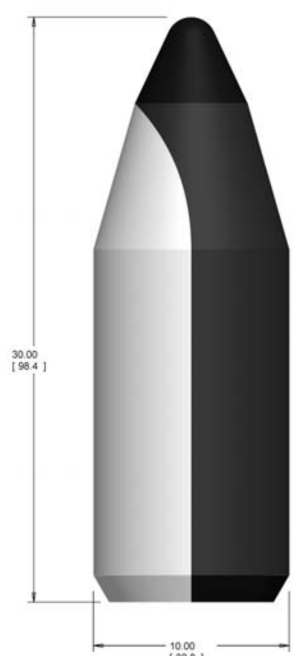



**Figure 5-10.** Dual-use shroud.

### 5.2.3 Mars mission low-Earth orbit analysis

The performance of shroud Options A, B, and C was analyzed on the lunar campaign reference vehicle to several LEOs, but some modifications (and other associated assumptions) to the vehicle are required to support the larger shroud options.

Firstly, if the reference vehicle is used for the Mars campaign, no TLI maneuver will have to be performed. For Mars missions, it is desired that maximum payload to LEO be delivered where subsequent assembly and TMI will occur. Therefore, the propellant loading in the EDS for maximum payload delivery to LEO is optimized individually for shroud Options A, B, and C. This value is required to be less than the maximum usable propellant load of 223,509 kg (492,753 $lb_m$) that the reference vehicle tanks are sized to hold. However, the tanks on the EDS are not resized to the calculated, optimal propellant load. Rather, it is assumed that a partial fill of the EDS tanks will be sufficient to deliver payload for Mars missions.

Secondly, the flight environment is impacted by the partial filling of the EDS tanks, different shroud sizes, and differing payload values from the reference vehicle. The trajectory is impacted slightly, thus resulting in different dynamic pressure values and flight loads. In addition, the aerodynamic profile of the vehicle is altered due to the larger shrouds. Table 5-6 shows the factors that were used for the potential aerodynamic impacts that were used in place of detailed computational fluid dynamics (CFD)/wind tunnel analysis of the final configuration. These values in effect increase the drag that is experienced by the vehicle, which was in addition to the further drag that was caused by the increase in aerodynamic reference area of the larger shrouds.

**Table 5-6.** Aerodynamic Impact Factors

| Aerodynamic Impact Factors | |
|---|---|
| Option A | 0% |
| Option B | 5% |
| Option C | 10% |

In addition, the structural support that is required to support the Option B and Option C shrouds is calculated. These larger, heavier shrouds impact almost every component along the outer mold line (OML) of the vehicle. Again, LVA is used to find the change in the structural components of the vehicle. Once performance is found for both the initial, non-structurally supported case and the structurally supported case, the resulting performance degradation factors are found. Seen in table 5-7, these factors are the potential structural scarring impacts for support of the alternate shroud configurations on the LEO payload.

**Table 5-7.** Resulting Payload-reduction Factors from Structural Support

| Structural Impact Factors | |
|---|---|
| Option A | 0% |
| Option B | 3.2% |
| Option C | 6.8% |

It was also assumed that a structural support system would be needed to integrate the payload with the launch vehicle. This system is calculated as 5% of the resulting LEO payload mass. This value is basically the payload adapter, but it also includes airborne support equipment and other systems that would be needed to support the payload during flight.

An LEO loiter package was not included on the EDS. Since a 14-day loiter package is included on the reference vehicle for its loiter period preceding CEV rendezvous, it was determined that the Mars campaign payloads would serve the function of rendezvousing with the “TMI module.” In this manner, no loiter package would be needed to maintain the spacecraft during a loiter period.

A flight performance reserve (FPR) was provided for the suborbital burn. The reference vehicle provides a 1% delta-V reserve for the injected mass. This reserve is accounted for in both the suborbital burn and the TLI maneuver. For the Mars campaign mission, only the suborbital portion is accounted for.

#### *5.2.3.1 Orbits*

The orbits that were analyzed are defined by the MAWG and shown in table 5-8. Both 222 km (120 nmi) and 300 km (162 nmi) are frequently analyzed for other Ares V concepts. The ISS orbital altitude is typically around 407 km (220 nmi) and 420 km (227 nmi). Finally, 750 km (405 nmi) and 1,000 km (540 nmi) are considered sufficiently high for launching systems that contain nuclear material. All orbits are direct circular injection orbits (i.e., no separate circularization burn).

**Table 5-8.** Orbital Altitude Summary

| Circular Orbit Altitude | |
|---|---|
| **km** | **nmi** |
| 222 | 120 |
| 300 | 162 |
| 407 | 220 |
| 420 | 227 |
| 750 | 405 |
| 1,000 | 540 |

#### *5.2.3.2 Shroud Options A, B, and C performance*

The trajectory of the reference vehicle with the Options A, B, and C shroud is optimized using the Program to Optimize Simulated Trajectories (POST) in three dimensions. Once all of the aforementioned performance factors have been considered, the gross payload to the six different circular orbits is found. The resulting performance is shown in metric units (figure 5-11) along with the associated EDS propellant load (figure 5-12). In addition, the same charts are provided in English units for convenience (figures 5-13 and 5-14).

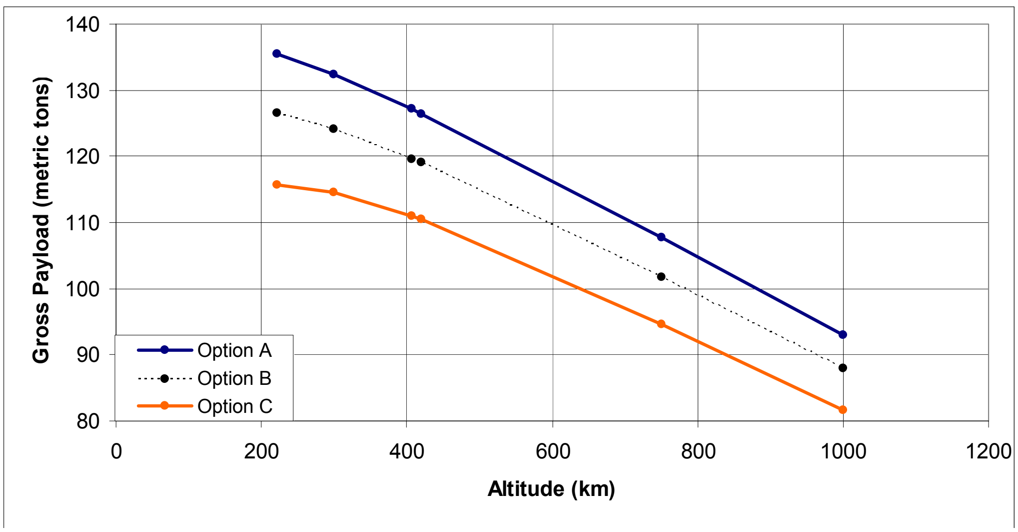



**Figure 5-11.** Reference vehicle performance vs. orbital altitude (metric).

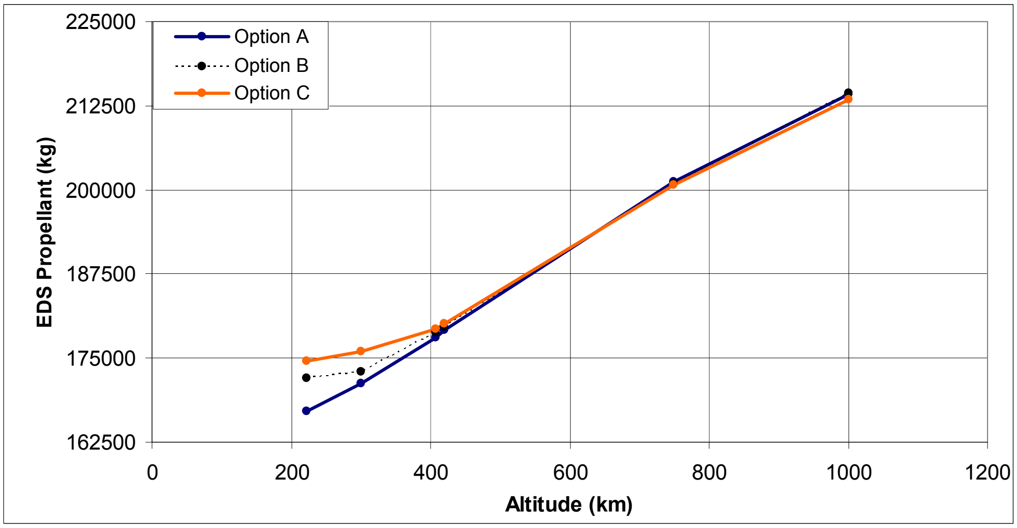



**Figure 5-12.** Earth departure stage propellant load vs. orbital altitude (metric).

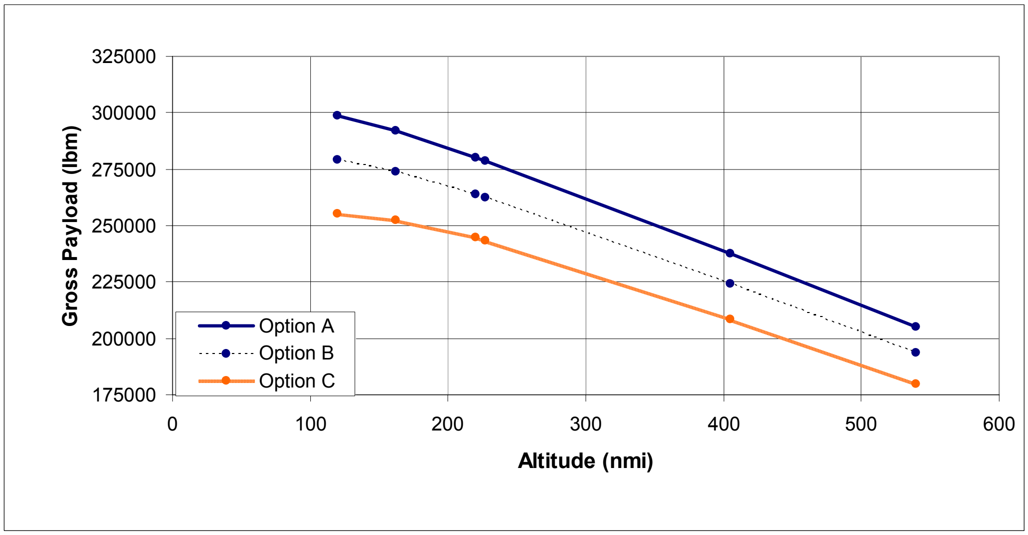



**Figure 5-13.** Reference vehicle performance vs. orbital altitude (English).

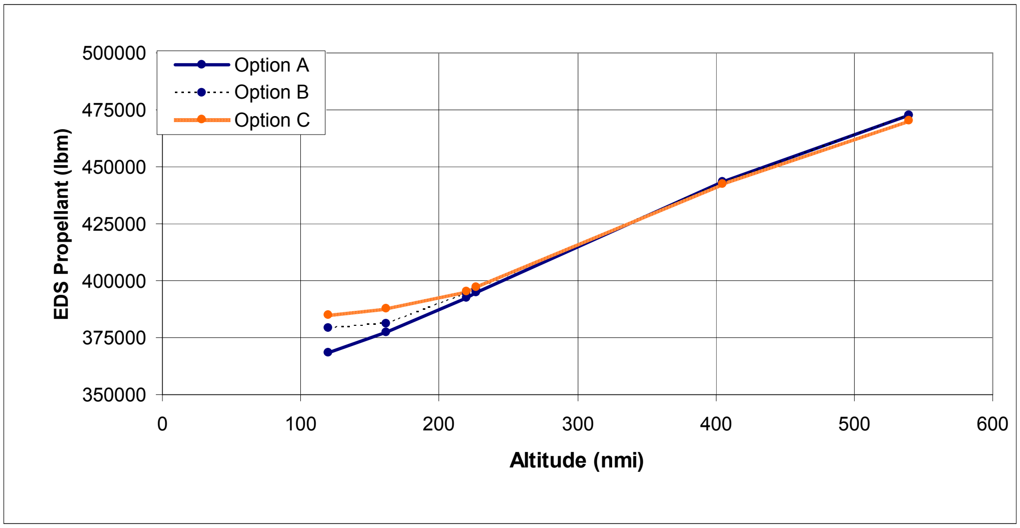



**Figure 5-14.** Earth departure stage propellant load vs. orbital altitude (English).

### *5.2.3.3 Mars fuel stage performance*

For the Mars fuel stage study, only a nosecone is used. Therefore, no aerodynamic impact or structural growth penalties are used, but a loiter package is included. This loiter package will sustain the EDS for however long it is required to remain in orbit prior to TLI. Estimates of capability also account for an in-space adapter mechanism and additional payload support equipment of 3,000 kg (6,600 $lb_m$). Finally, the Mars fuel stage is only flown to a 407-km (220-nmi) circular orbit. It can be seen in figure 5-15 that approximately 111,000 kg (246,000 $lb_m$) of fuel can be launched into a 407-km (220-nmi) circular orbit. When used for the lunar mission, the reference vehicle launches about 92,000 kg (203,000 $lb_m$) of propellant and an Altair with a mass of 40,500 kg (89,000 $lb_m$). However, the orbital altitude for this mission is only to 222 km (120 nmi). Therefore, not only is the performance impacted by being constrained with the reference vehicle tank size, but the orbital altitude is increasing as well.

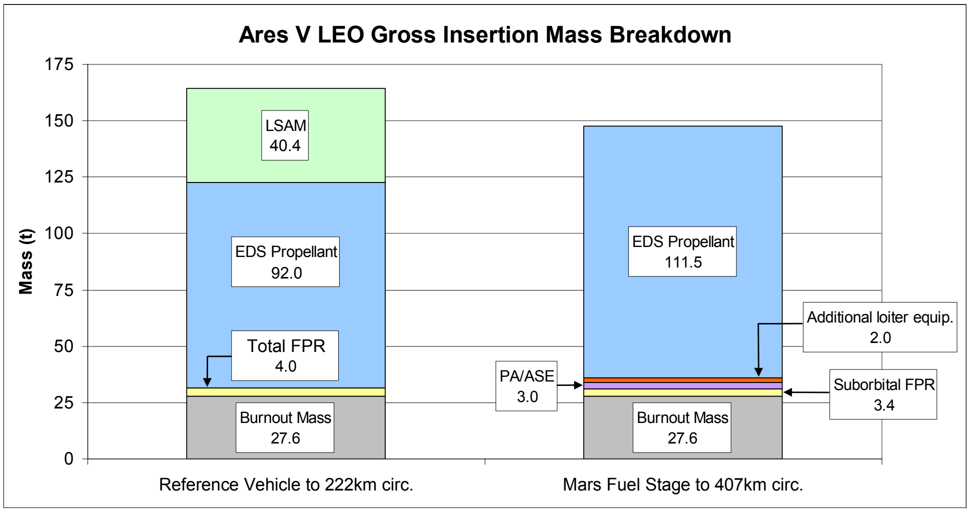



**Figure 5-15.** Mars fuel stage compared to lunar campaign reference vehicle.

### *5.2.3.4 Dual-use aero shroud performance*

When analyzing the performance of the dual-use aero shroud, the shroud is retained to a 407-km (220-nmi) circular orbit. In essence, the shroud itself is considered part of the LEO payload. In addition to the shroud mass of 50,000 kg (110,231 $lb_m$), a lander/ballast will be located inside of the shroud, in addition to any parachute assembly (PA)/ASE. Also included is an adapter section that would integrate the shroud with the reference vehicle. An aerodynamic impact factor of 5% is used, as well as a structural growth impact factor of 5.6%. Finally, a performance margin is held on the lander/ballast mass that is calculated as 10% of the resulting, maximized value. The performance of the dual-use aero shroud can be seen in figure 5-16.

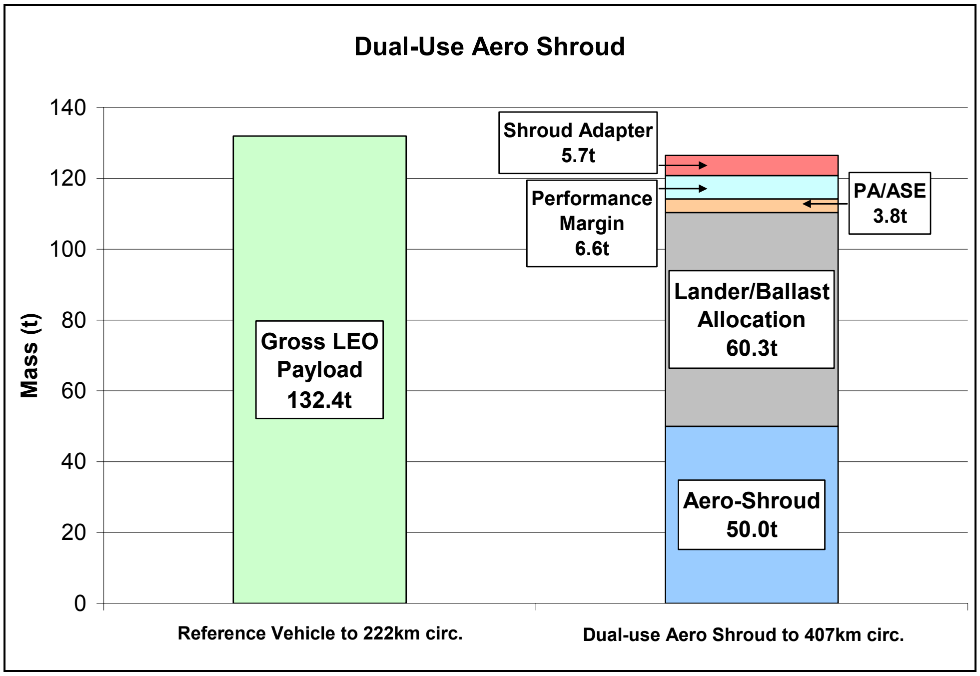



**Figure 5-16.** Dual-use aero shroud compared to lunar campaign reference vehicle.

It can be seen in figure 5-16 that the reference vehicle delivers approximately 132.4 t to LEO, while the dual-use aero shroud case delivers 126.5 t. The reduction in LEO payload is a result of the reference vehicle flying to a 222-km (120-nmi) circular orbit, while the dual-use aero shroud is flown to a 407-km (220-nmi) circular orbit.

### 5.2.4 Lunar/Mars mission synergism

Possible areas of synergism between the currently envisioned lunar campaign and the Mars campaign include common EDS designs, RSRB designs, core stage designs, shroud designs, and, obviously, the associated ground infrastructure. In turn, the total development time and cost for the Mars campaign will be significantly reduced. In addition, the total system reliability will be well defined due to the multitude of uses for the lunar campaign.

The current reference vehicle with minimal scarring has the potential to meet most of the architecture needs of the Mars campaign. Lunar campaign performance impacts would need to be assessed to examine whether the current vehicle needs to be altered to allow for Mars mission enhancements now or whether a block-upgrade, Mars-specific EDS would best suit the Mars architecture needs. All Ares V lunar performance enhancements will directly benefit the Mars architecture. Potential enhancements (minus cost assessments) are discussed in section 7.2.1.

## *5.3 Launch Processing*

This section describes the ground operations concepts that are required to process and launch the nuclear thermal and chemical variants of the crew and cargo MTVs at NASA KSC. The effects of launch vehicle configuration, number of launches, launch spacing, and crew and cargo MTV configuration were the focus of this study. The section begins with a brief description of the ground systems and ground operations concepts that are planned for Constellation lunar missions as of this writing (autumn 2007); more detailed descriptions of current ground systems can be found in official CxP documents). A high-level ground processing concept is then described for the Mars campaign. Notional processing timelines are provided from the time the flight hardware element arrives at KSC through launch. Modifications to infrastructure and requirements for additional infrastructure over the planned baseline for lunar missions are identified at a high level. The section concludes with the forward work that is suggested for additional study once the Ares V launch vehicle, Mars surface systems, and MTV configurations and ground processing requirements are further defined.

### 5.3.1 Ground operations lunar baseline

A phased approach is planned for the development of ground systems to support missions to the ISS and the moon. The first phase, which is referred to as "Block 1," includes the required capabilities to process and launch the Orion spacecraft and Ares I launch vehicle for missions to the ISS. The second development phase is referred to as "Block 2"; it includes additional capabilities to process the lunar lander (Altair) and the Ares V cargo launch vehicle. The ground operations lunar baseline consists of both Block 1 and Block 2 ground systems. High-level processing concepts and infrastructure descriptions are provided to offer a comparison between ground system requirements for lunar and Mars missions. Information that is provided in the following three subsections is meant to be a brief synopsis that is derived directly from CxP 72119, "Constellation Program Ground System Operational Concepts Document: (GS-OCD), and CxP 72197, "Constellation Program Ground System (GS) Architecture Description Document (ADD)."

### 5.3.2 Constellation Program ground systems architecture overview

The Constellation ground system consists of the physical support equipment, systems, and facilities that are required to perform services at the launch, landing, and retrieval sites in support of the Constellation missions. Functions provided by the ground system include receipt of flight hardware elements, software, cargo, and ground support equipment; spacecraft, launch vehicle, and cargo offline processing; spacecraft and launch vehicle integration; integrated testing; launch; recovery; search and rescue; logistics; and command, control, and communications in support of ground processing.

The ground system architecture is composed of eight elements, as shown in figure 5-17. Each element in the architecture represents an asset or a group of assets that provides functionality for the ground system.

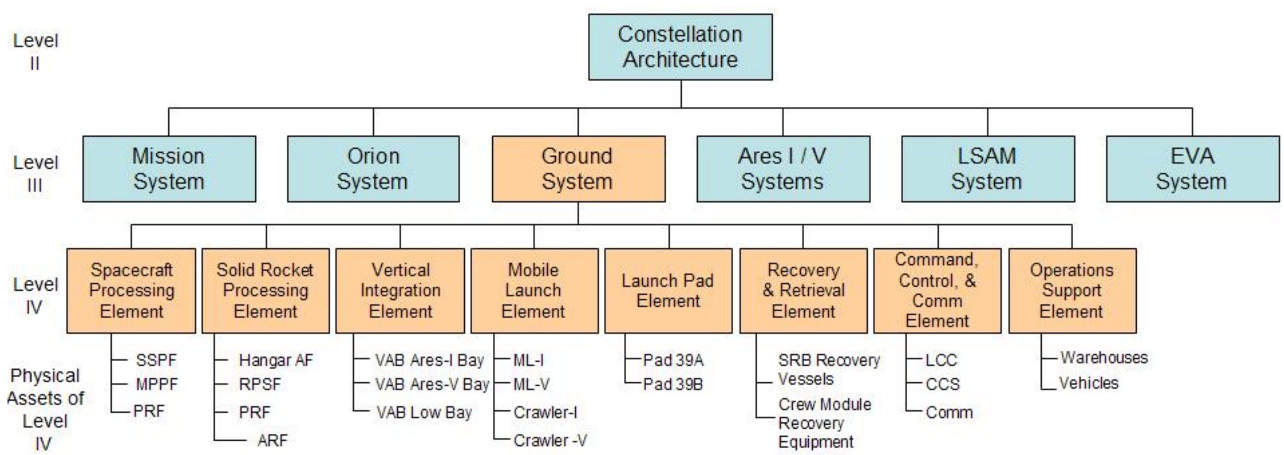



**Figure 5-17.** Ground system architecture.

The ground systems architecture is developed with a goal of minimizing work at the launch pad to ensure greater launch availability following the commitment to transfer the vehicle to the pad. In this architecture, the launch vehicle and spacecraft elements are processed in offline facilities and transported to a vertical integration facility for total vehicle assembly and integration before rolling to the pad via the crawler transporter. In the vertical integration facility (Vehicle Assembly Building (VAB)), the total vehicle (Ares I/Orion or Ares V/Altair) is assembled vertically on the mobile launch platform. Once the vehicle has been completely assembled, integrated with the ground system mobile launch element, functionally checked out, and initially serviced or preconditioned, the platform and vehicle are transferred to a launch pad element where final fluid servicing, including cryogenic propellant loading, launch countdown, flight crew ingress, and launch, are performed. Implementation of the ground system architecture is organized around GS subsystems. The GS subsystems, which include fluid, pneumatic, electrical, mechanical, facility, communication, and instrumentation systems, provide the capabilities that are needed to implement the GS elements.

#### *5.3.2.1 Block 1 – initial capability ground system architecture*

The initial capability ground system architecture supports flight testing of the Ares I/Orion vehicles and operational missions to the ISS. The Block 1 GS architecture relies on the conversion of existing space shuttle assets to provide a minimal set of ground processing and launch infrastructure. The ground system supports all flight systems and works in conjunction with other land-based support systems. The relationships of the ground system to the other Constellation systems are summarized below:

- The Ares I System uses the ground system for vehicle receipt, offline processing, integration, servicing, launch processing, recovery, and refurbishment.
- The Orion system interfaces with the ground system for spacecraft receipt, offline processing, vertical integration and checkout with the Ares I vehicle, servicing, launch, recovery and retrieval, and logistical support.
- The mission system provides mission-specific data products that are used to configure the ground system.
- The EVA system uses the ground system for pre-flight test, checkout, servicing, and compatibility for purposes of crew transport and recovery, crew ingress, and emergency egress.

#### *5.3.2.2 Block 2 – lunar capability ground system architecture*

For Constellation lunar missions, the ground systems architecture will be modified to provide the facilities and systems that are necessary to process the cargo launch vehicle (Ares V) and Altair. Block 2 will build on Block 1 with several notable additions. The receipt, processing, integration, and launch of the Ares V and Altair will occur with this architecture. Recovery of Ares V assets will be required. Additionally, a multi-element integration test (MEIT) will be required between the Orion and Altair to verify interface compatibility before the two spacecraft are mated for the first time in LEO.

- The Orion system continues to use the ground system for receipt, processing, vertical integration and checkout with the Ares I vehicle, launch, recovery and retrieval, and logistical support.
- The Ares I/Ares V systems use the ground system for vehicle receipt, processing, integration, launch processing, recovery, and refurbishment.
- The mission system provides mission-specific data products that are used to configure the ground system.
- The EVA and Altair system use the ground system for pre-flight test, checkout, and integration.
- The surface system provides a recharge station for the suit portable life support system and any required donning equipment

### 5.3.3 Mars campaign ground processing concepts

#### *5.3.3.1 Mars campaign ground operations assessment*

The ground operations assessment for the Mars campaign evaluated the infrastructure changes that are required to support 30-day launch centers for NTR propulsion (Case 1) and chemical propulsion (case 3). The assessments focused primarily on Ares V launch vehicle processing requirements. At the time of this writing, the Ares V launch vehicle ground operations concepts are still in the very early stages of development. Several Ares V vehicle concept trades are under way that eventually will likely affect facility usage requirements as well as ground operations timelines. A "Ship to Integrate" ground processing concept was assumed for the Ares V Core and some MTV elements. A Ship to Integrate concept assumes that very limited ground processing activities are required at the launch site to prepare flight hardware for processing. The flight hardware is essentially unloaded from the transporter, inspected for damage, and stacked directly on the mobile launcher. No provisions are made for the long term storage of the element or for significant repair capabilities at the launch site (i.e., changing an engine). The ground processing descriptions below only show the changes that are required from the lunar baseline.

As of this writing, the current CxP flight rate requirements for Ares V are four flights per year with a "surge"[8] to six flights per year. The minimum launch spacing requirement is 45 days (TBR). Numerous trade studies are under way as part of ongoing lunar architecture concept development work to resolve these and many other aspects of

[8]Surge implies that the six flights per year occur infrequently (not in consecutive years).

future lunar missions. The current budget baseline (FY2008) for the lunar campaign includes one Ares V VAB Integration Bay and one Ares V mobile launcher. For both Case 1 and Case 3, launch spacing between consecutive Ares V launches is assumed to be 30 days. The timelines that were used for the analysis are based on existing knowledge of space shuttle solid rocket booster (SRB) processing procedures, more detailed timelines developed for the Ares I, and historical data for assembling and integrating large boosters.

#### *5.3.3.2 Ares V facility and ground systems options*

To accommodate the minimum launch spacing and number of launches per year, the following new ground systems and facilities were evaluated:

- Up to two additional VAB high bays[9]
- Up to four additional Ares V mobile launchers
- New launch mount[10]: Allows an SRB to be stacked in a stacking cell in the VAB or in an offline stacking facility (OSF) (figure 5-18), thus removing the SRB stacking process from the primary critical path resource – the mobile launcher. The launch mount is required for the SRB processing options that are described below: VAB offline stacking cell (figure 5-19), VAB offline stacking bay; and OSF.

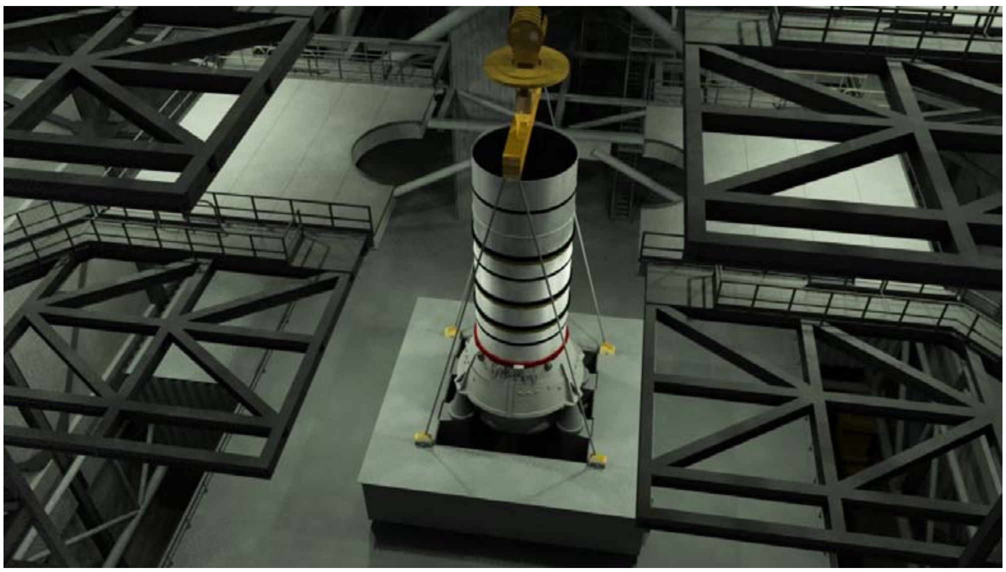


**Figure 5-18.** Launch mount concept.

[9]There are four VAB high bays (also called integration cells or integration bays). The high bays are used to stack and integrate the launch vehicle on the mobile launchers. The lunar baseline includes one VAB high bay for Ares I SRB stacking and vehicle integration and one VAB high bay for Ares V stacking and vehicle integration. The remaining two high bays are unused and will require modifications to platforms to process the Ares V or to be used as an offline stacking bay.

[10]The benefits of the launch mount were presented to the CxP in 2007. The program approved the launch mount option for Ares I. The addition of an offline stacking cell was not approved.

**Figure 5-19.** Solid rocket booster offline stacking cell adjacent to the VAB integration cell.

- *Offline stacking cell*: SRB stacking occurs on a pedestal that is located in the same cell in which vehicle integration will occur. The SRB stack is then transferred to the mobile launcher for any remaining vehicle integration. This provides the capability to remove SRB stacking and closeouts from the mobile launcher critical path. This does not address SRB quantity-distance constraints in the VAB.
- *Offline stacking bay*: Offline stacking and vehicle integration occur in separate VAB high bays. The SRBs are stacked and moved to the mobile launcher. This does not address SRB quantity-distance constraints in the VAB (figure 5-20).

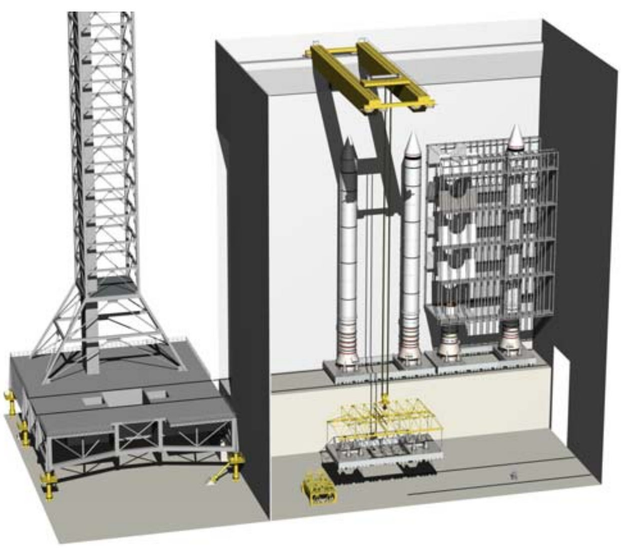


**Figure 5-20.** Solid rocket booster offline stacking facility.

- *Offline stacking facility*: New facility that is dedicated to stacking boosters for Ares I and Ares V. Facility will include two stacking bays with the capability to stack two boosters inside each bay. SRBs are stacked on a launch mount in the facility. Once SRBs are stacked, the mobile launcher will be moved into the facility. The SRBs will then be translated onto the mobile launcher, and the mobile launcher will be moved back to the VAB to continue the buildup and integration of the Ares V. This option does address SRB quantity-distance constraints in the VAB.

- Vertical Integration Facility (VIF): New facility for Ares V buildup and integration with payload (MTV elements, MAV, etc.). The VAB essentially becomes an SRB OSF. This option does address SRB quantity-distance constraints in the VAB by moving vehicle integration to the VIF (figure 5-21).

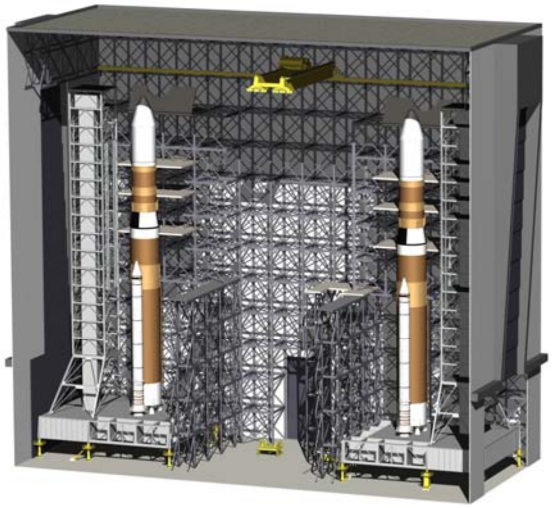


**Figure 5-21.** Ares V Vertical Integration Facility.

### 5.3.3.3 *Flight rate and minimum launch spacing analysis and results*

Element processing schedules were developed for each ground systems configuration that was described in the previous sections.

Based on the task durations that are included in figure 5-22, each ground system configuration was assessed to determine maximum annual flight rate minimum as well as minimum launch spacing for the Ares V.

NOMINAL DURATIONS

**Nominal**

| Task: | Critical Path Work Days (WD): |
|---|---|
| ML Pad Refurb: | 7 |
| VAB ML Preps: | 5 |
| SRB Stacking: | 23 |
| Core: | 7 |
| EDS: | 5 |
| Lander: | 2 |
| Integrated Test: | 6 |
| Ares V-Pad: | 6 |
| PAD Refurb: | 0 |
| VAB: Lockout: | 3 |
| ML Downtime: | 180 |
| VAB Downtime: | 90 |
| PAD Downtime: | 90 |

**Launch Mount**

| Task: | Critical Path Work Days (WD): |
|---|---|
| ML Pad Refurb: | 7 |
| VAB ML Preps: | 2 |
| SRB Stacking: | 23 |
| Core: | 7 |
| EDS: | 5 |
| Lander: | 2 |
| Integrated Test: | 6 |
| Ares V PAD: | 6 |
| PAD Refurb: | 0 |
| VAB Lockout: | 3 |
| ML Downtime: | 180 |
| VAB Downtime: | 90 |
| PAD Downtime: | 90 |

**Offline Stacking Cell**

| Task: | Critical Path Work Days (WD): |
|---|---|
| ML Pad Refurb: | 7 |
| VAB ML Preps: | 2 |
| SRB Transfer: | 2 |
| SRB Offline Stacking: | 25 |
| Core: | 5 |
| EDS: | 5 |
| Lander: | 2 |
| Integrated Test: | 6 |
| Ares V PAD: | 6 |
| PAD Refurb: | 0 |
| VAB Lockout: | 3 |
| ML Downtime: | 180 |
| VAB Downtime: | 90 |
| PAD Downtime: | 90 |

**Offline Facility**

| Task: | Critical Path Work Days (WD): |
|---|---|
| ML Pad Refurb: | 7 |
| VAB ML Preps: | 2 |
| Core: | 5 |
| EDS: | 5 |
| Lander: | 2 |
| Integrated Test: | 6 |
| Ares V PAD: | 6 |
| PAD Refurb: | 0 |
| VAB Lockout: | 3 |
| ML Downtime: | 180 |
| VAB Downtime: | 90 |
| PAD Downtime: | 90 |

**Offline Stacking Facility**

| Task: | Critical Path Work Days (WD): |
|---|---|
| Offline ML Preps (CLV): | 2 |
| FS Stacking: | 15 |
| Transfer (CLV): | 1 |
| Offline ML Preps (CaLV): | 2 |
| SRB Stacking: | 26 |
| Transfer (CaLV): | 2 |
| Lockout: | 0 |
| OSF Downtime: | 90 |

**CaLV VIF (with VAB Offline Stacking Bay)**

| Task: | Critical Path Work Days (WD): |
|---|---|
| ML Pad Refurb: | 7 |
| VIF ML Preps: | 2 |
| Core: | 5 |
| EDS: | 5 |
| Lander: | 2 |
| Integrated Test: | 6 |
| VIF Servicing and C/O: | 4 |
| Ares V PAD: | 2 |
| PAD Refurb: | 0 |
| VIF Lockout: | 0 |
| ML Downtime: | 180 |
| VAB Downtime: | 90 |
| PAD Downtime: | 90 |
| VIF Downtime: | 90 |

**Offline Stacking Bay - VAB**
(Included in VIF scenario)

| Task: | Critical Path Work Days (WD): |
|---|---|
| Offline ML Preps (CLV): | 2 |
| FS Stacking: | 15 |
| Transfer (CLV): | 1 |
| Offline ML Preps (CaLV): | 2 |
| SRB Stacking: | 26 |
| SRB Transfer (CaLV): | 2 |
| Lockout: | 3 |
| Bay Downtime: | 90 |

**CLV VIF (with VAB Offline Stacking Bay) (Reference)**

| | |
|---|---|
| VIF ML Preps: | 2 |
| US Mate & C/O: | 2 |
| CEV/LAS Installation: | 3 |
| CLV/CEV Integrated Test | 7 |
| VIF Servicing and C/O: | 4 |

**Figure 5-22.** Nominal processing durations.

Two flight rates are shown: "No MOD" and "w/MOD" (i.e., modification [MOD]). No MOD indicates when a major ground system and facility resources are not taken out of service for maintenance and refurbishment for a given year. Similarly, "w/MOD" includes estimated maintenance downtime periods, as shown in figure 5-23. As expected, including the downtime that is required for maintenance periods reduces annual flight rate capabilities for each option. Typically, large systems such as mobile launchers and the pad need to be taken out of service approximately every 3 years for corrosion control and other periodic maintenance procedures. Down periods can last 3 to 6 months. It is possible for a Mars launch campaign to support a TMI without being impacted by a MOD period.

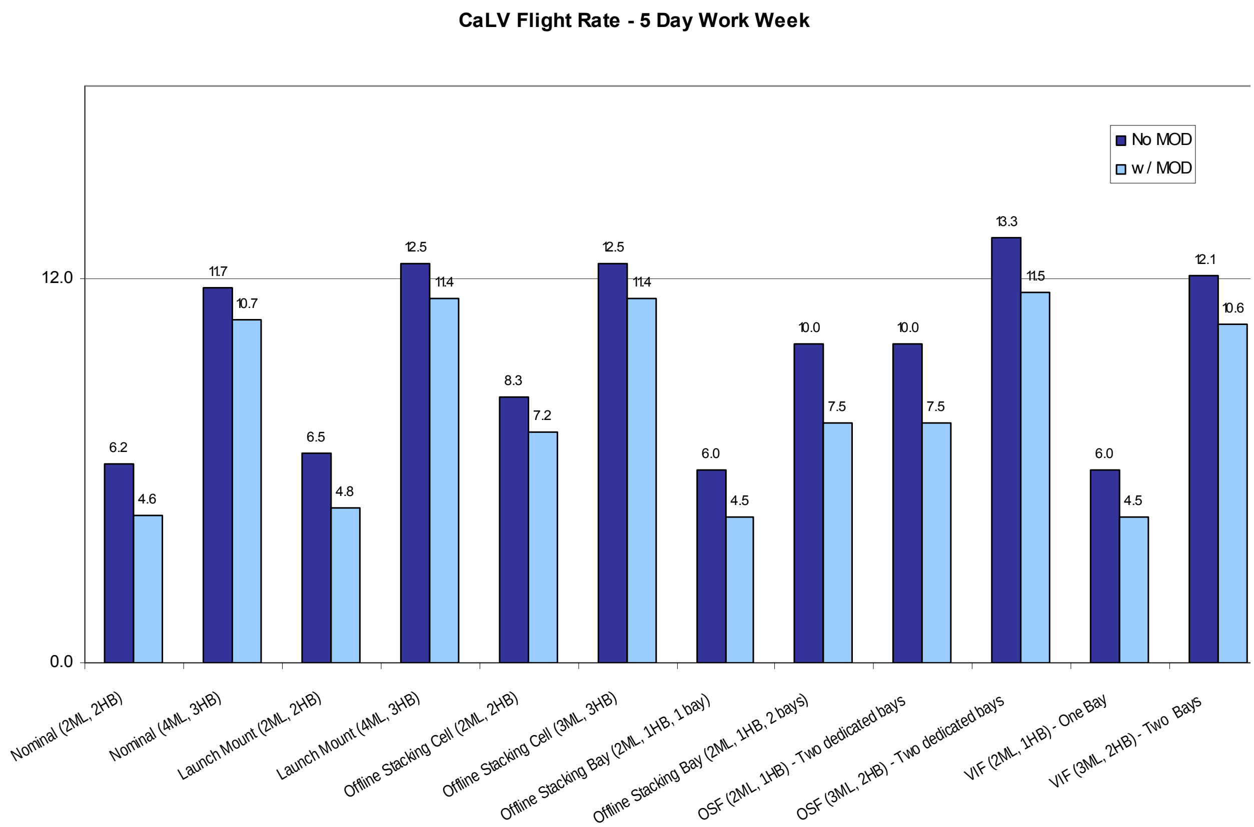



**Figure 5-23.** Annual flight rate results.

For the launch spacing assessment, three work schedules were assessed:

- *Standard*: Sustainable launch spacing based on a 5-day work week.
- *Minimum*: Shortest launch spacing that could be achieved at a certain point in time based on a 5-day work week. This is essentially a short-term surge capability and is not sustainable.
- *Compressed minimum*: Same as minimum, but based on a 7-day work week. Launch rate is not sustainable.

As shown in figure 5-24, seven out of 12 configurations meet the 30-day launch spacing requirements. When the results of the minimum annual flight rate and 30-day launch spacing results are combined, the following configurations are viable options to support the Mars launch campaigns:

1. OSF with three mobile launchers and two high bays
   - 13.3 flights – 22-day launch spacing
2. Launch mount with four mobile launchers and three high bays
   - 12.5 flights – 23-day launch spacing

3. Offline stacking cell with three mobile launchers and three high bays
    - 12.5 flights – 23-day launch spacing
4. VIF with three mobile launchers, two high bays, and two integration bays
    - 12.1 flights – 24-day launch spacing

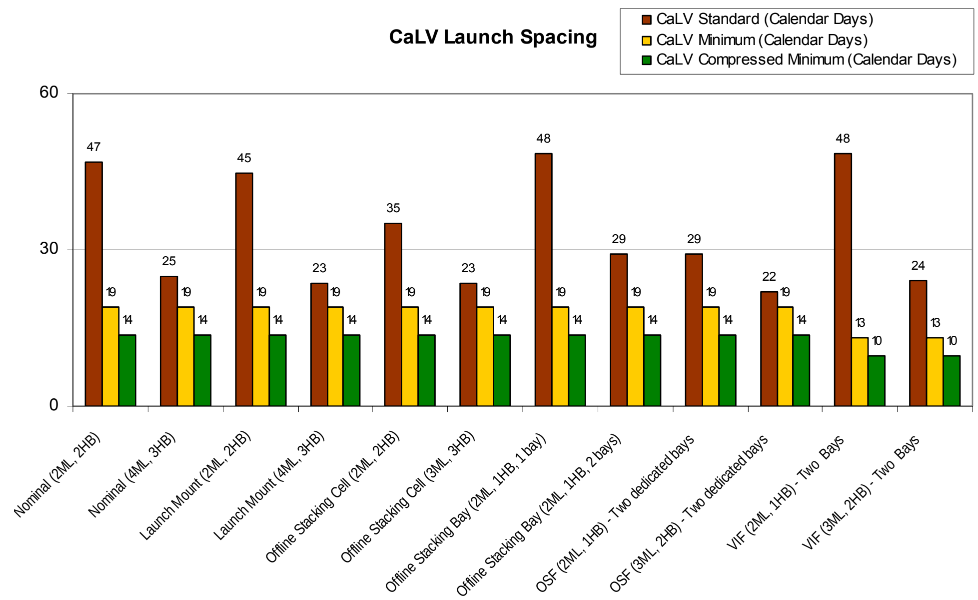



**Figure 5-24.** Day launch spacing results.

For Constellation lunar missions, discrete event simulation models indicate that operations in the VAB will likely be the limiting factor in determining minimum launch spacing and annual launch rates. Conflicts in the VAB transfer aisle during lifting operations as well as SRB quantity distance issues decrease the desirability of Options 2 and 3. High-level trade studies indicate that the cost of the VIF will likely be higher than that of the OSF. Therefore, the OSF (Option 1) was chosen to develop overall processing times for the Mars launch campaigns that are shown in figures 5-25 and 5-26.

Figures 5-25 and 5-26 show over processing flow timelines in weeks and approximate ground system utilization for NTR and chemical launch campaigns. For the NTR case, it was assumed that a new nuclear processing facility (NPF) was required to process the core stages that contain nuclear material and require special processing considerations that are not planned in the lunar baseline. For both the chemical and the nuclear cases, a new Hazardous Processing Facility (HPF) is assumed to process spacecraft and MTV elements prior to integration with the launch vehicle in the VAB. The facility requirements are driven primarily by the size of the spacecraft elements and the desire to perform as much spacecraft processing as possible prior to critical path operations in the VAB and on the pad.

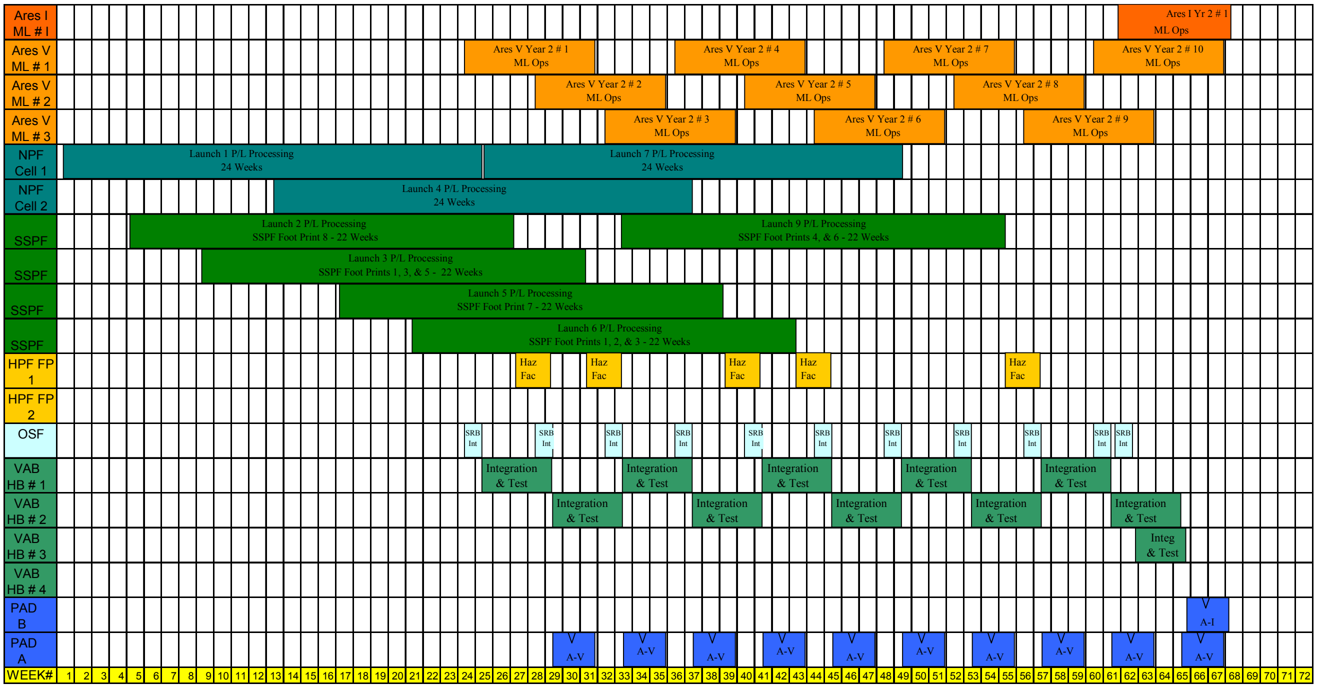



**Figure 5-25.** Kennedy Space Center nuclear thermal rocket launch campaign.

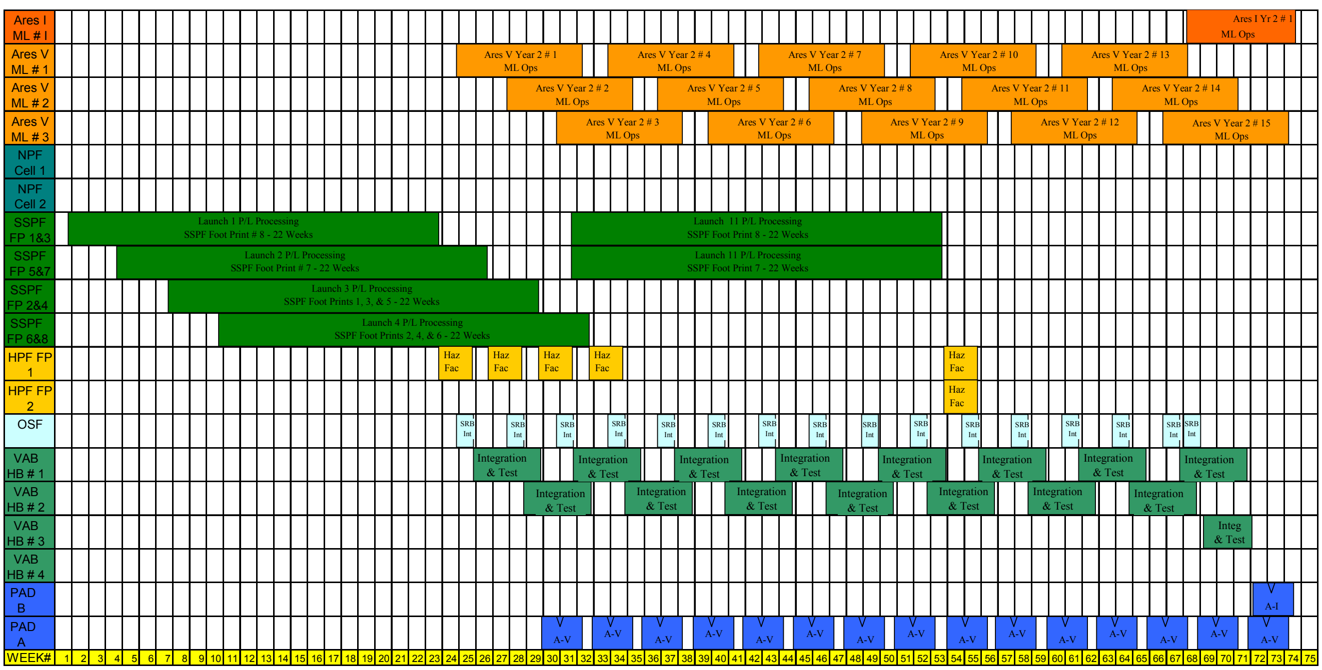



**Figure 5-26.** Kennedy Space Center chemical launch campaign.

### 5.3.3.4 Mars campaign ground processing concepts

**Mars ascent vehicle, descent vehicle, and transit habitat ground processing concept**

The hardware configurations for the ascent vehicle, descent vehicle, and transit habitat were not the focus of this DRM update; therefore, detailed ground processing requirements were not available. High level assumptions were made for spacecraft processing based on experience with space station, including element processing methods and initial plans for processing the Altair. All elements were assumed to require a clean work area and some form of hazardous processing. It was also assumed that 6 months' processing time was required in the Space Station

Processing Facility (SSPF). Following nonhazardous processing in the SSPF, the element is transferred to a new HPF[11] that is to be serviced and encapsulated in the Ares V aeroshell or shroud. The servicing and encapsulation processes were estimated to take approximately 2 weeks. Following servicing and closeouts, the encapsulated spacecraft are transported to the VAB to integrate with the Ares V launch vehicle as shown in figures 5-27 and 5-28 below.

**Figure 5-27.** Mars spacecraft encapsulated by the aeroshell in the transfer aisle of the VAB.

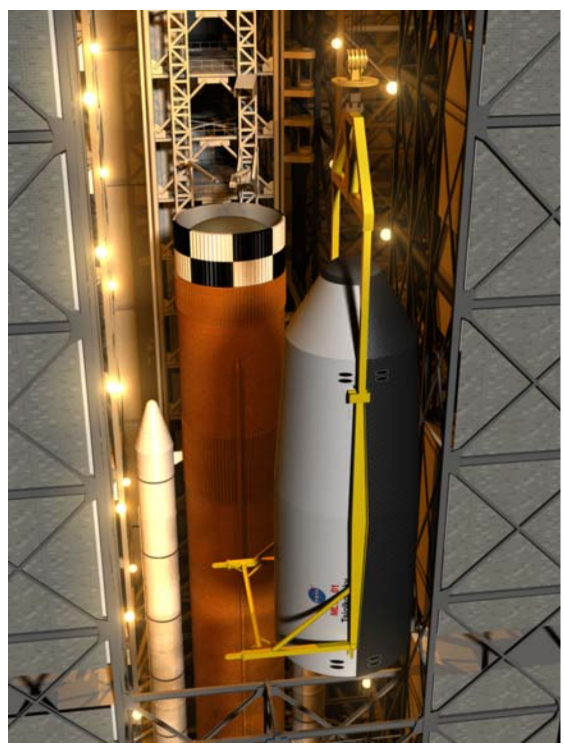


**Figure 5-28.** Mars Aeroshell and Payload Lifting Operations in the Vehicle Assembly Building.

[11]A new HPF was baselined for this study because it is desirable to perform as much flight hardware processing as possible before rolling to the pad for several reasons. There is extremely limited access at the pad to perform repairs. Contingency repair analyses that were performed for Orion indicate that as much as 65% of potential LRUs are not accessible at the pad. Any repairs to the spacecraft would require a roll back, thus significantly impacting the Mars launch campaign.

**Chemical Mars transit vehicle processing**

The configuration for the chemical crew and cargo MTVs is described in section 5.4. A chemical MTV element is shown arriving at the KSC turn basin in figure 5-29. Reboost modules were assumed to require some level of spacecraft processing in the SSPF. The TMI, TEI, and MOI modules were considered to be similar to the Ares V EDS that is planned for lunar missions. Therefore, the Ship to Integrate approach was used to develop ground processing timelines. TMIs, TEIs, and MOIs arrive at KSC via barge, are unloaded, and immediately are transported to the VAB. Covers and other restraining devices that are used during shipping are removed. The element is inspected for damage, and lifting ground support equipment (GSE) is attached to the element to support mating operations with the VAB. The Ares V shroud is then installed on the Ares V, thus encapsulating the MTV elements.

**Figure 5-29.** A chemical Mars transit vehicle element arriving at NASA Kennedy Space Center.

**Nuclear thermal rocket Mars transfer vehicle processing**

The configuration for the NTR crew and cargo MTVs is described in section 5.4. The NTR core stages contain significant amounts of nuclear material that require special ground processing considerations related to security and worker safety. Therefore, the ground processing concept for the core stages assumes a new NPF with two processing cells. All elements arrive at the launch site via barge and are transported to the SSPF, to the NPF, or directly to the VAB. An MTV NTR core stage is shown being lifted in the VAB transfer aisle for mating operations with the Ares V in figure 5-30. Prelaunch processing for the saddle truss and inline $LH_2$ tanks is performed in the SSPF. A Ship to Integrate philosophy was assumed for the $LH_2$ drop tanks. The $LH_2$ drop tanks arrive at NASA KSC via ship, are unloaded, and are immediately transported to the VAB. Covers and other restraining devices that were used during shipping are removed. The element is inspected for damage, and lifting GSE is attached to the element to support mating operations with the VAB.

**Figure 5-30.** Nuclear thermal rocket core stage lifting operations in the Vertical Assembly Building.

**Ares V vehicle processing and launch operations**
The 30-day spacing for the Ares V launches requires ground processing for the Ares V core stage and SRBs followed the same concept as was described for the lunar missions (see relevant CxP documents). Thirty-day launch spacing requires two additional mobile launchers, an additional VAB Ares V integration cell, and an OSF. The OSF allows the SRB segments to be stacked on a mobile launcher, thereby allowing Ares V and Ares V payload integration activities to proceed without violating the quantity distance constraints in the VAB. Once integrated tests in VAB are complete, final closeouts are performed and the vehicle is rolled to the pad (see figure 5-31). Launch operations proceed as described the Ground Operations Lunar Campaign section for Ares I/Orion and the Ares V.

**Figure 5-31.** Mars aeroshell/spacecraft and Ares V prior to launch on Launch Complex 39-A.

### *5.3.3.5 Mars campaign ground processing forward work*
Ground processing concepts for this study focused primarily on launch vehicle facility and infrastructure impacts to meet 30-day launch spacing requirements. The results only provide very-high-level insights into the changes that are required at the launch site to support the proposed Mars campaign above what is planned to support lunar campaigns. Several areas warrant additional study:

- As this assessment was nearing completion, detailed studies were under way to determine the launch site impacts of changing minimum scrub turnaround times for an Ares V from 26 days to 6 days and of changing the launch spacing between the Ares V/Altair and Ares 1/Orion from 24 hours to 90 minutes. The studies indicate potential issues with national cryogenic production capabilities and the effects of limited storage capabilities at the launch pads. Follow-on mitigation studies are planned that include examining the benefits of building a new on-site poly-generation plant ($H_2$ and $O_2$), thereby minimizing losses during transportation and between launch scrubs[12] through technology improvements and building additional storage capacity at Launch Complex 39 (LC-39).

- The Ship to Integrate philosophy is driven to a large degree by the desire to reduce costs at the launch site by minimizing special ground processing as much as possible[13]. Additional studies should be conducted to determine the effects of ground operations if a Ship to Integrate philosophy is not used for the Ares V core and chemical TMI, MOI, and TEI stages. New offline processing facilities may be required due to the physical size and number of MTV elements.
- A high-level ground processing concept was developed in 2007 to support a surface nuclear power system study for the lunar outpost. This study was the basis for the assumptions that were made for processing the MTV NTR stages. Additional work is required to understand ground processing required for the NTR stages that are proposed for the MTV.
- The Mars campaign assumed a launch schedule margin of an additional 3 to 6 months prior to the TMI window opening to account for launch delays that may be caused by problems with the flight hardware at KSC or weather delays. The impacts of launch delays on the entire "supply-chain" from manufacturing and test facilities all the way through launch at KSC need to be further explored. This study assumed "just-in-time" delivery for launch vehicle hardware. As noted, a launch campaign consisting of up to 12 Ares V launches in a row will likely experience some delays due to hardware problems or weather. Storage capabilities at the launch site and the launch vehicle manufacturing facility need to be assessed to determine potential storage requirements of the Ares V core stages, SRB segments, MAV/descent vehicle, SHAB, and TransHab as well as the elements of the MTV.
- Ares V shroud operations in the VAB require additional study. At the time of this writing, Ares V shroud studies are under way. The shroud architecture along with clean work area requirements for Altair will determine the feasibility of performing payload shroud encapsulation in the VAB. The study may result in requirements for a new shroud-encapsulation facility or significant modifications to the VAB to provide a clean work area in which to perform encapsulation operations in the VAB.
- To support minimum launch spacing requirements, innovative means of stacking and integrating Ares V launch vehicles will be needed. One means of shortening the mobile launcher flow, reducing the number of mobile launchers, and mitigating VAB stacking restrictions is the SRB "offline stacking" concept that was described earlier in this section. This concept would allow the SRB units to be assembled separately and transferred to the mobile launcher, likely shortening the critical path buildup from 5 weeks to 1 day. To do this, the Ares V mobile launcher will have to be outfitted with two removable "launch mounts" that would allow the SRB to be stacked offline. The launch mounts will have to be incorporated into the mobile launcher design, and will have to provide clearance to the core stage RS-68 engines. As this study was nearing completion, new Ares V variants were proposed for the lunar phase of the program. Some of these variants included an additional RS-68 engine for the Ares V core stage. Detailed study is required to ensure that the Ares V ML design can accommodate the SRB launch mounts while providing sufficient required clearance to eliminate interferences with the core stage RS-68s and servicing umbilicals.

## *5.4 Interplanetary Transportation*

### 5.4.1 Nuclear thermal propulsion option

The NTR is a leading propulsion system option for human Mars missions because of its high thrust (10's of pounds force ($klb_f$))/high $I_{sp}$ ($I_{sp}$ ~875-950 s) capability, which is twice that of today's $LO_2/LH_2$ chemical rocket engines. Demonstrated in 20 rocket/reactor ground tests during the Rover/ Nuclear Engine for Rocket Vehicle Applications (NERVA) Programs (Koeing, 1996[14]), the NTR uses fission-reactor-generated thermal power rather than chemical combustion of an oxidizer-fuel mixture, to directly heat $LH_2$ propellant for rocket thrust. NASA's previous Mars DRM studies – DRM 3.0 in 1998 (Borowski, et al., 1998[15]), (Drake, 1998[16]) and DRM 4.0 in 1999 – used a

[12]Up to 20% of $H_2$ is lost during transportation from $H_2$ production facilities in Louisiana. Up to 40% of the $H_2$ that is pumped into the tank is lost during a launch scrub turnaround (vehicle is tanked and drained back into the cryo spheres).

[13]The same approach was planned for ISS elements. As the program matured, it was determined that additional testing was required at the launch site.

[14]D.R. Koeing, "Experience Gained from the Space Nuclear Rocket Programs (Rover/NERVA)," LA-10062-H, Los Alamos National Laboratory (May 1986).

[15]S.K. Borowski, L.A. Dudzinski and M.L. McGuire, "Vehicle and Mission Design Options for the Human Exploration of Mars/Phobos Using "Bimodal" NTR and LANTR Propulsion," AIAA-98-3883, American Institute of Aeronautics and Astronautics (July 1998) and NASA/TM – 1998-208834 (Dec. 1998).

“common” propulsion module with three 15,000 $klb_f$ NTR engines. The use of clustered, lower-thrust (~15–25 $klb_f$) engines provides an “engine-out” capability that can increase crew safety and reduce mission risk. The time and cost to develop and ground test these smaller engines is also expected to be less then that required for higher-thrust engines. Both conventional NTR engines (thrust-only) and “bimodal” engines (bimodal nuclear thermal rocket (BNTR)), which are capable of producing both thrust and modest amounts of electrical power (few 10’s of $kW_e$) during the mission coast phase, were examined in addition to “0-$g_E$” and AG crewed MTV design concepts. The current Mars DRA 5.0 Phase 1 and 2 study efforts to date have considered “thrust-only” NTR engines, “0-$g_E$” crewed MTV designs, and photovoltaic arrays (PVAs) to supply spacecraft electrical power.

This section of the Mars DRA 5.0 report reviews mission, engine, and vehicle design considerations and assumptions, presents Phase 1 and 2 study results, and summarizes findings. The operating principles of a NERVA-based engine are outlined first, then the performance characteristics of a 25-$klb_f$ “Pewee-class” engine, which was baselined in this study, are presented. NTP-specific mission and transportation system ground rules and assumptions are then reviewed, followed by a brief description of the “reference” NTP long-surface-stay Mars mission scenario and MTV LEO assembly operations. Design features and operating characteristics for Phase 1 and 2 cargo and crewed MTVs are discussed afterwards along with proposed design changes for the crewed vehicle that are being adopted for Phase 3. The section concludes with a summary of key findings and recommendations on required future work.

#### *5.4.1.1 System description and performance characteristics*

As mentioned in the introduction, the NTR uses a compact fission reactor core containing “enriched” $^{235}U$ fuel to generate the large quantities of thermal power (100’s of $MW_t$) that are required to heat the $LH_2$ propellant to high exhaust temperatures for rocket thrust. In a typical “expander cycle” NERVA-type engine (figure 5-32), high-pressure $LH_2$ flowing from twin turbopump assemblies (TPAs) cools the nozzle, pressure vessel, neutron reflector, and control drums of the engine and, in the process, picks up heat to drive the turbines. The turbine exhaust is then routed through the core support structure, internal radiation shield, and coolant channels in the fuel elements of the reaction core where it absorbs energy from the fissioning $^{235}U$ atoms, is superheated to high exhaust temperatures ($T_{ex}$ ~2550–2800 K depending on fuel type and uranium loading), and is expanded out to a high-area-ratio ($\varepsilon$~300:1-500:1) nozzle for thrust generation. Controlling the NTR during its various operational phases (startup, full thrust, and shutdown) is accomplished by matching the TPA-supplied $LH_2$ flow to the reactor power level. Multiple control drums, which are located in the reflector region surrounding the core, regulate the neutron population and reactor power level over the operational lifetime of the NTR. The internal neutron and gamma radiation shield, which is located within the pressure vessel of the engine, contains its own interior coolant channels. It is placed between the reactor core and key engine components (e.g., TPAs) to prevent excessive radiation heating and material damage.

A NERVA-derived engine uses a “graphite matrix” material fuel element (FE) containing the $^{235}U$ fuel in the form of uranium-carbide ($UC_2$) microspheres or as a dispersion of uranium carbide and zirconium carbide ($UC_2$-ZrC) within the matrix material, which is referred to as “composite” fuel. The typical NERVA FE has a hexagonal cross section (~0.75 in. across the flats), is 52 in. long, and produces approximately 1 MW of thermal power. Each FE has 19 axial coolant channels, which along with the exterior surfaces of the element, are chemical vapor deposition (CVD)-coated with ZrC to reduce $H_2$ erosion of the graphite. Composite fuel, with its higher temperature capability ($T_{ex}$~2,550–2,800 K), was the preferred fuel form at the end of Rover/NERVA program, and is used here. The performance characteristics for the 25-$klb_f$ NTR that was baselined in this study include: $T_{ex}$~2,650–2,700 K, $p_{ch}$ ~1,000 psi, $\varepsilon$ ~300:1, and $I_{sp}$ ~900–910 s. At an approximately 900-second $I_{sp}$, the $LH_2$ flow rate is about 12.6 kg/s. The thrust-to-weight ratio for a dual-TPA, expander cycle 25-$klb_f$ engine is approximately 3.43. The overall engine length is about 7.01 m, which includes an approximately 2.16-m-long retractable radiation-cooled nozzle skirt extension. The corresponding nozzle exit diameter is about 1.87 m.

[16]Reference Mission Version 3.0 Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team, B.G. Drake, ed., Exploration Office Document EX13-98-036 (June 1998).

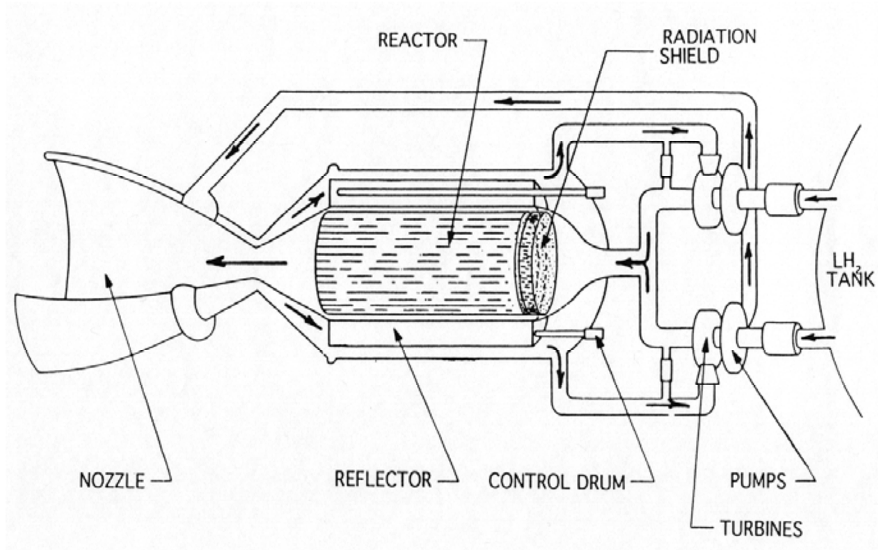



**Figure 5-32.** Schematic of expander cycle, dual-liquid-hydrogen turbopump NTR engine.

### *5.4.1.2 Mission and transportation system ground rules and assumptions*

The NTP-specific mission and transportation system GR&As that were used in DRA 5.0 are summarized in tables 5-9 and 5-10, respectively. Table 5-9 provides information about the assumed parking orbits at Earth and Mars, along with representative delta-V budgets for the "one-way" minimum energy cargo missions and the round-trip, "fast-conjunction" crewed mission. To size the cargo MTV components to accommodate all mission opportunities, the largest total delta-V across the 15-year synodic cycle (~2,028–2,043) was selected for both the propulsive capture and the aerocapture options. For the crewed mission, both short- and long-surface-stay opportunities, which will occur in the 2030–2046 timeframe, were examined during the Phase 1 analysis period. Long-surface-stay missions were selected because of their lower energy requirements and delta-V budgets, and their relatively short "one-way" transit times (~130–210 days) out to Mars and back. The crewed mission profile also assumed only propulsive capture at Mars. Besides the large delta-V requirements that were shown for the primary mission maneuvers (TMI, Mars orbital capture (MOC), and TEI), additional smaller delta-V maneuvers are needed for rendezvous and docking (R&D) of MTV components during the LEO assembly phase, for spacecraft attitude during interplanetary coast, and for Mars orbital operations and maintenance.

**Table 5-9.** NTR Specific Mission Ground Rules and Assumptions

| | |
|---|---|
| Mission Profile | • Split mission; cargo pre-deployed to Mars before crew leaves Earth<br>• Cargo missions uses "one-way" minimum-energy trajectories<br>• Round-trip crewed missions use "fast-conjunction" trajectories |
| Earth and Mars Parking Orbits | • Earth: 407-km circular<br>• Mars: 250 km × 33,793 km |
| Cargo Mission $\Delta$V Budget: Largest total $\Delta$V across 15-year synodic cycle (~2028 – 2045) used for both propulsive capture (PC) and aerocapture (AC) options | • Propulsive MOC: Earth Departure $C_3$ ~10.794 km$^2$/s$^2$, $\Delta V_{TMI}$ ~3.662 km/s, arrival $V_{inf}$ ~3.480 km/s, $\Delta V_{MOC}$ ~1.341 km/s<br>• AC at Mars: Earth Departure $C_3$ ~14.849 km$^2$/s$^2$, $\Delta V_{TMI}$ ~3.839 km/s<br>• **Note**: Gravity losses added to above ideal $\Delta$Vs (value of g-loss depends on $C_3$ vehicle thrust-to-weight (T/W), $I_{sp}$) |
| Crewed Mission $\Delta$V Budget: An "all-propulsive" mission profile with long surface stay times at Mars is the baselined approach | • Propulsive MOC: Earth Departure $C_3$ ~18.40 km$^2$/s$^2$, $\Delta V_{TMI}$ ~3.992 km/s, arrival $V_{inf}$ ~4.176 km/s, $\Delta V_{MOC}$ ~1.771 km/s<br>• Mars Departure $C_3$ ~ 14.80 km$^2$/s$^2$, $\Delta V_{TEI}$ ~1.562 km/s<br>• **Note**: Gravity losses added to above ideal $\Delta$Vs |
| Additional $\Delta$V Requirements | • LEO R&D between orbital elements: ~100 m/s<br>• Coast attitude control and mid-course correct: ~15 m/s and ~50 m/s, respectively<br>• Mars orbit maintenance: ~100 m/s |

A range of cargo payload masses was developed during Phases 1 and 2 that established the physical size and overall mass for the cargo MTVs. In Phase 1, the use of ISRU for MAV propellant production was not considered. This led to a heavier MAV, EDL system, and aeroshell (~110.9 t) for the PC option. The need for additional TPS mass on the aeroshell for the AC option increased the cargo payload mass further (~138.2 t). In Phase 2, reductions in the aeroshell and TPS masses as well as the use of a nuclear surface power system (NSPS) and ISRU-produced ascent propellant decreased the payload masses to ~99.5 t and 103 t for the PC and AC options, respectively.

For the crewed mission, the outbound payload mass remained fixed at approximately 51.3 t during Phases 1 and 2. For long-surface-stay Mars missions, the crewed MTV carries contingency consumables equivalent to those found on the habitat lander. This allows the crew MTV to function as an orbital "safe haven" in the event of a major failure of a key surface system. In the case of a nominal surface mission, the contingency consumables are jettisoned prior to the TEI maneuver. Assuming that the crew collects and returns with approximately 0.5 t of Mars samples, the total return payload mass for the crewed mission is about 43.9 t.

Table 5-10 lists the key transportation system GR&As that were used in this study. The NTP engine and fuel type, thrust level, and operating characteristics are summarized first. The 25-klb$_f$ NERVA-derived engine design that is baselined here uses "composite" fuel, operates with $T_{ex}$ ~2,700 K, and has an $I_{sp}$ of ~910 s, although 900 seconds were used in the majority of Phase 1 and 2 analysis for conservatism. The total $LH_2$ propellant loading for a Mars mission consists of the usable propellant plus performance reserve, post-burn engine cool down, and tank trapped residuals. For the smaller auxiliary maneuvers, an established storable bipropellant RCS is used.

**Table 5-10.** NTR Specific Transportation System Ground Rules and Assumptions

| | |
|---|---|
| SPS Options and Use of ISRU | • Phase1I: SPS unspecified, no ISRU, MAV carries its own propellant<br>• Phase 2: Nuclear and solar SPS options compared along with benefits of ISRU |
| Cargo Mission Payload Masses | • Propulsive MOC: 110.9 t (Phase 1) / 99.5 t – 122 t (Phase 2)<br>• Aerocapture (AC): 138.2 t (Phase 1) / 103 t – 133 t (Phase 2) |
| Crewed Mission Payload Mass: Total crew consumables based on 900-day mission that includes 180-day transit times to and from Mars and 540 days at Mars | • Trans Habitat: 27.5 t<br>• Crew (6): 0.6 t<br>• Total Crew Consumables: 13.23 t; ~5.29 t (transit to and from Mars), ~7.94 t (contingency); assumes crew consumption rate of ~2.45 kg/person/day)<br>• CEV/service module: 10.0 t<br>• Returned Mars Samples: 0.5 t |
| Mission Abort Strategy | • Outbound: Abort to Mars Surface<br>• At Mars: Abort to orbiting crew MTV, which carries contingency consumables |

The $LH_2$ propellant that will be used in the NTP cargo and crewed MTVs will be stored in the same "state-of-the-art" Al/Li $LH_2$ propellant tank that will be developed and used in the Ares-V heavy-lift launch vehicle. Tank sizing assumes a 30-psi ullage pressure, 5-g axial/2.5-g lateral launch loads, and a safety factor of 1.5. A 3% ullage factor is also assumed. All NTP propellant tanks have a combination foam/multilayer insulation (MLI) system for passive thermal protection. An active zero-boiloff (ZBO) cryocooler system is used on all tanks (except drop tanks) to minimize/eliminate $LH_2$ boiloff during the long-duration Mars missions. The heat that will be load into the propellant tanks is largest in LEO during the vehicle assembly phase. Because non-"bimodal" NTR engines are assumed in this study, it is necessary to use solar PVAs to supply needed primary electrical power for the MTV systems. Because of the decreased solar intensity at Mars (~486 W/m$^2$), array areas can become quite large (~10 m$^2$/kW$_e$), necessitating multiple arrays. Lastly, table 5-10 provides information on the assumed dry-weight contingency (DWC) factors, Ares V LEO lift requirements, and shroud cylindrical payload envelope dimensions that will be used during the Phase 1 and 2 analysis cycles. A 30% DWC is used on the NTP system and advanced composite structures (e.g., trusses), and 15% is used on heritage systems (e.g., Al/Li tanks, RCS, etc.,). The maximum Ares V lift requirement was determined primarily by the mass of the various cargo payload elements that were used in the study.

#### *5.4.1.3 Design reference architecture 5.0 NTR mission description*

The current Mars DRA (i.e., DRA 5.0) is again centered around a long-surface-stay, split cargo/piloted mission approach. Two cargo flights are used to pre-deploy a cargo lander to the surface and a habitat lander into Mars orbit where it remains until the arrival of the crew on the next mission opportunity. The cargo flights use "one-way" minimum-energy, long-transit-time trajectories. Each cargo vehicle is assembled in LEO via autonomous R&D with vehicle and payload components delivered on Ares V cargo heavy-lift launch vehicles. Each cargo vehicle uses a common "core" propulsion stage with 3–25-klb$_f$ NTR engines operating with a, $I_{sp}$ of approximately 900 s. Assembling both cargo vehicles requires about five to six Ares V, launches depending on the assumed payload mass and whether the PC or AC option is used at Mars. Figure 5-33 illustrates the five-launch AC option. The first two Ares V launches deliver the NTR "core" propulsion stages, while the third Ares V launch delivers two short, "in-line" $LH_2$ tanks that are packaged end-to-end. Because of the significant increase in the current aeroshell mass (~40–45 t vs. ~10 t used in earlier DRM studies), additional propellant is required for the TMI maneuver. The in-line tanks supply extra propellant to augment that contained in the core propulsion stages. Once in orbit, the in-line tanks separate and dock with the propulsion stages, which function as the active element in this R&D maneuver. The cargo transfer vehicles then R&D with the two AC'ed payload elements that will be delivered on the final two Ares V launches. For the AC option, the large aeroshell, which is configured as either a triconic or ellipsled geometry, has multiple functions that include a payload shroud during launch, and an aerobrake and a TPS during AC into Mars orbit and subsequent EDL on Mars.

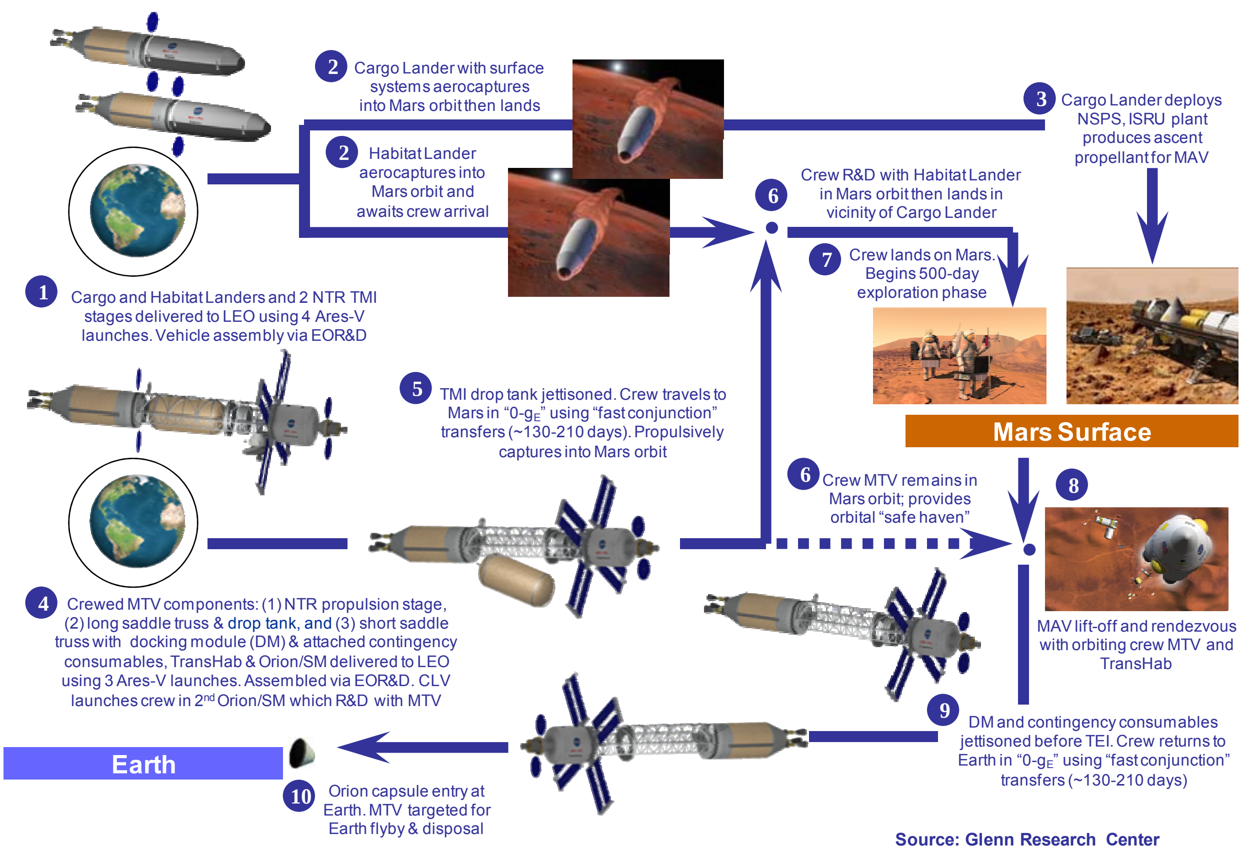



**Figure 5-33.** DRA 5.0 long-stay NTR mission overview.

Following the TMI maneuver, the NTR transfer vehicle, which remains with the payload, uses its on-board RCS to provide mid-course correction and attitude control during the coast out to Mars. The core propulsion stage can also use its small PVA to supply kilowatts of electrical power to the payload up to the point of vehicle-payload separation near Mars. The AC'ed habitat lander must carry its own multi-$kW_e$, deployable/retractable PVA for use in Mars orbit while awaiting the arrival of the crew and, possibly later, on the Mars surface.

Once the operational functions of the orbiting habitat and surface cargo landers are verified and the MAV is supplied with ISRU-produced ascent propellant, the crewed mission will be cleared to go on the next available mission opportunity (~26 months later). The "all-propulsive" crewed MTV uses the same "common" NTR propulsion stage but includes additional external radiation shielding on each engine for crew protection during engine operation. Four 12.5-$kW_e$/125-$m^2$ rectangular PVAs provide the crewed MTV with approximately 50 $kW_e$ of electrical power for crew life support (~30 $kW_e$), ZBO cryocoolers (~15 $kW_e$), and high data-rate communications (~5 $kW_e$) with Earth. Four Ares V flights are required to deliver the main MTV components, which are launched in the following preferred order: (1) a four-sided "star truss" with deployable PVAs, TransHab module containing consumables for the crew of six, and a long-lived Orion/SM for vehicle-to-vehicle transfer and "end-of-mission" Earth entry; (2) an "in-line" $LH_2$ propellant tank with ZBO system; (3) the NTR propulsion stage, also with ZBO system; and (4) twin $LH_2$ drop tanks that are attached to the integrated star truss-/propellant feed line assembly and are launched last to reduce LEO boiloff. When assembly is complete, the Mars crew is launched on the CLV and the Orion/SM is used for R&D with the crewed MTV.

Following the TMI maneuver, the two drained drop tanks are jettisoned and the crewed MTV coasts to Mars under "0-$g_E$" conditions and with its four PVAs tracking the sun. Attitude control, mid-course correction, and vehicle orientation maneuvers are provided by a split RCS with thrusters and bipropellant located on the rear NTR propulsion stage and the star truss forward adaptor ring. After propulsively capturing into Mars orbit, the crewed MTV will rendezvous with the orbiting Hab lander using engine cool-down thrust and the vehicle RCS. The crew then transfers over to the lander using the Orion/SM that subsequently returns and docks to the TransHab autonomously. At the end of the Mars exploration phase, the crew lifts off and returns to the MTV using the MAV. Following the transfer of the crew and samples to the MTV, the MAV is jettisoned. After checkout and verification of all MTV systems, the crew jettisons the contingency consumables, performs the TEI burn, and begins the journey back to Earth. After an approximately 6-month trip time, the crew enters the Orion/SM, separates from the MTV, and subsequently enters the atmosphere while the MTV flies by Earth at a "sufficiently high altitude" and is disposed of into heliocentric space.

#### *5.4.1.4 Cargo and crewed Mars transfer vehicle design features and characteristics*

**PHASE 1 VEHICLE CONCEPTS**

A variety of MTV designs were developed during the recent Phase 1 and 2 analysis periods. Figure 5-34 shows a sampling of cargo and crewed vehicle options that was evaluated during Phase 1. Higher delta-V "short-stay" missions require larger amounts of propellant resulting in increased vehicle size and mass. For the difficult 2039 mission opportunity (547-day round-trip time with "30-day" stay), the crewed MTV IMLEO is approximately 501 t and requires seven Ares V launches, each with a lift capability of about 109 t and a cylindrical payload volume of approximately 8.4 m D × 30 m L. The NTR core propulsion stage, in-line tank, star truss and crewed payload, and four large drop tanks each use a separate launch, but only the core propulsion stage and in-line tank are near the lift and payload length limits. For the "generic" long-surface-stay mission (~180-day transits to and from Mars with ~540 days at Mars), the total mission delta-V is reduced by more than 3 km/s, thereby lowering the IMLEO of the crewed MTV to approximately 315 t and the number of Ares V launches to four. The propulsion stage, in-line tank, star truss and crewed payload, and twin drop tanks, which are packaged end-to-end inside the payload shroud, each use a separate Ares V launch. The truss and crewed payload launch is only at 55% of capacity.

The PC and AC cargo MTV concepts have comparable IMLEO values (~319 t and 311 t, respectively). Both options use Earth orbit rendezvous and docking (EOR&D) for assembly and require three Ares V launches. The PC cargo vehicle employs a "saddle truss" (with attached inside drop tank) that connects the NTR propulsion stage to the payload (~111 t). The "saddle truss" configuration allows the drop tank to be easily jettisoned after the TMI maneuver, thus reducing vehicle mass and the propellant requirements for the MOC burn. Although the cargo payload mass is larger (~138 t) for the AC option (because of increased aeroshell TPS mass), the cargo vehicle is only used for the TMI maneuver. To augment the propellant capacity of the core propulsion stage, an "in-line" propellant tank is added to the vehicle configuration.

**PHASE 2 VEHICLE CONCEPTS**

During the Phase 2 analysis cycle, AC was selected over PC for the cargo mission. A "common" 10-m diameter (D) shroud/aeroshell was used for launch, AC and EDL were also analyzed, and a range of surface payloads was established that reflected different surface exploration strategies (e.g., the "commuter" option), power systems (nuclear or solar), as well as the impact of ISRU. Additional analysis and refinements of the crewed mission payload was postponed until a later Phase 3 effort.

The Phase 2 cargo and crewed MTV concepts are shown in figure 5-35. Five Ares V flights are required for LEO assembly of the two cargo vehicles. Each vehicle has an IMLEO of approximately 246.2 t and an overall length (L) of about 72.6 m, which includes the 30-m-long AC'ed payload. The total payload mass (aeroshell, propulsive lander, and surface payload) is approximately 103 t and is consistent with a surface strategy using nuclear power and ISRU, one that is similar to that used in DRM 4.0. The NTR propulsion stage has an overall L of about 28.8 m (~26.6 m with retracted nozzles for launch) and a launch mass of about 96.6 t. The stage $LH_2$ tank has an ID and L of approximately 8.9 m and approximately 16.3 m, respectively, and a propellant capacity of approximately 59.4 t. The in-line tank (one of two that will be delivered on the third Ares V launch) has a launch mass of 46.6 t and overall L of about 13.3 m, including the forward and rear adaptor sections. The 8.9-m ID tank has an approximately 10.23-m L and holds an additional approximately 34.1 t of $LH_2$. The NTR cargo vehicle also carries about 4.5 t of RCS propellant for LEO assembly operations, coast attitude control, MCC, and Mars orbit maintenance. Approximately 91 t of $LH_2$ are used during the TMI maneuver (including the "post-burn" cool-down propellant). The corresponding engine burn time was approximately 39 minutes, well within the 62-minute single-burn duration that was demonstrated by the NRX-A6 engine during the NERVA program.

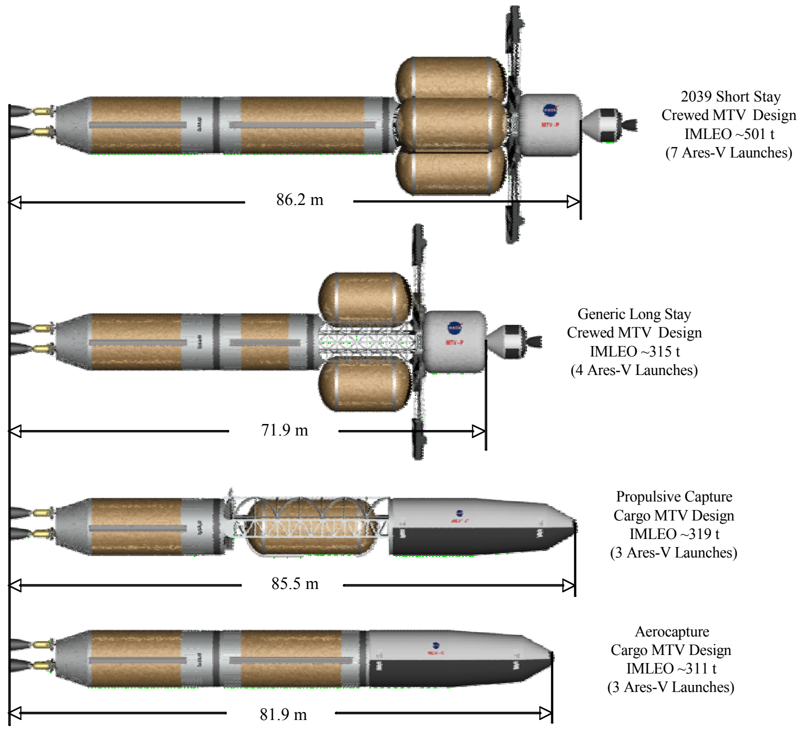



**Figure 5-34.** Phase 1 NTR crewed and cargo Mars transfer vehicle design concepts.

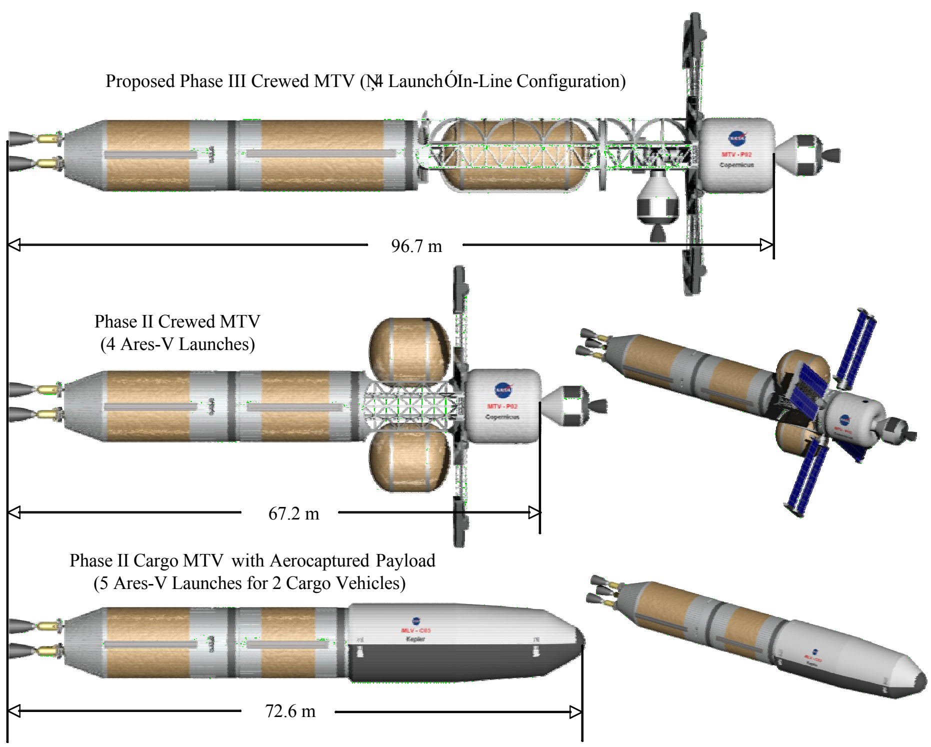



**Figure 5-35.** Crewed and cargo Mars transfer vehicle design concepts (Phase 2/proposed Phase 3).

The crewed MTV requires four Ares V flights to deliver its key components. It has an IMLEO of about 333 t and an overall vehicle length of approximately 67.2 m as compared to approximately 315 t and 71.8 m for the equivalent Phase 1 configuration. In the Phase 2 analysis, a larger D (~9.1 m OD) but shorter L (~26.6 m) cylindrical payload enveloped was used. The use of shorter (~10.23 m) drop tanks on the crewed MTV, which was done to achieve commonality with the cargo vehicle in-line tank, required a longer (~12-m) in-line tank, thus increasing the vehicle dry mass and resulting in the larger IMLEO value. The core NTR propulsion stage has a larger launch mass (~106.3 t) than that used on the cargo vehicle due to the addition of an external radiation shield on the engines. The $LH_2$ tank size and propellant capacity was approximately the same, however. The in-line tank launch mass was about 68.1 t, which included approximately 50.9 t of $LH_2$ propellant. The four-sided star truss and crewed payload had a combined mass of approximately 58.9 t. Lastly, the twin drop tanks had a combined launch mass of about 99.7. The total RCS propellant load (~8 t) was split between the core stage and truss forward cylindrical adaptor ring. For the round-trip crewed mission, the total usable $LH_2$ propellant loading was approximately 180.2 t and the corresponding total mission engine burn duration was approximately 80 minutes (~55 minutes for TMI, ~15 minutes for MOC, and ~9.8 minutes for TEI), well within the approximately 2-hour accumulated engine burn time that was demonstrated by the NERVA experimental engine (NRX-XE).

Looking back at the Phase 1 and 2 crewed vehicle assumptions, they were somewhat ill-defined. They did not address how nearly 8 t of contingency consumable would be jettisoned from the crewed MTV prior to TEI, or how and where a second Orion/SM or the MAV would dock to the TransHab module with an Orion/SM-type vehicle (part of the assumed crewed payload) that would already be attached to the front docking port on the TransHab. Figure 5-36 shows a "preliminary" Phase 3 crewed vehicle configuration that addresses these issues. Like the other crewed vehicle, it is also a "four-launch" configuration that uses the following elements: (1) the NTR propulsion stage; (2) a longer in-line propellant tank (used for parts of the TMI and MOC maneuvers); (3) a "saddle truss" with

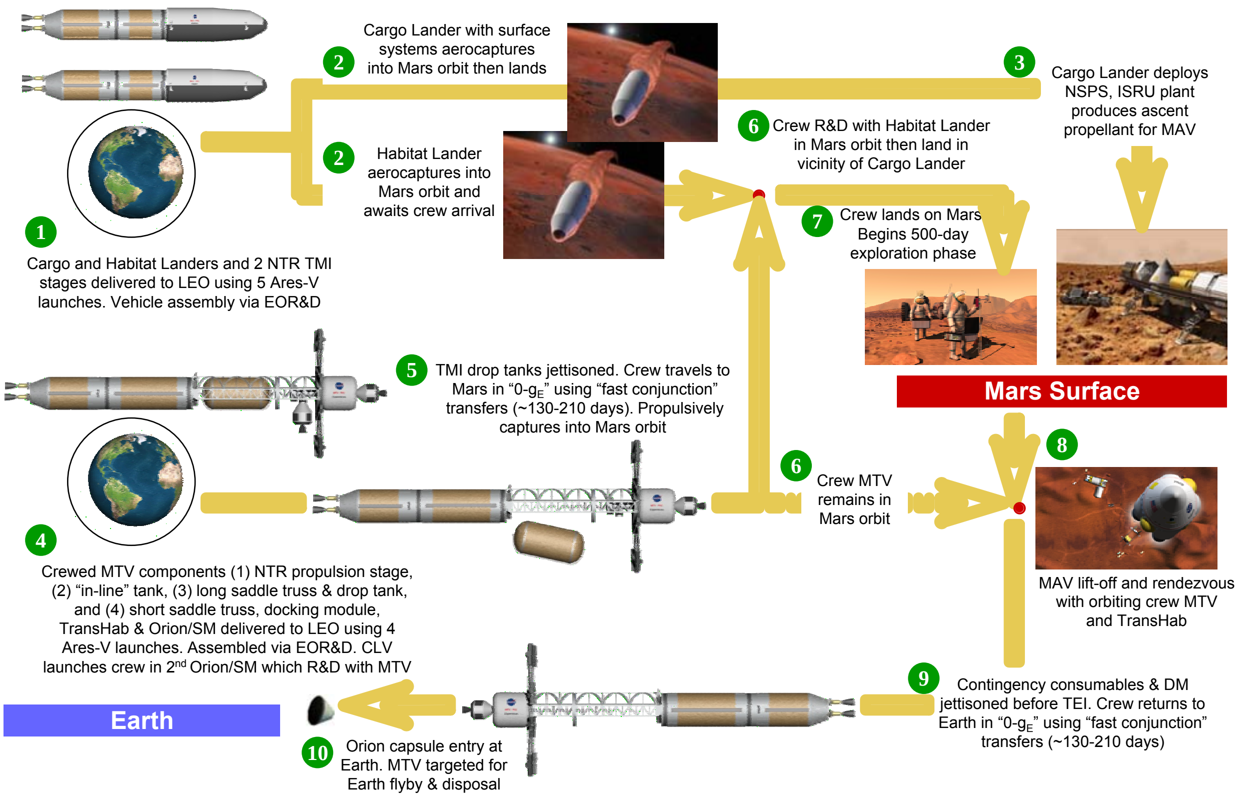



**Figure 5-36.** DRA 5.0 long-stay Mars mission overview: alternative NTR crewed MTV.

a single large drop tank (used for TMI only); and (4) a "revised" crew payload. The revised payload manifest includes a short saddle truss with a "T-shaped" docking module (DM) attached to the saddle truss forward adaptor ring. The new DM provides access to the TransHab to the right and to the jettisonable contingency consumables canister mounted at the left. In the middle of the DM is a third hatch that provides docking access for a second Orion/SM or the MAV. Following the crew's return from Mars and MAV separation, the DM and attached contingency consumables canister are both jettisoned to reduce vehicle mass prior to TEI.

The addition of the short saddle truss (~4.7 t), DM (~1.8 t), and consumables canister (~1.9 t) increases the total crewed payload mass by approximately 8.4 t, from approximately 51.3 t to approximately 59.7 t. With the added payload, the crewed vehicle IMLEO increases by about 23.5 t, from about 333 t to about 356.5 t. The launch mass for the individual vehicle elements is as follows: (1) NTR propulsion stage approximately 106.2 t; (2) in-line tank approximately 94.6 t; (3) saddle truss and drop tank approximately 96 t; and (4) short saddle truss, DM, and remaining payload approximately 59.7 t. With the additional payload mass, the total usable $LH_2$ propellant loading for the crewed mission increases to about 191.7 t and the total mission burn time for the engines to about 84.5 minutes (~57.8 minutes for TMI, ~16 minutes for MOC, and ~10.7 minutes for TEI).

#### *5.4.1.5 Summary of findings and recommended future work*

Phase 1 and 2 analysis efforts to date show that there are many performance, system, mission, and operational benefits to using NTP. Its high-thrust/high-$I_{sp}$ (~2× chemical) capability means short burn durations, less propellant mass, and fewer Ares V launches. At launch, NTR engines contain negligible amounts of radioactivity, thus simplifying shipping and handling as well as the engine, stage, payload, and launch vehicle integration function at KSC. The use of multiple, smaller-thrust 25-$klb_f$ engines, each with dual-$LH_2$ TPAs, provides a "pump-out" and "engine-out" capability that can increase crew safety and reduce mission risk. Small engine size is also expected to help reduce the time and cost to develop and ground and flight test these engines. The strong technology synergy between NTP and chemical systems (e.g., $LH_2$ TPAs, radiation-cooled nozzle extensions, and large Al/Li propellant tanks) should provide further cost savings.

From a mission and operations perspective, the NTP-based space transportation system has fewer vehicle elements and simpler space operations. No complex orbital assembly is required as with chemical propulsion, just EOR&D of several vehicle elements – two for each cargo vehicle and three for the crewed MTV. A higher-performance NTP-based space transportation system is more tolerant of mass growth. It also provides NASA planners with greater mission flexibility, such as a propulsive capture option for cargo payloads should technical difficulties arise with the aerocapture approach. Finally, the use of NTP allows greater future growth capability including use of higher-temperature fuels and "bimodal" engine (BNTR) operation, which can eliminate the need for deploying and operating large sun-tracking PVAs. The configuration of the BNTR-powered MTV (long and linear) is also naturally compatible with artificial-gravity operations (Borowski, et al., 1999, 2000[17,18]) that can help maintain crew fitness during transit to Mars and back, also while in Mars orbit in the event of an "abort-to-orbit" scenario.

Recommendations for future work include the need to select and mature the designs for the different surface systems and their packaging on the Mars lander as well as the configuration of the lander design (vertical or horizontal) and volume to help define payload mass and volume sizing requirements for a "Mars-relevant" Ares V launch vehicle. Higher-fidelity payload masses are also required to improve performance and sizing estimates for the transportation systems. For NTP, further detailed design is recommended (in the engine and vehicle component and subsystems areas) on the "preliminary" Phase 3 crewed MTV, which was discussed above. Lastly, the large number of Ares V flights has been identified as a mission-risk area. Further analysis is needed to determine attractive performance levels for both the Ares V (e.g., greater lift and larger payload envelope) and NTP system (e.g., higher temperature fuel to increase $I_{sp}$) that could help reduce the launch count.

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[17] S.K. Borowski, L.A. Dudzinski and M.L. McGuire, "Artificial Gravity Vehicle Design Option for NASA's Human Mars Mission Using "Bimodal" NTR Propulsion," AIAA-99-2545, American Institute of Aeronautics and Astronautics (June 1999).
[18] S.K. Borowski, L.A. Dudzinski and M.L. McGuire, "Bimodal" Nuclear Thermal Rocket (NTR) Propulsion for Power-Rich, Artificial Gravity Human Exploration Missions to Mars," IAA-01-IAA.13.3.05, 52nd International Astronautical Congress, Toulouse, France (October 2001).

## 5.4.2 Chemical propulsion architecture

### *5.4.2.1 Trade tree options*

One of the objectives of this study was to consider a broad range of architecture options and identify the most promising transportation architectures as a basis for defining an updated Mars DRA. The MAWG defined the top-level trade tree (shown in figure 5-37) that included a full range of vehicle and mission options for Mars transportation architectures. The study was divided into two phases; the first phase focused on mission analysis and parametric analysis of the Mars transportation system to support primary architecture trades to compare long-stay vs. Short-stay missions and pre-deploy architectures vs. all-up architectures. The second phase focused on development of higher-fidelity vehicle design concepts for comparison of aerocapture vs. propulsive MOI as well as analysis of potential application of ISRU. The specific branches of the trade tree that were analyzed in Phase 1 and Phase 2 are shown on the trade tree. To narrow the architecture options, the trade tree was trimmed based on four primary decisions:

- Long Mars Stay Time vs. Short Mars Stay Time
- All-up Mission vs. Pre-deploy Mission (split cargo and crew missions)
- AC MOI vs. Propulsive MOI
- No ISRU vs. ISRU

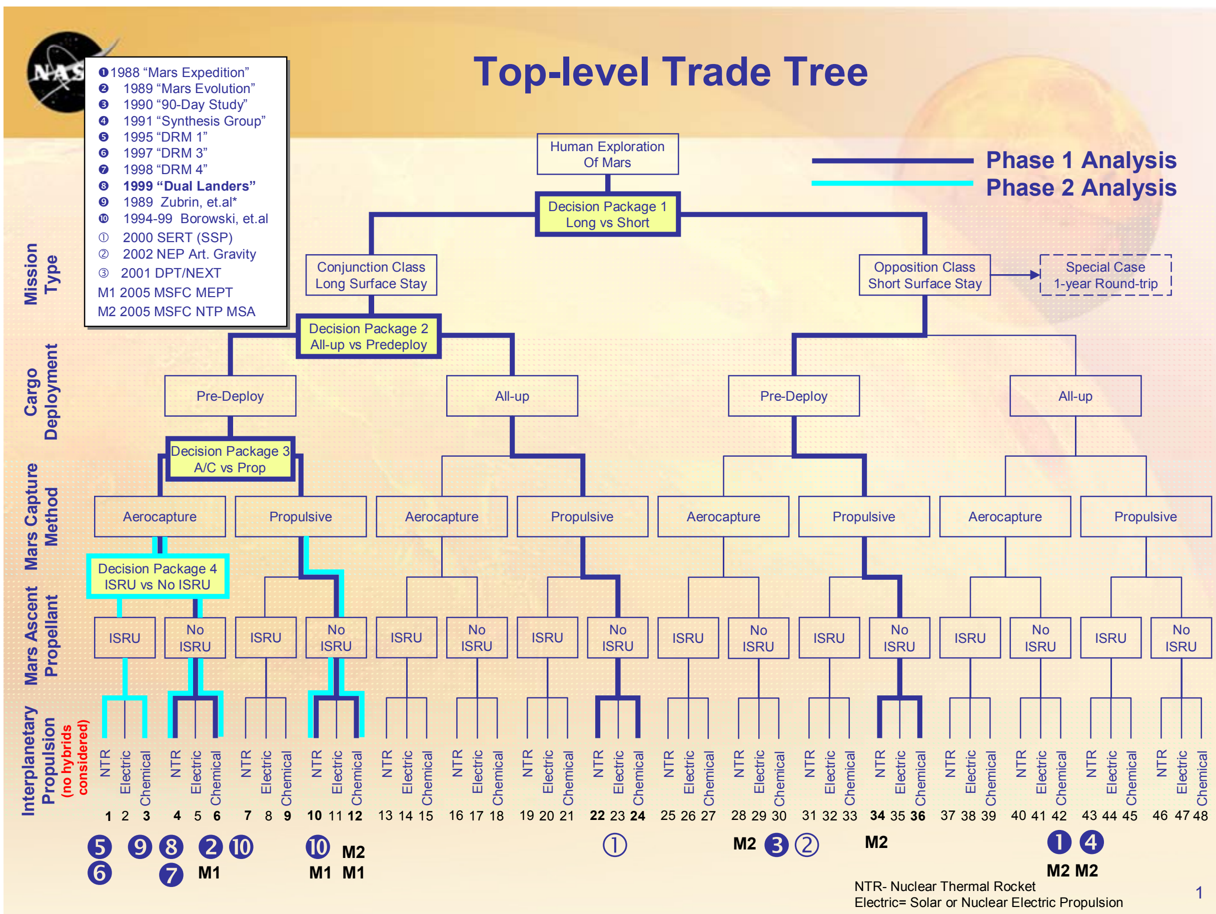



**Figure 5-37.** Top-level Mars transportation architecture trade tree.

The mission and vehicle options that are associated with the selected trade tree branches are shown in table 5-11. The application of ISRU and choice of surface power were significant drivers for the cargo mission Mars payload mass. The chemical propulsion trade tree branches that were selected for analysis in Phase 2 were based on long-stay, pre-deploy mission architectures. Trade tree branches 3 and 6 use aerocapture for the cargo mission MOI maneuvers, and trade tree branch 12 uses propulsive maneuvers for MOI. The crew missions use propulsive MOI for all of these cases. Trade tree branches 3 and 6 provide a comparison of ISRU vs. non-ISRU. In this study, it was assumed that ISRU would be used to produce the $LO_2$ for the Mars crew ascent vehicle $LO_2/CH_4$ propulsion system. The $LO_2$ would be produced on the Mars surface using atmospheric processing. Cases 6 and 12 provide a comparison of aerocapture vs. propulsive MOI for the cargo mission.

**Table 5-11.** Phase 2 Chemical Propulsion Analysis Cases

A/C* - Aero-Assist Technology (LaRC)
Chem - Chemical Propulsion (MSFC)

| F&SS Trade Tree Branch | Trajectory Type | Mission Type | MTS Configuration | Ascent Stage Propellalnt | Surface Power | Mission | Trans-Mars Injection (TMI) | Mars Orbit Insertion (MOI) | Trans-Earth Injection (TEI) | Earth Return (ER) | Cargo Mission Payload (kg) |
|---|---|---|---|---|---|---|---|---|---|---|---|
| 3 | Long-Stay | Pre-deploy | Cargo - Aeocapture | ISRU | Nuclear | Cargo | Chem | A/C* | | | 103 |
| | | | Crew - All Propulsive | | | Crew | Chem | Chem | Chem | Dir. Entry | |
| 3a | Long-Stay | Pre-deploy | Cargo - Aeocapture | ISRU | Solar | Cargo | Chem | A/C* | | | 113 |
| | | | Crew - All Propulsive | | | Crew | Chem | Chem | Chem | Dir. Entry | |
| | | | | | | | | | | | |
| 6 | Long-Stay | Pre-deploy | Cargo - Aeocapture | Non-ISRU | Nuclear | Cargo | Chem | A/C* | | | 126 |
| | | | Crew - All Propulsive | | | Crew | Chem | Chem | Chem | Dir. Entry | |
| 6a | Long-Stay | Pre-deploy | Cargo - Aeocapture | Non-ISRU | Solar | Cargo | Chem | A/C* | | | 133 |
| | | | Crew - All Propulsive | | | Crew | Chem | Chem | Chem | Dir. Entry | |
| | | | | | | | | | | | |
| 12 | Long-Stay | Pre-deploy | Cargo - All Propulsive | Non-ISRU | Nuclear | Cargo | Chem | Chem | | | 122 |
| | | | Crew - All Propulsive | | | Crew | Chem | Chem | Chem | Dir. Entry | |

* Chemical propulsion used for post aerocapture orbit adjustment

### *5.4.2.2 Chemical transportation architecture overview*

The MTV concept options for this study consisted of multiple-stage vehicles that were made up of separate propulsive elements for each major mission maneuver. The vehicle elements were designed to allow maximum design commonality, efficient Earth-to-orbit delivery, and efficient assembly in LEO. The mission architectures that were considered in this study use two cargo vehicles and one crew vehicle for each Mars mission, as shown in figure 5-38.

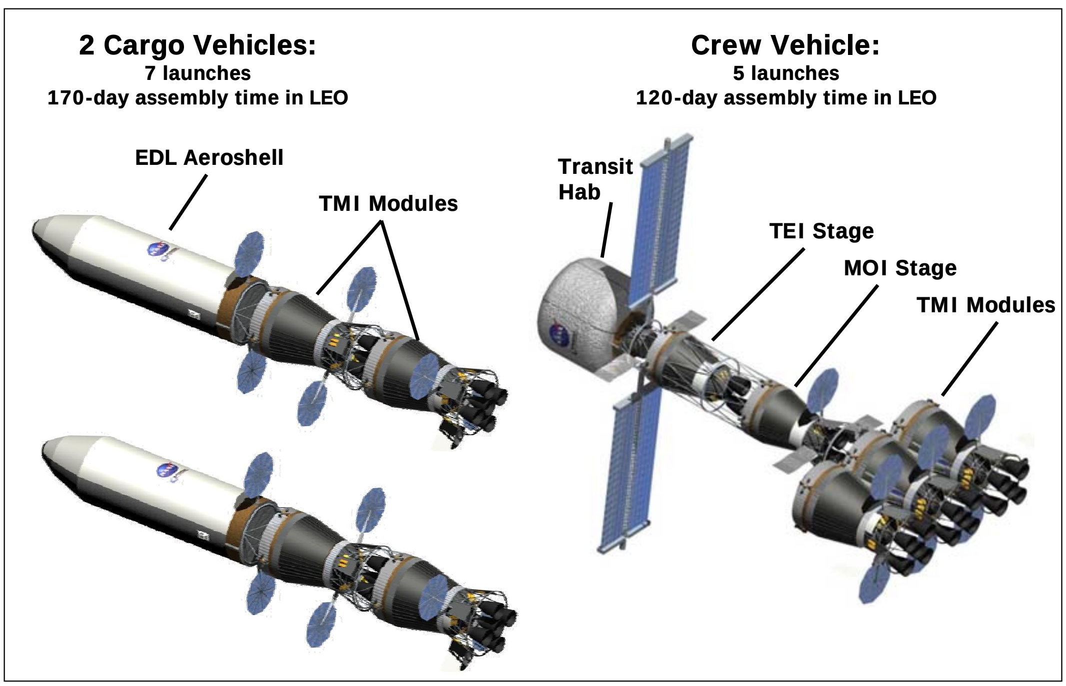



**Figure 5-38.** Chemical propulsion Mars transportation system architecture.

The cargo vehicles would depart Earth approximately 2 years before the crew vehicle. One cargo vehicle would transport the Mars SHAB as payload, and the other would transport the Mars DAV as payload. The cargo vehicles consist of a payload that is enclosed in a cylindrical aeroshell and propulsive stages for TMI. A propulsive MOI stage would be added to the cargo vehicles for trade tree Case 12. The aeroshell would serve as a payload shroud for Earth-to-orbit launch of the payloads and an aerodynamic lifting body for Mars aerocapture, entry, and descent. Since Case 12 uses propulsive MOI, the TPS mass would be reduced, and other systems that are required for aerocapture would be eliminated or replaced with the systems that are required for propulsive MOI. Depending on the specific case, two or three TMI modules are required for each cargo vehicle.

The crewed vehicle consists of the CEV/ERV, TransHab, and propulsive stages. This stack undergoes TMI, MOI, and TEI. Each crew vehicle requires three TMI modules, one MOI module, and one TEI module. The CEV is used to transport the crew to LEO prior to TMI. It remains docked to the TransHab until shortly before Earth return, when the crew would separate from the TransHab and perform a direct-entry Earth return. The propulsive elements of the crew vehicle include the TEI stage, the MOI stage, and two TMI stages. The first TMI stage consists of two TMI propulsion modules, which would perform the first TMI burn, and the second TMI stage consists of one TMI propulsion module, which would perform the second TMI burn.

The assembly of each MTV vehicle in LEO requires a LEO assembly reboost module, which performs attitude control and orbital reboost of the MTV during the assembly and LEO loiter periods. The reboost modules are jettisoned from the vehicle stack prior to TMI.

#### *5.4.2.3 Mars transit vehicle chemical propulsion elements*

**Trans-Mars injection module**

All of the TMI modules are common among the cargo and crew vehicles. Each vehicle performs a 2-burn departure. The required propellant for the crew mission drove all but one design; so in the case of the cargo vehicles and one crew vehicle, some propellant was offloaded. Each TMI module is jettisoned after it performs its burn. In the baseline cases, five RL10-B2 engines were used; the avionics package on each TMI module provided independent guidance, navigation, and control (GN&C) until it was docked to the entire stack during assembly. However, each module was responsible for fulfilling its own power requirements through launch and while connected to the rest of the vehicle for the duration of the LEO loiter period. Launch loads were considered to size the structural members, and primarily metallic materials were used for the beams and propellant tanks. Some composite materials were sized for panels. All of the propellant tanks were cryogenic, and a vehicle longitudinal axis sun-pointing orientation was used for the 407-km circular orbit to size the thermal components. Additional information regarding the TMI modules is located in the results section for each of the cases that were run, and specific subsystem details are summarized in the appendices.

**Mars orbit insertion modules (cargo and crew missions)**

Custom MOI modules were designed for the Case 12 cargo and all crew missions to save mass to meet the requirements of the mission. Despite the propellant loadings being different, overall design assumptions were the same. For the all-propulsive crew and cargo missions, the MOI modules used two RL10-B2 engines to complete the MOI burn. For the crew vehicle, the MOI stage provided independent GN&C until it was docked to the stack during assembly and the TEI module took over. In the case of the cargo vehicle where there is no TEI stage, the MOI module provides GN&C for stack throughout the entire mission. It also supplies the command and data links to all other stages, the high-gain antenna (HGA) communications to Earth, and the lander payload data link and standby power. Again, primarily metallic materials were assumed, and the thermal approach was the same when compared to the TMI modules.

**Trans-Earth injection module**

The TEI stage for the crew mission was a scaled-down version of the crew MOI module with a few additional components. It uses two RL10-B2 engines and performs the TEI burn as well as the plane change while in Mars orbit. It is responsible for providing the following: the GN&C for the stack throughout mission, the command and data links to all other stages, the HGA communications to Earth, and the TransHab data link and power. Structure and thermal assumptions were the same as compared to the TMI module.

**Low-Earth orbit assembly reboost module**

For this study, a generic re-boost module was designed to perform all on-orbit station-keeping operations during the assembly phase of the mission. The re-boost module was required due to the long on-orbit assembly time and the mass of the vehicle that is being assembled. With an assembly and departure orbit of 407 km, the vehicle is subjected to a small amount of drag that could decrease its orbit over time. As with the ISS, the vehicle that is being assembled on orbit would periodically need to be re-boosted so the rendezvous and docking altitude remains constant. Table 5-12 summarizes the delta-V assumptions that were used in designing the re-boost module; the maximum delta-V required per year is 161 m/s (13.5 m/s per month).

**Table 5-12. Reboost Module Delta-V**

| Maneuver | Delta-V per year (m/s) |
|---|---|
| Station Keeping | 55 |
| Drag Compensation | 100 |
| Attitude Control (3-Axis) | 6 |
| Total | 161 |

In addition to the delta-V assumptions, the re-boost module was designed for a year of operations with the assumption that a launch would occur each month. The total required propellant that the re-boost module was designed for was 32.8 t. Again, the re-boost module was designed as a generic module; for the worst-case scenario, the re-boost module could be offloaded so that it only carries enough propellant for the specific vehicle that is being assembled on orbit.

Lastly, figure 5-39 shows the re-boost module configuration. The re-boost module was sized with a diameter of 8.4 m. This diameter allowed the re-boost module to be launched with any payload. The solar arrays for the re-boost module were only designed to supply power to the re-boost module. The re-boost module contained an RCS consisting of redundant 100-$lb_f$ bipropellant monomethyl hydrazine nitrogen tetroxide (MMH/NTO) pressure-feed thrusters that were mounted in forward, aft, port, and starboard facing mounts on the four quadrants of the module. With all engines available, the total axial thrust available was 800 $lb_f$. Each quadrant had a set of tanks that consisted of one fuel and one oxidizer tank, with each tank containing four pressurization tanks. This configuration provided redundancy in case of loss of tank or lines that feed from that tank to the engines. The RCS that was used for this configuration is similar to the systems that are used for long-lift (12-year) communication satellites.

Table 5-13 shows the mass breakdown for the re-boost module. A 20% contingency was applied to all dry masses.

**Table 5-13. Reboost Module Mass Summary**

| Components | Mass (kg) |
|---|---|
| Structures | 5306.5 |
| RCS | 3396.6 |
| Avionics | 6 461.0 |
| TCS | 495.9 |
| RCS Propellant | 32804.0 |
| Total Mass without Contingency | 48464.0 |

### *5.4.2.4 Low-Earth orbit assembly*

**Earth-to-orbit launch vehicle configurations**

The elements of the Mars transportation system would be assembled in LEO. The reference assembly orbit was assumed to be a 407-km-altitude circular orbit. The Ares V would transport the MTV elements to the assembly orbit using the two launch configurations that are shown in figure 5-39. The gross payload capability of the reference Ares V vehicle was assumed to be 131.4 t. The net payload lift capability to transport the MTV elements to orbit depends on which launch vehicle configuration is used. The cargo mission payloads would be delivered to orbit packaged in the EDL aeroshell. Since the aeroshell serves as the ETO launch shroud, the launch vehicle lift capability corresponds to the gross lift capability of 131.4 t. The propulsive stages, TransHab's, and reboost modules would be delivered to orbit using a 10-m-diameter by 25-m-long payload shroud. The estimated mass of the payload shroud was determined to be 20 t; therefore, the Ares V payload capability for the shrouded payloads was assumed to be 111.4 t.

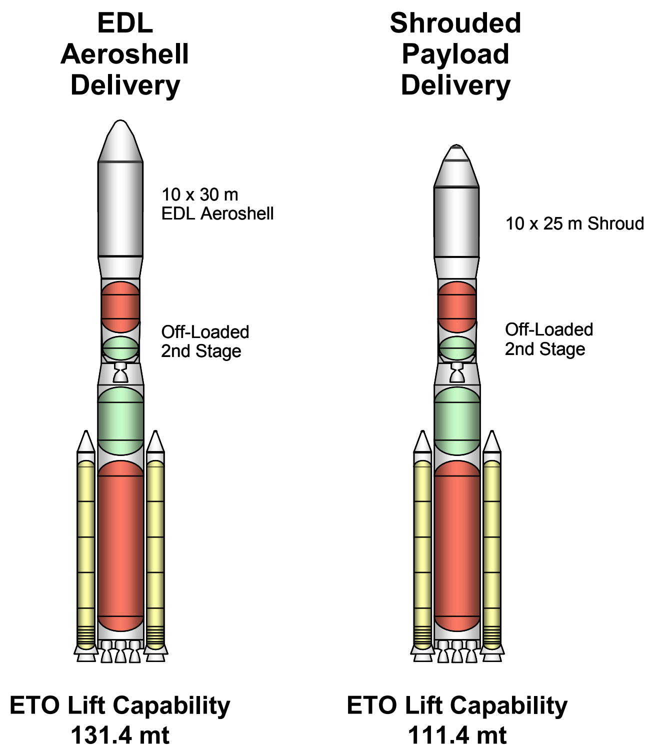



**Figure 5-39**. Mars transportation system Ares V Earth-to-orbit (ETO) configurations.

**Assembly sequence**

To define a launch assembly sequence and timeline, a number of issues must be considered. The first step is to take the individual module masses and determine the total number of launches that would be needed. Each launch is then required to meet the following two criteria: (1) the mass of each payload must be less than or equal to the lift capabilities of the launch vehicle that is being used; and (2) the volume of each payload must not exceed the payload volume that is allowed by the launch vehicle. After the launch masses and number of launches are known, the launch order can be determined. Next, the ground operations and on-orbit assembly operations were developed followed by the transit, destination, and return operations, which are dictated by the trajectory. Finally, the delivery of the initial components and launch vehicle were set such that the departure date in the operational timeline matched the departure date that was specified by the “worst-case” trajectory.

This baseline architecture (trade tree branch 3) requires 12 Ares V launches. The first seven launches are dedicated to assembling the two cargo vehicles simultaneously; the final five launches are dedicated to assembling the crew vehicle. Launches are assumed to occur every 30 days, so the cargo vehicles are assembled from day –390 to –210 days before the crew TMI date. The crew vehicle is assembled from day –180 to –60 days before the crew TMI date. This leaves 60 days of contingency in the vehicle assembly timeline. It should be noted that the cargo mission vehicles have a separate TMI date that falls within 11 to 60 days of the crew mission TMI date. It should also be noted that the cargo vehicles support the crew mission of the next mission opportunity. Figures 5-40 and 5-41 depict the specific modules that go up on each launch for the cargo and crew vehicles.

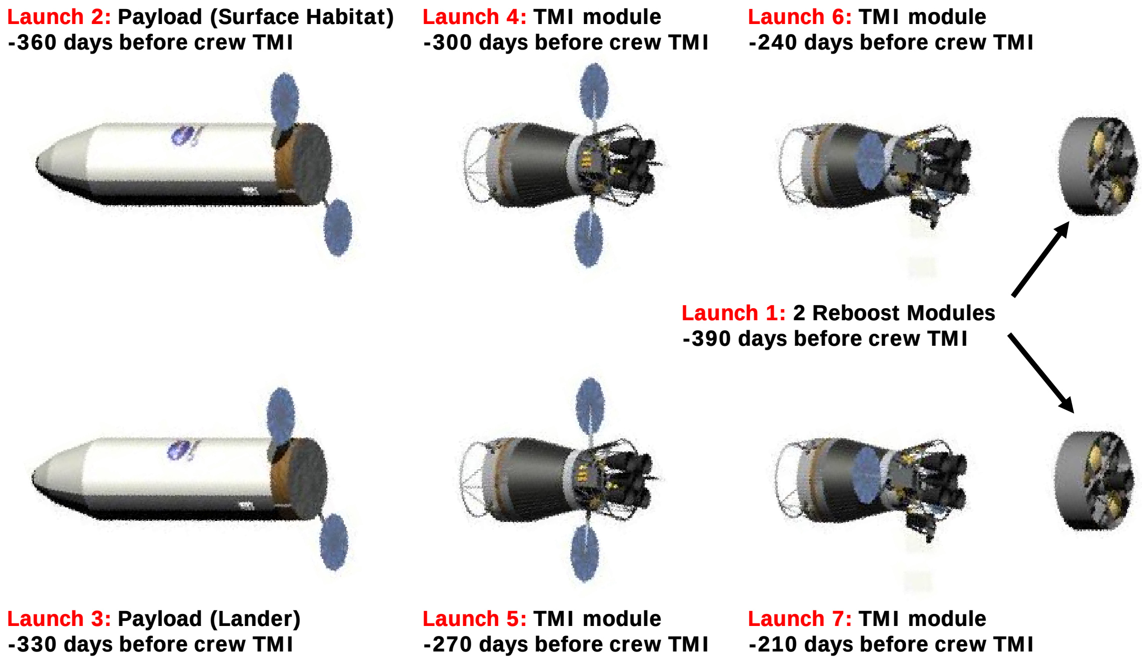



**Figure 5-40.** Launch assembly sequence and timeline for baseline case – cargo vehicle.

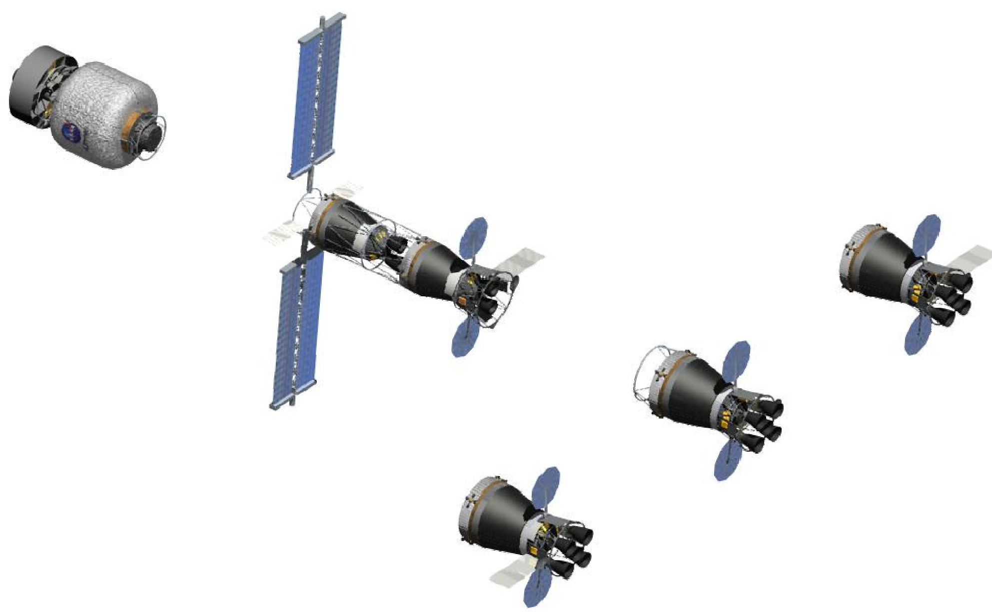


**Figure 5-41.** Launch assembly sequence and timeline for baseline case – crew vehicle.

#### *5.4.2.5 Mars transfer vehicle analysis ground rules and assumptions*

The tables on the following page list the major GR&As that were used in concept definition and vehicle analyses for the chemical propulsion architecture vehicle options. Table 5-14 lists the top-level vehicle design GR&As; and table 5-15 lists the trajectory and mission assumptions, including the mission delta-V budget.

**Table 5-14.** Chemical Propulsion Vehicle Design Ground Rules and Assumptions

| | |
|---|---|
| **ETO Shroud Size** | 10-m diameter x 25-m length |
| **Ares V ETO Lift Capability** | Gross: 131.3 t<br>w/Shroud: 111.4 t |
| **Delivery and Assembly Orbit** | 407 km x 407 km |
| **Mars Parking Orbit** | 250 km x 33,793 km (1 sol) |
| **Surface Power** | Nuclear for Cases 3, 6, 12<br>Solar for Cases 3a, 6a |
| **ISRU Used?** | ISRU for Cases 3, 3a<br>Non-ISRU for Cases 6, 6a, 12 |
| **Residuals and Ullage** | Ullage: 3% for main propulsion system (MPS) and 5% for RCS<br>Residuals: 2% for MPS and 3% for RCS |
| **Cryo-fluid Management** | ZBO and all active cooling |
| **Payloads – Cargo Mission (MOI Payload)** | Case 3: 103,000 kg<br>Case 3a: 113,000 kg<br>Case 6: 126,000 kg<br>Case 6a: 113,000 kg<br>Case 12: 106,842 kg |
| **Payloads – Crew Mission** | TransHab: 27,500 kg<br>Consumables: 2.45 kg/person/day<br>CEV: 10,000 kg |
| **Subsystem Growth** | 20% |
| **Main Propulsion System: TMI Stage (Reference Configuration** | Propellant Type: $LO_2/LH_2$<br>No. of engines: 5<br>Engine Type: RL10-B2<br>Nominal $I_{sp}$: 462.2 seconds<br>Thrust: 24,750 $lb_f$/engine |
| **Main Propulsion System: TMI Stage - Alternative (Application of Lunar Earth Departure Stage for TMI)** | Propellant Type: $LO_2/LH_2$<br>No. of engines: 1<br>Engine Type: J2-X<br>Nominal $I_{sp}$: 448 seconds<br>Thrust: 294,000 $lb_f$/engine |
| **Main Propulsion System: MOI Stage for Case 12 and TEI Stage for all cases** | Propellant Type: $LO_2/LH_2$<br>No. of engines: 2<br>Engine Type: RL10-B2<br>Nominal $I_{sp}$: 462.2 seconds<br>Thrust: 24,750 $lb_f$/engine |
| **Reaction Control System (Settling, Attitude, and Directional control burns)** | Propellant: MMH/NTO<br>No. of engines: 16 (4 banks of 4 thrusts, 2 axial, 2 lateral thrusters per bank)<br>Engine Type: Axial = R-40B, Lateral = R-42<br>Engine Thrust: Axial R-40B = 900 $lb_f$; Lateral R-42 = 200 $lb_f$<br>Nominal $I_{sp}$: Axial R-40B = 293 sec, |

**Table 5-15.** Chemical Propulsion Trajectory and Mission Analysis Ground Rules and Assumptions

| | **Long-stay Cargo Mission** | **Long-stay Crew Mission** |
|---|---|---|
| **Mission duration (days)** | Optimized | ~940 |
| **Transit times (days) – (Outbound/Inbound)** | Optimized | ~180/180 |
| **TMI $C_3$ ($km^2/s^2$)** | 16.7 | 18.4 |
| **Mars arrival $V_{inf}$ (km/s)** | 4.4 | 4.2 |
| **TEI $C_3$ ($km^2/s^2$)** | N/A | 14.8 |
| **Earth arrival entry speed (km/s)** | N/A | 13.5 |
| **Delta-V Budget:** (Assumed the worst-case delta-Vs for all mission opportunities from 2030 to 2046) | | |
| **Earth orbit circularization/rendezvous and cocking (m/s)** | 103/33 | 103/33 |
| **Settling burns (m/s)** | 1 | 1 |
| **TMI (m/s)** **(from 185-km LEO, gravity losses included)** | 4,053 | 4,140 |
| **Mid-course corrections (m/s per leg)** | 50 | 50 |
| **Earth/Mars coast A/C (m/s)** | 15 | 15 |
| **Propulsive MOI (m/s)** **(250x33,793-km Mars parking orbit (1 sol), gravity losses included)** | 1,350 | 1,789 |
| **Mars orbit rendezvous with lander (m/s)** | | 45 |
| **Mars Orbit Maintenance (m/s)** | | 100 |
| **TEI (m/s)** **(250x33,793-km Mars parking orbit (1 sol), gravity losses included)** | n/a | 1,573 |
| **Mars/Earth coast attitude control (m/s)** | | 15 |

#### *5.4.2.6 Chemical Propulsion Mars transfer vehicle analysis results*

Vehicle design concepts for each of the trade tree cases were developed using an integrated systems engineering analysis approach. Each major vehicle subsystem was defined based on the GR&A that was outlined in the previous section. The analysis was conducted using Level 1 system design models and analysis tools and a collaborative engineering analysis process. Each subsystem design was integrated into the overall vehicle design concept to ensure that all mission requirements were satisfied, and to produce an optimized transportation system. Table 5-16 provides a summary of the number of ETO launches, launch vehicle lift requirements, LEO assembly times and vehicle element masses for each of the trade tree cases that were evaluated in Phase 2 of this study. The sensitivities (i.e., "gear ratios") of MTV IMLEO to mass delivered to Mars orbit and mass returning from Mars are shown at the bottom of the table. It should be noted that Cases 6a and 12 slightly exceed the assumed Ares V lift capability, but this problem could be mitigated through further mission optimization for the worst-case Mars mission opportunities.

**Table 5-16.** Chemical Propulsion Vehicle Analysis Summary

| Trade Tree Branch:<br>Mars Orbit Capture:<br>Ascent Stage Propellant:<br>Surface Power: | Branch 3<br>Aerocapture<br>ISRU<br>Nuclear<br>Power | Branch 3a<br>Aerocapture<br>ISRU<br>Solar<br>Power | Branch 6<br>Aerocapture<br>ISRU<br>Nuclear<br>Power | Branch 6a<br>Aerocapture<br>ISRU<br>Solar<br>Power | Branch 12<br>Aerocapture<br>Non-ISRU<br>Nuclear<br>Power |
|---|---|---|---|---|---|
| **Ares V Payload Delivery Capability** | | | | | |
| **Direct Insertion to 407-km Orbit** | | | | | |
| **Gross Payload (EDL Delivery) (t)** | **131.4** | **131.4** | **131.4** | **131.4** | **131.4** |
| **Net Payload w/10x25-m Shroud (t)** | **111.4** | **111.4** | **111.4** | **111.4** | **111.4** |
| | | | | | |
| **MTS Analysis Results** | | | | | |
| **Ares V Launches** | **12.0** | **12.0** | **14.0** | **14.0** | **17.0** |
| **Ares V ETO Requirement for EDL Sys (10x30 m) (t)** | **103.0** | **113.0** | **126.0** | **133.0** | **122.0** |
| **Ares V ETO Requirement (w/10x25-m Shroud) (t)** | **108.5** | **108.5** | **108.5** | **108.5** | **112.6** |
| **LEO Assembly Time (days before TMI)** | **–390** | **–390** | **–450** | **–450** | **–540** |
| **Total IMLEO** | **1,252** | **1,303** | **1,420** | **1,454** | **1,775** |
| **Cargo Mission: IMLEO** | 359 | 384 | 443 | 460 | 620 |
| **No. of TMI modules** | 2 | 2 | 3 | 3 | 4 |
| **TMI Propellant per Module (t)** | 86 | 93 | 72 | 76 | 79 |
| **TMI Module Mass (t)** | 104 | 111 | 89 | 93 | 96 |
| **MOI Stage Mass (t)** | 0 | 0 | 0 | 0 | 64 |
| **Crew Mission: IMLEO** | 534 | 536 | 534 | 534 | 534 |
| **No. of TMI modules** | 3 | 3 | 3 | 3 | 3 |
| **TMI Propellant per Module (t)** | 91 | 91 | 91 | 91 | 91 |
| **TMI Module Mass (t)** | 109 | 109 | 109 | 109 | 109 |
| **MOI Stage Mass (t)** | 66 | 66 | 66 | 66 | 66 |
| **TEI Stage Mass (t)** | 43 | 43 | 43 | 43 | 43 |
| **TransHab Mass (t)** | 41 | 41 | 41 | 41 | 41 |
| | | | | | |
| **Gear Ratios** | | | | | |
| **Cargo: ΔMimleo/ΔMmoi** | 2.49 | 2.49 | 2.49 | 2.49 | 3.49 |
| **Crew: ΔMimleo/ΔMmoi** | 3.80 | 3.80 | 3.80 | 3.80 | 3.80 |
| **Crew: ΔMimleo/ΔMtei** | 6.12 | 6.12 | 6.12 | 6.12 | 6.12 |

#### *5.4.2.7 Application of lunar Earth departure stage for trans-Mars injection*

Application of the Ares V lunar mission EDS as a TMI module provides potential programmatic synergy between the Lunar and Mars mission architectures by eliminating the need to develop a completely new propulsive TMI stage. The lunar EDS would be augmented with cryogenic fluid management and other systems necessary to function during the LEO assembly period and perform the TMI maneuvers. There would be no changes to the EDS propellant tanks or the main propulsion system. The added vehicle systems that would be needed during the LEO assembly period would be located on a LEO loiter skirt mounted below the $LO_2$ tank. This loiter skirt would be jettisoned prior to TMI. The EDS/TMI modules would be launched to the LEO assembly orbit in the configuration that is shown in figure 5-42, with a nosecone on top of the EDS/TMI stage. The module would be launched with a full propellant load of 999 t of propellant. Since the EDS/TMI module also serves as the second stage of the Ares V, 222 t of propellant must be expended to perform a suborbital burn and a circularization burn to reach the LEO assembly orbit. This leaves 113 t of propellant for the TMI maneuvers. The number of TMI modules that would be required and other MTV summary information are listed in table 5-17.

**Figure 5-42.** Common lunar Earth departure stage/trans-Mars insertion module concept.

**Table 5-17.** Sizing Summary Using a Common Lunar EDS/Trans-Mars Injection Module

| Trade Tree Branch:<br>Mars Orbit Capture:<br>Ascent Stage Propellant:<br>Surface Power: | Branch 3<br>Aerocapture<br>ISRU<br>Nuclear<br>Power | Branch 3a<br>Aerocapture<br>ISRU<br>Solar<br>Power | Branch 6<br>Aerocapture<br>ISRU<br>Nuclear<br>Power | Branch 6a<br>Aerocapture<br>ISRU<br>Solar<br>Power | Branch 12<br>Aerocapture<br>Non-ISRU<br>Nuclear<br>Power |
|---|---|---|---|---|---|
| **Ares V Payload Delivery Capability** | | | | | |
| **Direct Insertion to 407-km Orbit** | | | | | |
| **Gross Payload (EDL Delivery) (t)** | **131.4** | **131.4** | **131.4** | **131.4** | **131.4** |
| **TMI Propellant (t)** | **113.0** | **113.0** | **113.0** | **113.0** | **113.0** |
| **Net Payload w/10x25-m Shroud (t)** | **111.4** | **111.4** | **111.4** | **111.4** | **111.4** |
| | | | | | |
| **MTS Analysis Results** | | | | | |
| **Ares V Launches** | **12.0** | **12.0** | **14.0** | **14.0** | **15.0** |
| **Ares V ETO Requirement for EDL Sys (10x30 m) (t)** | **103.0** | **113.0** | **126.0** | **133.0** | **122.0** |
| **Ares V ETO Requirement (w/10x25-m Shroud) (t)** | **108.5** | **108.5** | **108.5** | **108.5** | **112.6** |
| **LEO Assembly Time (days before TMI)** | **–390** | **–390** | **–390** | **–450** | **–480** |
| **Total IMLEO** | **1,391** | **1,440** | **1,503** | **1,630** | **1,890** |
| **Cargo Mission: IMLEO** | 399 | 423 | 455 | 518 | 648 |
| **No. of TMI modules** | 2 | 2 | 2 | 3 | 3 |
| **TMI Propellant per Module (t)** | 96 | 103 | 113 | 85 | 110 |
| **TMI Module Mass (t)** | 124 | 131 | 140 | 112 | 138 |
| **MOI Stage Mass (t)** | 0 | 0 | 0 | 0 | 64 |
| **Crew Mission: IMLEO** | 594 | 594 | 594 | 594 | 594 |
| **No. of TMI modules** | 3 | 3 | 3 | 3 | 3 |
| **TMI Propellant per Module (t)** | 101 | 101 | 101 | 101 | 101 |
| **TMI Module Mass (t)** | 128 | 128 | 128 | 128 | 128 |
| **MOI Stage Mass (t)** | 66 | 66 | 66 | 66 | 66 |
| **TEI Stage Mass (t)** | 43 | 43 | 43 | 43 | 43 |
| **TransHab Mass (t)** | 41 | 41 | 41 | 41 | 41 |
| | | | | | |
| **Gear Ratios** | | | | | |
| **Cargo: ΔMimleo/ΔMmoi** | 2.49 | 2.49 | 2.49 | 2.49 | 3.49 |
| **Crew: ΔMimleo/ΔMmoi** | 3.80 | 3.80 | 3.80 | 3.80 | 3.80 |
| **Crew: ΔMimleo/ΔMtei** | 6.12 | 6.12 | 6.12 | 6.12 | 6.12 |

### *5.4.2.8 Lunar and Mars architecture subsystem commonality*

It is highly desirable to maximize the synergy between the lunar and Mars transportation architectures. This potential synergy consists of the use of common ETO launch vehicles, space transportation elements, and subsystem technologies. Although the lunar and Mars missions are very different, there is considerable potential for synergy. The Mars mission architecture can build on the lunar mission systems and vehicles. As part of this study, a comparison of the lunar EDS and TMI stage subsystems was performed to identify potential commonality between the two systems.

**Trajectory and mission analysis**
The LEO loiter time for the Mars mission will be longer due to the nature of the assembly. Hence, the RCS delta-V requirements – i.e., attitude and control as well as any re-boost capability that may be needed during that time – will be greater to handle. Also, there will need to be a larger allowance for mid-course correction maneuvers for the Mars mission. Since the TLI burn varies from the TMI burn, there will be a difference in the MPS delta-V requirements.

**Propulsion**
Depending on the final Ares V design, several similarities exist between the Ares V upper stage (i.e., EDS) and the various in-space stages that are being considered for the human Mars mission. The EDS and the Mars TMI, MOI, and TEI stages all use the $LO_2/LH_2$ propellant combination. All propulsion systems also contain pump-fed engines and include helium and/or autogenous tank pressurization systems.

One possible difference between the Ares V and Mars stages is engine choice. The Mars propulsive stages employ multiple RL10-B2 engine systems, whereas the current preliminary EDS design uses a single J-2X engine. The RL10-B2 is an expander cycle engine that produces a nominal thrust of 24,750 $lb_f$ at an $I_{sp}$ of 465.5 seconds. The current J-2X engine specifications state a much higher thrust of 294,000 $lb_f$ with a 448-second $I_{sp}$ rating. The engine cycle of the J-2X has yet to be specified; the original J-2 incorporated a gas generator cycle, while the J-2S version used a tap-off cycle. Both the RL10-B2 and J-2X engines are (or "will be," in the case of the J-2X) produced by Pratt & Whitney Rocketdyne, which is a United Technologies Company.

**Structures**
Although the launch and staging loads will size most components, some other differences will affect the overall structure. Components must be designed for end-of-life material properties, which means the effects of thermal cycling and possible material degradation in Mars orbit must be considered. The potential for composite material issues such as outgassing and micro-cracking during launch as well as fatigue exist. Lastly, increased MMOD protection would be needed for the Mars mission since there would be an increased risk of impact with non-negligible objects.

**Thermal**
The following thermal features are common between the lunar EDS and the Mars TMI stage: the heat collection and transport systems (e.g., circulating fluid systems, cold plates, and heat exchangers), heat rejection systems (e.g., space radiators), and the passive control systems (which include heaters, insulations, passive devices, phase-change materials, and special materials and coatings). However, the lunar EDS is designed with passive cryogenic propellant storage while the Mars TMI stage will require ZBO active cryogenic propellant storage. The development of a two-stage cryocooler operating at 20 K for $H_2$ storage will be required for the Mars mission active cryogenic propellant storage. The active approach affords essentially indefinite storage duration at the price of increased complexity, system hardware mass, power consumption, and heat rejection.

**Power**
Both the lunar EDS and the Mars TMI stages are powered by solar arrays. The lunar EDS contains a power management and distribution (PMAD) system to power itself and the lander during LEO loiter and TLI. There is no cryogenic cooling requirement on the EDS; it has an all-passive thermal system. On the other hand, the TMI stages will contain solar arrays and a PMAD system to power themselves only during LEO loiter and TMI, and have no payload power capability. Since the TMI stages are actively cryogenic cooled, however, they will require significantly more power than the EDS (about four times more), which means a much larger PMAD system than in the EDS. Also, because of the longer assembly and loiter time in LEO, the TMI solar arrays will need to handle more Radiation degradation and MMOD damage.

**Avionics**
The following control features are common between the lunar EDS and the Mars TMI stage: Both designs contain solar power generation and have navigation and communication components for LEO (i.e., sun and star trackers, IMUs, S-band transceivers, global positioning system (GPS), etc.). However, there are several differences. The instrumentation and data handling components will need to be larger in capacity in the TMI for the Mars mission, in particular to accommodate the active thermal cooling system. Also, the TMI requires a rendezvous and docking system and low-gain inter-stage communication system for docking and assembly operations. The lunar EDS avionics system provides the GN&C capabilities of the stack during LEO loiter. It has a one-fault-tolerant system since the CEV takes over navigation control after docking, providing two-fault tolerance for the crewed TLI operation. The Mars TMI stage avionics systems also consists of a one-fault-tolerant GN&C system to navigate the stage to assembly orbit and dock with the stack, but the re-boost module will provide GN&C for the stack during assembly and testing. The TEI stage will be the controlling element during TMI burns and thereafter, and will provide the two-fault-tolerant GN&C that is required with a crew aboard. These operational differences mean that the two avionic systems will have differences in system architecture, interconnections, and software.

#### *5.4.2.9 Chemical propulsion vehicle analysis conclusions*
One of the most significant FOMs for evaluating the Mars architecture options is the number of ETO launches that will be required for a complete Mars mission. This would include the two cargo missions and the crew mission. The number of launches is driven by the 25-m payload shroud length, or by the number of TMI modules. Minimizing the number of launches could be achieved by attempting to limit the number of TMI modules to just two per vehicle. For the architecture options that were considered in this study, the vehicles require two to four TMI modules. Cases 3 and 3a are only one launch above the minimum, Cases 6 and 6a are three launches above the minimum, and Case 12 is five launches above the minimum. Using the lunar EDS as a TMI module has the potential to reduce the number of launches for Cases 6 and 12 by two launches.

The assumed Ares V gross LEO payload capability of 131 t allows close to the minimum number of ETO launches for each Mars mission. The minimum number of launches for the cargo mission can be achieved if the addition of an EDL system mass keeps the total payload below 114 t, as in Cases 3 and 3a. The minimum number of launches for the crew mission can be achieved if the propellant capacity of the TMI modules is sufficiently large, such that only two modules are required. This would result in a larger mass for the TMI modules and require an increases ETO lift capability. Table 5-18 shows the number of ETO launches for the architecture options that were considered in this study, which were based on the assumed reference Ares V capability, and also the required Ares V capability that is needed to achieve the minimum number of launches.

**Table 5-18.** Number of Earth-to-Orbit Launches Required for Chemical Propulsion MTV

| Trade Tree Branch: | Case 3 | Case 3a | Case 6 | Case 6a | Case 12 |
|---|---|---|---|---|---|
| Reference Ares V Capability: (Gross LEO Payload, t) | 131 | 131 | 131 | 131 | 131 |
| No. of ETO Launches, Reference DRA 5.0 | 12 | 12 | 14 | 14 | 17 |
| No. of ETO Launches, Lunar EDS/TMI Module | 12 | 12 | 12 | 14 | 15 |
| | | | | | |
| Ares V Gross LEO Payload Capability Required for Minimum Number of Launches | 167 | 167 | 167 | 167 | 185 |
| Minimum Number of Launches Possible (w/25-m payload shroud) | 11 | 11 | 11 | 11 | 12 |

## *5.5 Crew Exploration Vehicle/Earth Return Vehicle*
The ESAS)reference Mars mission called for a “Block 3 CEV” (a future upgrade of the Orion vehicle that is currently under development) to transfer a crew of up to six between Earth and an MTV at the beginning and end of the Mars exploration mission. A Block 3 CEV (Command Module and SM with a 3-year in-space certification) is launched by the Ares 1 into an orbit, matching the inclination and altitude (~420 km) of the loitering MTV stack. It then takes the CEV up to 2 days to perform orbit-raising maneuvers to close on the MTV, conducting a standard ISS-type rendezvous and docking approach to the MTV. After docking, the CEV crew performs a leak check, equalizes pressure with the MTV, and opens hatches. Once crew and cargo transfer activities are complete, the CEV

is configured to a quiescent state and remains docked to the MTV for the trip to Mars and back. Periodic systems health checks and monitoring are performed by the ground and flight crew throughout the mission.

As the MTV approaches Earth upon completion of the 1.5- to 2.5-year round-trip mission, the crew performs a pre-undock health check of all entry-critical systems, transfers to the CEV, closes hatches, performs leak checks, and undocks from the MTV. The CEV departs 24 to 48 hours prior to Earth entry, at which time the MTV will most likely perform a diversion maneuver to sail harmlessly past Earth. After undocking, the CEV conducts an on-board-targeted, ground-validated burn to target for the proper entry corridor; and as entry approaches, the CEV CM maneuvers to the proper entry interface (EI) attitude for a direct-guided entry to the landing site. The CEV performs a nominal landing at the primary land-based landing site, and the crew and vehicle are recovered. Earth entry speeds from a nominal Mars return trajectory may be as high as 14 km/s, compared to the approximately 11 km/s for the Block 2 (lunar) CEV. This difference will necessitate either the development of a higher-density, lightweight TPS or the use of the CEV SM to perform a burn that slows the velocity of the vehicle as it approaches EI. The suggestion coming out of DRA 5.0 was to limit the speed at EI to 12 km/s. Future studies must address specifically how this will be done (e.g., properly size the service module of the ERV to perform the burn). Details of the TPS requirements as a function of trajectory-return velocities are provided below in section 5.5.1. A further discussion regarding the TPS technology development challenges of the Block 3 upgrade is provided in section 7.2.5.

Two other factors (besides the primary concern of Earth entry speed) will drive the evolution of the CEV from a Block 2 lunar vehicle to a Block 3 Mars vehicle. The first is the need to recertify the Orion for an approximately 3-year on-orbit lifetime. This is a non-trivial effort that will focus mainly on upgrading the power system and the TPS. Additionally, a science-driven mission to Mars will likely result in the desire to bring back an adequate amount of martian material. Given the gear ratios that are involved in a round-trip to Mars, the mass of such material would either have to be kept to a minimum, or else the Block 3 upgrade would have to adopt an undetermined strategy by which to accommodate the mass and volume of this scientific material.

It was not within the scope of the DRA 5.0 activity to recommend specific design upgrades for the Block 3 vehicle or to scope out an upgrade strategy. Instead, a mass estimate of approximately 10,000 kg was used for the vehicle command module to size propulsion stages. An additional approximately 4 t was book- kept for an SM, which may be needed to perform an Earth-targeting burn. Future activities, which will likely be in conjunction with the Orion Project Office, will better define an upgrade strategy. A conceptual drawing of an ERV that is based on a future block upgrade of the Orion vehicle is shown in figure 5-43.

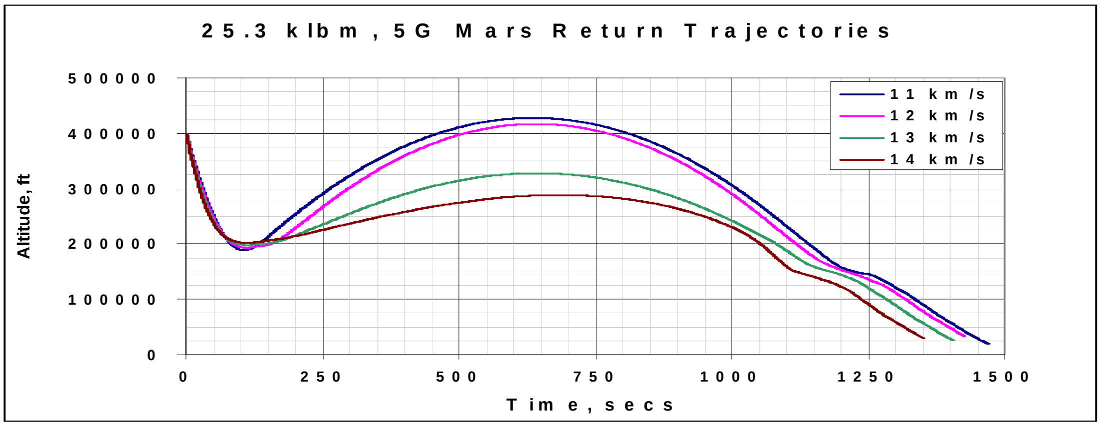
**Figure 5-43.** Conceptual Earth return vehicle based on a future block upgrade of the Orion.

### 5.5.1 Earth return vehicle thermal protection system requirements

As previously alluded to, the ESAS specified a Mars CEV variant (Block 3) that would have the same requirements as the Block 2 version of Orion with the following exceptions: Block 3 will be outfitted for six people plus additional (TBD) cargo, and would assume no fewer than 2 days of stand-alone free-flight capability. Since Orion Block 3 is a Mars return vehicle, its entry speed may be significantly higher than that for a lunar return at 11 km/s. Furthermore, since the crew is six rather that the four who are being returned from the moon, a heavier vehicle was studied than a Block 2 vehicle: 11,531 kg vs. that for the May 2007 point of departure (POD) Orion vehicle at 9,258 kg. The current Block 3 Orion vehicle has a maximum base diameter of 5 m, which is consistent with the May 2007 POD Block 2 version of Orion.

### 5.5.2 Mars Earth return – entry trajectories

Figure 5-44 depicts the Mars return, direct entry trajectories that span the probable range of speeds (11 to 14 km/s) for this mission. The reasonable upper limit of the Earth entry speeds is 13 to14 km/s, with the higher entry speeds serving to reduce the Mars-to-Earth transit time, which limits the exposure of vehicle and crew to in-space hazards from solar flares, radiation, and micrometeorite impacts. As stated above, reducing the ERV, which is also known as the Mars CEV, entry velocity to 12 m/s may lessen the need for advanced TPS. Two trajectories for the 11,531-kg vehicles were considered, one with the accepted maximum of 5 Earth g-loads for crewed vehicles and another held at no more than 8 g's as a limiting case.



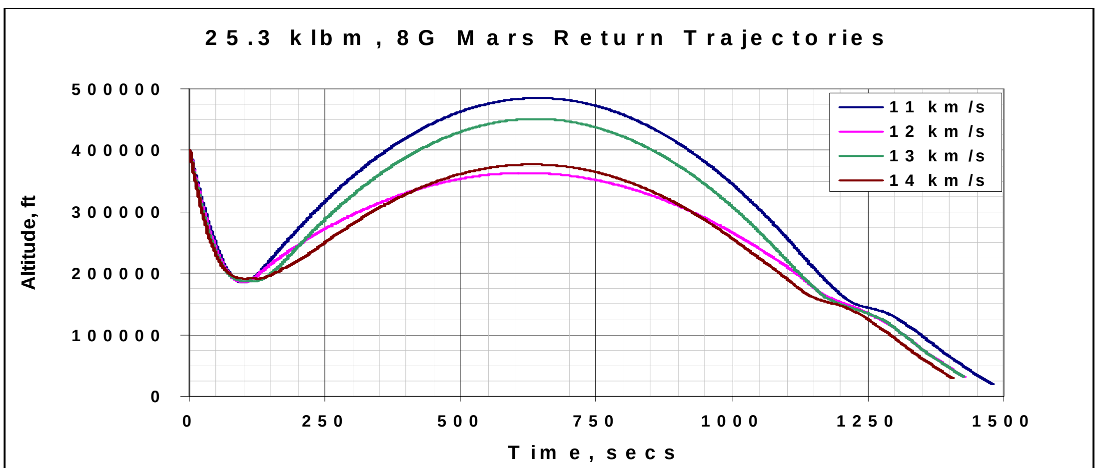




**Figure 5-44.** Orion Block 3 Mars-Earth return trajectories.

### 5.5.3 Derived requirements: vehicle environments

Figure 5-45 depicts a contour map of the peak surface heating rates for the Orion Block 3 vehicle at Mach 41, for an angle-of-attack of 157 degrees along the 14 km/s trajectory. As can be seen from the figure, the heating rates for this condition are in excess of 1,500 W/cm$^2$, about a factor of 1.5 times the Orion Block 2 conditions at approximately 1,000 W/cm$^2$ with full margin and a factor of four higher than the comparable values for the smaller Apollo lunar return vehicle at approximately 400 W/cm$^2$. It is very important to note that the profile includes no margins, as are currently being carried for the Orion Block 2 design.

To account for uncertainties in the Orion aerothermal environments for Mars return to those in current use by the Block 2 community (i.e., the CEV Aerosciences Project (CAP) and the CEV TPS Advanced Development Project (ADP)) have been adopted are the following multiplicative factors:

- .35 on convective heating
- 2.00 on radiative heating
- 1.10 on total heating rates for trajectory dispersions
- 1.35 on heating loads for trajectory dispersions

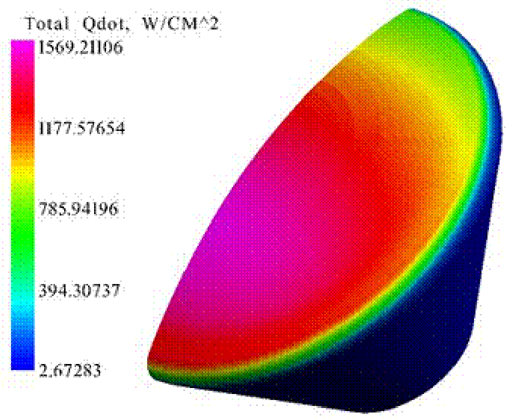



**Figure 5-45.** Orion Block 3 Mars-Earth return forebody surface heat flux.

The heating rates that were computed as a function of time during entry for the 11,531-kg Orion vehicle with maximum g-loading of 5 g's are depicted in figures 5-46, 5-47, and 5-48. As can be seen from figure 5-46, the peak heating for the sum of radiative and convective heating varies from around 1,000 W/cm$^2$ at the lunar return speed of 11 km/s, and increases substantially to over 3,000 W/cm$^2$ at the highest speed considered of 14 km/s. It is important to note that the heating values, with margins, are about a factor of 7.5 times that for Apollo at lunar return speeds of 11 km/s.

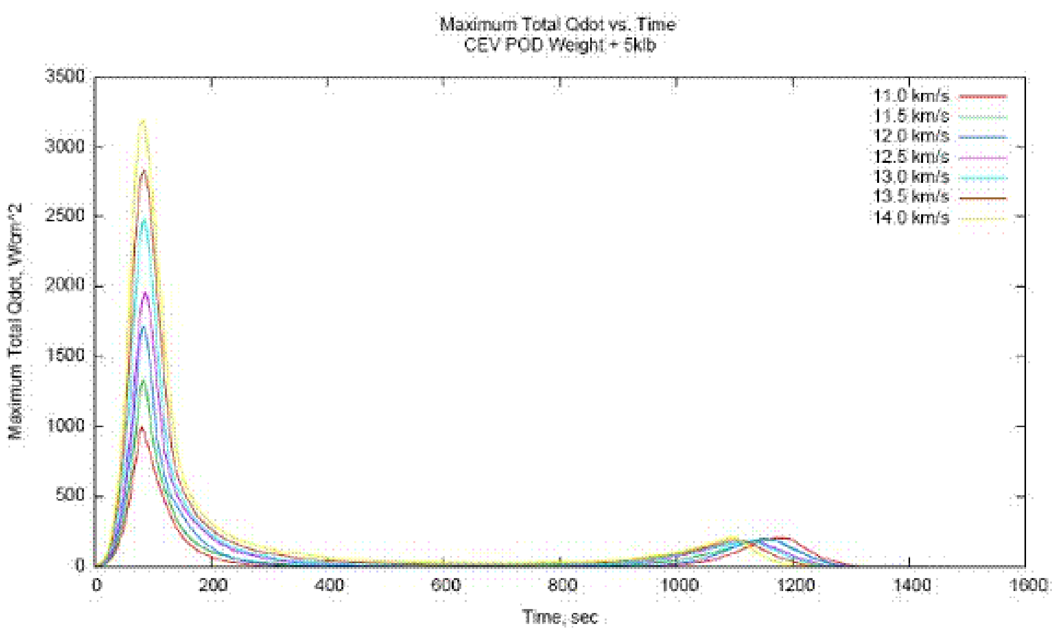



**Figure 5-46.** Maximum total heating for Orion Block 3.

**Figure 5-47.** Maximum convective heating for Orion Block 3.

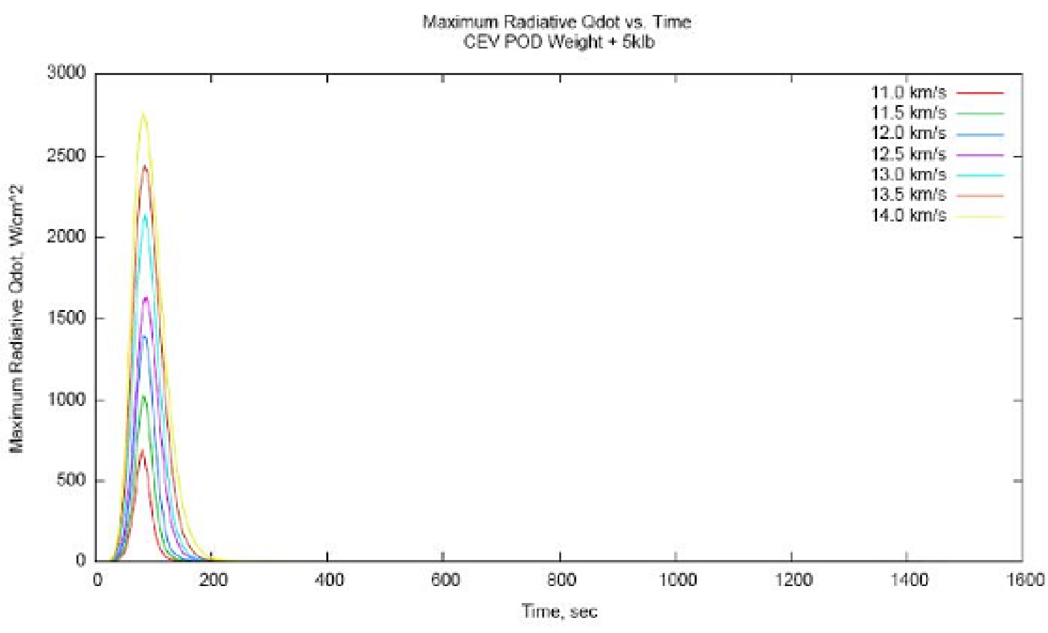



**Figure 5-48.** Maximum radiative heating for Orion Block 3.

The components of convective and radiative heating rates, with margins, are shown respectively. Note that these are margined, peak heating values that do not occur on the same physical location on the surface of the vehicle. From figures 5-47 and 5-48, it can be seen that the increase in convective heating going from 11 to 14 km/s is a factor of about 1.7 while the increase in radiative heating for these two speeds is nearly a factor of four. As is discussed below, at the higher Mars return entry speeds shock layer radiation becomes the driver for the heat loads and, consequently, TPS mass.

Figure 5-49 depicts the effect of g-constraints and vehicle mass on peak heating rates as a function of entry speed. The red dot at 11 km/s and mass of 9,227 kg corresponds to the Orion Block 2 vehicle with a heating rate of slightly less than 1,000 W/cm$^2$. Note that these heating rates are margined, as specified above. As can be seen, significant increases in heating rates are introduced as the speed increases from 11 to 14 km/s while increasing maximum g’s at a given entry speed result in less severe augmentations.

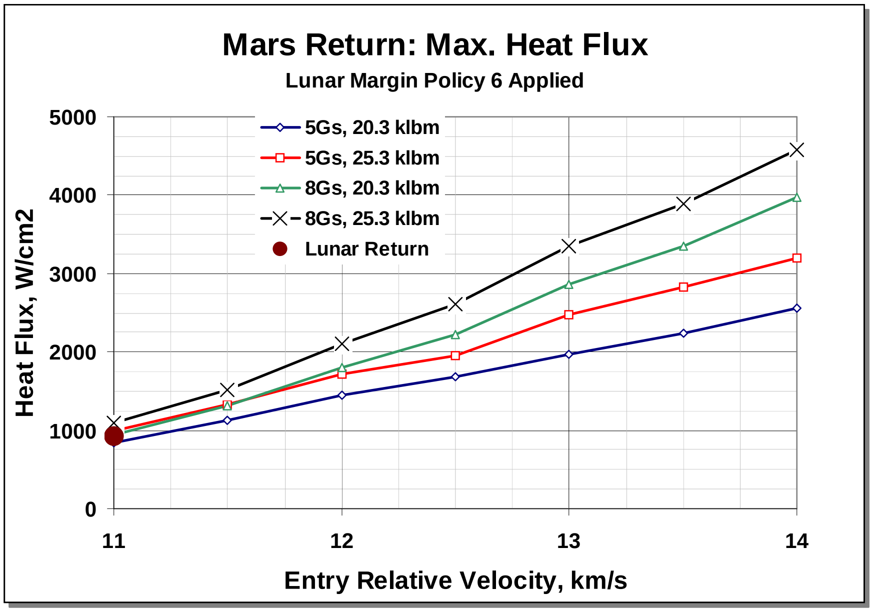



**Figure 5-49.** Effect of g-constraint and entry velocity on maximum total heating rate

Following the format for heating rates, figure 5-50 depicts the maximum heat loads as a function of entry speed. Again, the red dot corresponds to the Orion Block 2 vehicle at a value of slightly more than 1,000 MJ/$m^2$. Importantly, note that these heat loads account for the margins that are specified above. Following the trends for heat rates, increases of a factor of three are shown for heat load and less significant increases are noted in going from maximum g's of five to eight.

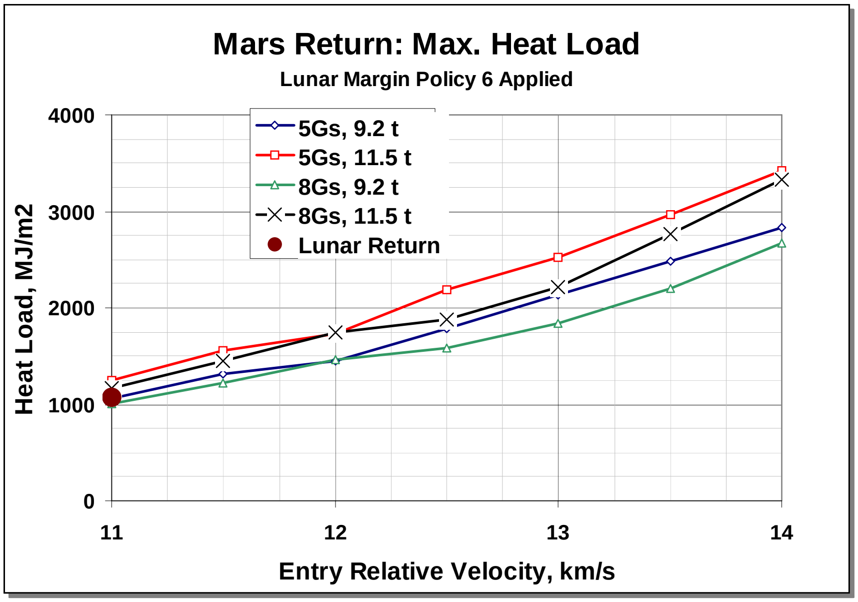



**Figure 5-50.** Effect of g-constraint and entry velocity on maximum total heat load

### 5.5.4 Thermal protection system sizing

Beginning with the margined aerothermal environments that were discussed in the previous section, the TPS sizing tools used by the CEV TPS ADP were applied to the Mars return cases (Orion Block 3)

As discussed earlier, the heating environment during Earth return is a strong function of the entry speed and also is a function of the location on the Block 3 vehicle. The heatshield will experience considerably higher heating, both heatflux and heatload, as compared to the Orion Block 2 (lunar return). If we are limited to use TPS materials with previous flight heritage, the choices are Carbon Phenolic, PICA and the Apollo material AVCOAT* for the heat shield. Similar to the heat shield, the aft shell also experiences relatively higher heating compared to Orion's Block 2 aft body. As a result, higher heatflux capable TPS for both the heat shield and the aft body had to be considered. SLA and BRI-8 were found to be adequate for the aft shell. SLA has been used on all the Mars Missions as heat shield material and also used on Stardust aft shell. BRI-8 is used on the Shuttle Orbiter. For the heat shield, Carbon Phenolic, a high density, robust but very heavy TPS material was considered along with PICA.

In order to perform sizing computations to determine the thickness of the TPS on the heat shield and the aft shell, boundary conditions such as bond line temperature, structural thermal mass underneath the TPS assumed were very similar to that of Orion Block 2 TPS sizing.

Once the TPS materials were selected and mapped on to the surface based on the material capability and the local heating environment, the sizing computations were performed to determine the region of applicability as well as the local thickness of the TPS. For 11 km/s, on the aft body, BRI-8 was found to be applicable for the majority of the region, and in higher heating regions, SLA was found to be adequate and sized accordingly (See figure 5-51). At higher entry speeds, SLA was required for a larger fraction of the aft body.

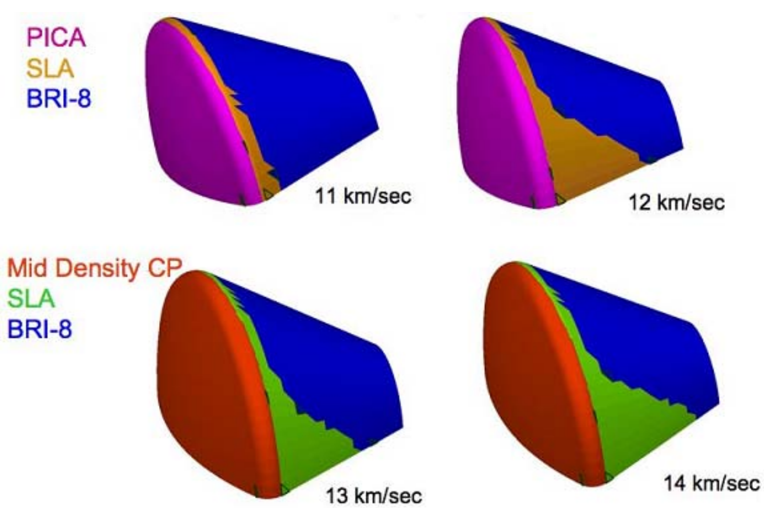
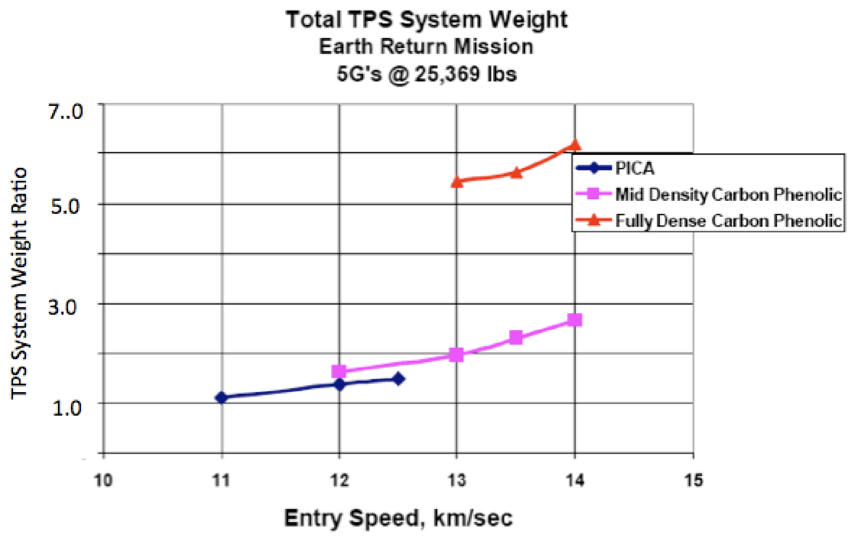



**Figure 5-51.** Thermal protection system split lines as a function of entry velocity.




Note: The TPS mass was scaled by the mass of the TPS system at 11 km/s

**Figure 5-52.** Orion Block 3 thermal protection system mass as a function of entry speed.

Carbon Phenolic, selected as a candidate for the heat shield, is a relatively very high-density ablator and it was used as heat shield material on the Galileo and the Pioneer Venus probes. Though the heating environment on Orion Block 3 vehicle for the Mars return conditions were not estimated to be as severe as that of Galileo or Pioneer Venus missions, Carbon Phenolic is the only material that can withstand the entry heating across the entire entry speed ( 11 km/s – 14 km/s) studied here. It is not an ideal candidate and resulted in relatively higher mass for the heat shield as a function of velocity. As compared to Carbon Phenolic, PICA has very limited applicability and hence at condition much beyond Lunar return (~11.5 km/s), PICA cannot be used. When applicable, PICA resulted in the lowest mass solution. As a result of the trade studies, we concluded that the development of a mid-density Carbon-Phenolic ablator can lead to both robust as well as mass efficient solution for the entire 11 km/s – 14 km/s range. A thermal response model based on theoretical considerations were constructed for a yet-to-be developed mid-density Carbon Phenolic ablator and applied across the range of entry conditions. The sizing study showed that mass savings between (200% - 300%) is likely with the mid-density ablator as compared to Carbon Phenolic (see figure 5-52)

> **Note: Although AVCOAT has been baselined as the TPS material for Orion Block 2 recently, during the performance of this study, PICA was the baseline for Lunar return conditions by Orion and hence PICA was considered as a candidate here for Block 3. AVCOAT has limitations similar to that of PICA and cannot be used at entry speeds for beyond 11.5 km/s. Although we did not size AVCOAT at entry speeds close to 11 km/s, the TPS system weight will not be significantly different than that of PICA.*

## *5.6 Entry, Descent, and Landing*

Several EDL configuration architectures were considered during this study. They included an all-propulsive entry with no aeroassist elements that was not selected because of the large orbit-to-landed-payload mass fraction (on the order of eight) that was required for the payload masses that were considered. Supersonic aerodynamic decelerators, including parachutes and inflatable aerodynamic devices, were also considered for use in the descent phase, but the performance and mass models for the scale and dimensions that are required for the systems in this study were felt to be lacking in sufficient detail to be considered here. Extrapolations in performance and masses from the references that were available were too large for these technologies to be weighed as viable options in the trade space. It is strongly recommended, however, that future development of improved models for these types of systems technologies be pursued so that credible trades can be conducted and more-optimal EDL system performance and reliability improvements can be realized. More discussion on this topic will be provided in section 7 of this report. The reference EDL architecture that was ultimately selected for this study was a hypersonic aeroassisted entry system with a mid L/D aeroshell that was ejected at low supersonic Mach numbers. An $LO_2/CH_4$-fueled propulsion system was used for deorbit delta-V maneuvers, RCS control during the entry phase, and final terminal descent to the surface.

An EDL parametric trade space, which is similar to the aerocapture parametric trade space that was defined in section 3.5.3, was defined to bound the estimated effective payload landed masses and architecture design options. The baseline EDL system design was developed using a 10-m × 30-m aeroshell and a reference Mars orbit with a 1-sol period and an apoapsis of 33,793 km. EDL system designs were developed for both the AC and propulsive MOI from cruise cases because the capture method affects the TPS masses, as described in section 3.5.3. Systems were sized for three landed, useful payload masses of 20, 40, and 70 t, which would cover the assumed range of Mars surface systems that are required for the human Mars architecture. In the case where aerocapture was used to achieve Mars orbit, the same aeroshell was used for both the aerocapture and the EDL phase, although additional TPS mass was required to accommodate the additional heating environment that was associated with the aerocapture maneuver. A pseudo-guidance methodology was developed to provide a realistic entry profile that would minimize terminal descent propulsive fuel requirements as well as the TPS mass and land the vehicle at 0 km above the Mars orbiter laser altimeter (MOLA). A summary of the EDL system parametric space is provided in table 5-18. What follows is a description of the entry aerothermal environment, TPS sizing, mass models, and parametric studies that contributed to the selection of the baseline configuration.

**Table 5-19.** Summary of Parameter Space

| Parameter | Values | | |
|---|---|---|---|
| Capture method | Aerocapture | Propulsive | |
| Aeroshell Size (m) | 10×30 | 12×35 | |
| Payload Mass (kg) | 20 | 40 | 70 |
| Initial Orbit | 500 km | 24 hours | |
| Entry profile hold constant deceleration (g's) | 1 | 2 | 3 |
| Delta-V margin (% of total) | 30 | 50 | 100 |
| Thrust-to-Weight of Engines ($lb_f/lb_m$) | 40 | 80 | |

The key to determining the EDL system mass of a human-scale mission to Mars is the vehicle components that are considered and the fidelity of the models that are used to approximate the mass of each. The major components of the entry vehicle that were considered here include the aeroshell structure, the TPS, the reaction control system (RCS), the descent stage structure, the terminal propulsion system including propellant tankage and plumbing, and the useful landed payload. A summary of the mass models that were used for each of the major components is provided, including a breakdown of component masses and propellants as well as key trajectory characteristics for this study. In all cases, a 30% margin allocation was assumed on structural mass, and a 20% margin was assumed on TPS masses (based on Orion/CEV "heritage" material and design principals). A 30% margin on terminal descent propellant was levied to account for terminal descent hazard avoidance/pinpoint landing. The entry trajectories were simulated using POST2 (Striepe, et al., 2004[19]).

### 5.6.1 Aeroshell structural sizing

The aerocapture and entry aeroshell structure mass estimates were made using preliminary estimates and guidance from the Ares V launch vehicle shroud development efforts. Initial ellipsled/biconic aerocapture and EDL aeroshell structural mass estimates were based on equivalent-area Ares V payload shroud mass sizing plus a 50% margin to allow for the additional lateral loads that are associated with entry and descent, TPS attachment scar mass, heat soakback, etc. Initially, two aeroshell geometries were developed: a 10-m × 30-m and a larger-scaled 12-m × 36-m variant. Once the packaging and volume requirements for the descent stage and lander systems were defined, a downselect was made to baseline the 10-m × 30-m aeroshell configuration. The Ares V shroud parametric mass data

[19]Scott A. Striepe, et al., "Program to Optimize Simulated Trajectories (POST II) – Utilization Manual, Volume II, Version 1.16.G," NASA Langley Research Center, January 2004.

are shown in table 5-17, with the equivalent-area 10-m × 30-m and 12-m × 36-m entry aeroshell shroud points. The mass estimates that were based on the Ares V shroud parametric data for the aeroshell shroud structure, absent any TPS, were 22.5 t and 42.5 t for the 10-m × 30-m and the 12-m × 36-m versions, respectively. More detailed structural sizing and additional load cases were defined, which resulted in confirmations of these estimate of the aeroshell mass of 22.5 t for the 10-m × 30-m aeroshell.

### 5.6.2 Thermal protection system sizing

The TPS analysis trade studies and sizing were conducted by personnel who were involved in the Orion CEV TPS ADP, or who were contracted to just the ADP herein. The focus of the ADP is on developing preliminary heat shield designs for the Orion Block 1 (LEO return) and Block 2 (hypervelocity lunar return) configurations. The tools for the current Mars DRM TPS sizing effort are those that are now in use by the ADP for the lunar return Block 2 heat shield design, with modifications that are appropriate for hypervelocity flight in the Mars atmosphere. Two cases were analyzed: the first, which includes aerocapture, cool down in orbit, and subsequent entry and entry only, each for a reference 24.6-hour (1-sol) orbit; and the second, which takes place from a 500-km circular orbit. In all cases, the system was sized to delivered a 40-t effective payload to the surface.

The aerothermal environments that are associated with the Mars aerocapture and entry trajectories were determined using the NASA CBAero tool (Kinney, et al., 2006[20]), which was modified for use in the martian atmosphere. The database for the CBAero code was developed from a sparse set of high-fidelity, real-gas CFD solutions from the DPLR code (Wright, et al., 2005[21]), combined with the line-by-line radiative heating code NEQAIR[22] to provide predictions of convective and radiative heating solutions. Each solution contained full-surface aerothermal environments including surface pressure, temperature, shear and uncoupled, and convective and radiative heating. While these codes represent the current state-of-the art, much uncertainty remains in understanding of large-scale entry vehicle environments for Mars. To account for uncertainties in aerothermal environments, margins were adopted using the following factors:

- 1.35/1.50 on convective heating for the fore and aft bodies, respectively
- 2.00 on radiative heating
- 1.10 on total heating rates for trajectory dispersions
- 1.35 on heating loads for trajectory dispersions

The margined, peak heating rates were computed as a function of time for the 1-sol orbit cases. The peak heating during aerocapture is 462 W/cm$^2$, while that for the out-of orbit phase is 132 W/cm$^2$. The out-of orbit peak heating profiles are the same for either mode of orbital insertion; i.e., aerocapture or propulsive. The long, in-orbit cool-down phase allows the TPS to return to a steady-state temperature of 294 K after the aerocapture phase-in preparation for the out-of-orbit entry

Figure 5-53 depicts the distributions of mission-maximum surface heating rates over the point design vehicle. The figure to the left shows the distribution during the aerocapture phase, while that to the right shows the distribution during the out-of-orbit (1-sol) entry. The heating distributions during the out-of-orbit entry phase are independent of the method of orbital insertion, either by aerocapture or by propulsive means.

[20]D. Kinney, J. Garcia, and L. Huynh, "Predictive Convective and Radiative Aerothermodynamic Environments for Various Reentry Vehicles using CBAero," AIAA paper 2006-659, Jan. 2006.
[21]Michael J. Wright, et al, "Computational Modeling of T5 Laminar and Turbulent Heating Data on Blunt Cones, Part 2: Mars Applications," AIAA paper 2005-0177, Jan., 2005.
[22]NEQAIR

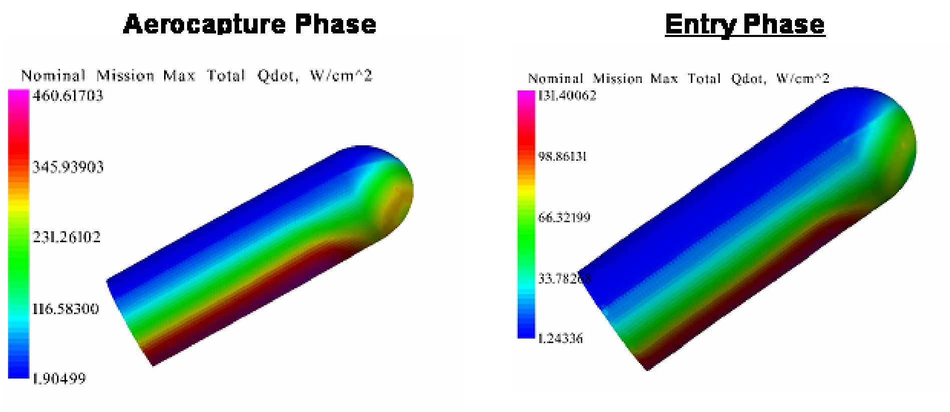



**Figure 5-53.** Distributed heating rates for the aerocapture and entry phases.

Using the margined, time-dependent aerothermal environments that are described above, the TPS sizing tools (Astronautix, 2007[23]), (Smith, et al., 2007[24]) that were employed by the ADP, which were appropriately modified for flight in the Mars atmosphere, were used to perform the TPS sizing estimates. The TPS materials that were selected for the aeroshell forebody heat shield were PICA and LI 2200. PICA is the baseline Orion/CEV ablator that is being developed for both the Block 1 (LEO) and Block 2 (lunar return) missions. PICA was the required TPS to account for the relatively high heating rates (462 W/cm$^2$) that were experienced during the aerocapture phase. For the leeward surfaces that will be exposed to less severe thermal environments, heritage shuttle TPS materials were selected, including LI-900 and FRSI blankets. The TPS mass sizing was performed for both a 20- and a 40-t lander payload case, and linear extrapolation was used to obtain aeroshell TPS values for the 70-t payload case. Table 5-20 provides a summary of the initial TPS masses estimates for both the 10-m × 30-m and the 12-m × 36-m aeroshell geometries for the aerocapture followed by entry from orbit trajectories for both a 500-km circular orbit and a 1-sol period elliptic orbit with a 33,793-km apoapsis.

**Table 5-20.** Initial Estimates of Thermal Protection System Masses for Various Configurations

| Ellipsled Geometry | Trajectory | 20-t Payload (kg) | 40-t Payload (kg) | 70-t Payload (kg) (extrapolated) |
|---|---|---|---|---|
| 10 × 30 m | 1 sol + Entry<br>500 km + Entry | 15,472<br>17,783 | 18,178<br>19,227 | 22,237<br>21,392 |
| 12 × 35 m | 1 sol + Entry<br>500 km + Entry | 20,748<br>21,860 | 22,909<br>24,857 | 26,150<br>29,351 |

The TPS materials that are discussed herein are at a reasonable TRL level. As will be discussed in subsequent sections, analysis of lower-TRL TPS materials and concepts could significantly reduce the TPS mass that is required for the human Mars missions, making the aerocapture, cool down, and subsequent out-of-orbit entry option even more attractive as compared to the all-propulsive orbital insertion alternative.

### 5.6.3 Descent stage

The descent stage engines were assumed from previous large lander studies to be RL10 derivatives and to have an engine T/W ratio of 40 $lb_f/lb_m$. Recognizing that the $LO_2/LH_2$ RL10 may not be the most appropriate analog for the $LO_2/CH_4$ engines that are currently baselined in this architecture, the parametric space was expanded to include engines that are derived from a RD-180 derivative that has a thrust-to-mass ratio of 80 $lb_f/lb_m$. The mass of the engines that were used in the thrust-to mass-ratio includes all associated turbopumps and all hardware that were attached to the engine before installation, but did not include the pressurant or tank-to-engine transfer line masses.

[23]Details of the RD-180 Pump Fed engine" www.astronautix.com/engines/rd180.htm.
[24]Michelle Smith,, et al., "Envision v1.11 Verification, Validation, & Accreditation", JSC-64040, November 2007.

The descent stage dry mass is based on mass characteristics that were modeled using the Envision mass sizing and simulation program. Point design data, which were obtained from Envision, were used to create response surface equations for ease of incorporation in the trajectory and sizing analysis. The descent stage is an all-propulsive, legged lander concept that uses four pump-fed $LO_2/CH_4$ engines with the following reference characteristics: an $I_{sp}$ of 369 seconds, an engine outfitting (O/F) of 3.5, a chamber pressure of 600 psi, and a nozzle area ratio of 200. The baseline vehicle was sized to conform to a 10-m inner-diameter aeroshell. The descent stage thrust structure was assumed to undergo maximum loading during the descent maneuver and is sized to withstand the user-defined system T/W without the aeroshell attached as payload, assuming that the aeroshell was deployed prior to terminal descent engine initiation. In addition, the tanks of the descent stage are sized to include the deorbit fuel. Additional margin was place on the terminal descent fuel budget to perform a “divert maneuver” following heatshield ejection, so that the heatshield debris does not impact the surface near any highly valued pre-deployed assets.

### 5.6.4 Trajectory calculations

Mathematical models for all masses were incorporated into the POST2 simulation, which is detailed in Striepe (2004). The POST2 optimization capability was used to converge the parameters that were used to generate the mass models with the optimal trajectory determined values.

For simplicity, in the first round of mass model analysis the entry trajectories were flown using a constant bank profile throughout entry. No constraints were set on the entry trajectory except for mass model convergence; the vehicle landed at an altitude of 0 km MOLA. Without further constraints, the entry trajectories from the 1-sol orbits lofted with a phugoid-like motion to altitudes in excess of 120 km to dissipate the entry energy, as shown in figure 5-54.

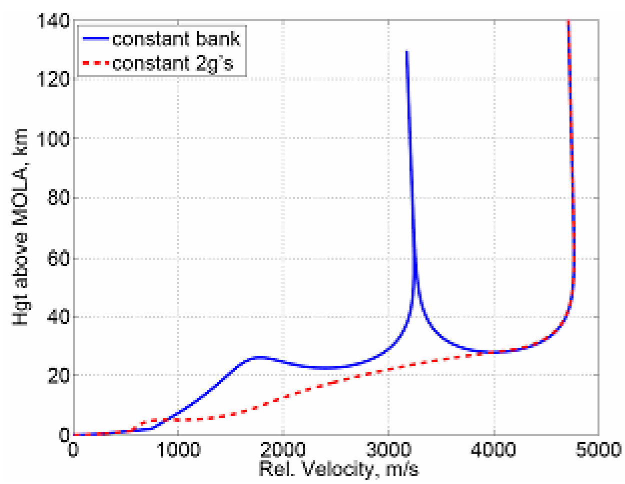



**Figure 5-54.** Entry trajectories for 10×30-m aeroshell 40-t payload from 1-sol orbit.

However this assumption was not realistic and was found to penalize both the TPS and the propulsion system design. To alleviate this phugoid entry trajectory behavior, a more realistic entry strategy was developed with the following elements: the entry flight path angle was determined such that the spacecraft would (1) pull out (i.e., the change in acceleration with time was zero) at the desired deceleration level; (2) maintain the pull-out deceleration level for a specified time; (3) fly full-lift up until engine initiation; (4) maintain a constant deceleration (3 g’s) until a velocity of 2.5 m/s was reached, and (5) maintain 2.5 m/s until touchdown (5 seconds). The more efficient trajectory, which is seen in the dashed line in figure 5-54, had the effect of reducing the entry mass by 4,800 kg or about 5% for the 40-t payload case.

Once this more-optimal trajectory-shaping strategy was defined, trajectories and mass sizing estimates were developed for the complete parametric design space that was described previously. The aeroshell parameter range included both the 10×30-m and 12×35-m aeroshell to evaluate the effect of packing density and ballistic number. The 10×30-m aeroshell was selected as a baseline because it corresponded to the reference Ares V launch shroud and required less TPS and structure mass; it also provided packing densities that were consistent with historical data for human missions, as described previously in section 3.5.3. Entries from the 500-km circular and 1-sol elliptic orbits were also considered. Entry from a 1-sol-period orbit was selected as the baseline for this study because it was considered the stressing case, but the actual orbit will depend on the overall Earth-Mars design reference mission architecture.

Results for the reference EDL configuration, which consisted of a 40-t payload in a 10×30-m aeroshell, and entry from a 1-sol period orbit are presented in table 5-19. The trajectory was designed such that the maximum deceleration for either the entry phase or the propulsive phase never exceeded 3 g’s. The propulsive phase was designed to lower the velocity to 2.5 m/s at an altitude of 12 m above the surface, which was then maintained until touchdown. A 30% delta–V margin was allocated for precision landing and terminal descent hazard avoidance. Mass results for both the baseline AC and the propulsive Mars orbit capture cases are also shown in table 5-21. Note that two values for the arrival mass are provided for the propulsive capture (one assuming a chemical propulsion system with an efficient $I_{sp}$ and the second assuming an NTP system with a more efficient $I_{sp}$. In either case, the AC option provides a significant mass savings over traditional propulsive capture options, and would result in lower initial mass in Earth orbit, which is a significant indicator of overall architecture cost.

**Table 5-21.** Baseline 40-t Landed Payload Design Mass and Trajectory Results

| **Baseline 1-sol orbit, 10 m × 30 m Aeroshell, 40 MT Payload, 2g entry, $T/W_{sys}$ = 3g’s, $T/W_{eng}$ = 80 $lb_f/lb_m$, 30% delta-V margin.** | | | |
|---|---|---|---|
| **Capture to 1-sol orbit** | | **Aerocapture** | **Propulsive Capture** |
| **Arrival Mass** | kg | **115,549** | **TBD** |
| Post-AC Orbit Insert Prop | kg | 4692 | N/A |
| **Orbit Mass** | kg | **110,857** | **106,842** |
| Deorbit Propellant | kg | 458 | 441 |
| **Entry Mass** | kg | **110,399** | **106,401** |
| Aeroshell Structure | kg | 22,500 | 22,500 |
| TPS | kg | 14,600 | 11,100 |
| RCS Dry Mass | kg | 994 | 958 |
| RCS Prop | kg | 1,218 | 1,174 |
| Terminal Descent Prop | kg | 13,887 | 13,545 |
| Pinpoint Landing Prop | kg | 3,030 | 2,964 |
| **Landed Mass** | kg | **57,200** | **57,124** |
| Dry Descent Stage | kg | 17,200 | 17,124 |
| **Payload Mass** | kg | **40,000** | **40,000** |
| Deorbit DV | m/s | 15 | 15 |
| Ballistic coefficient | $kg/m^2$ | 475 | 458 |
| Descent DV | m/s | 600 | 587 |
| Max heat rate | $W/cm^2$ | 462 | 132 |
| Total heat load | $MJ/cm^2$ | 597 | 172 |
| Altitude engine initiation | m | 1,455 | 1,353 |
| Mach at engine initiation | M | 2.30 | 2.25 |
| Time of flight | sec | 494 | 487 |
| Time at constant g’s | sec | 132 | 134 |

Additional trajectory shaping and optimization efforts were made to determine whether more-efficient trajectories and, thus, lower mass entry systems could be found. Various trajectory pull-out g-load levels were considered in the EDL parameter trade space. The trajectories for each payload size were designed to hold a 1g, 2g, and 3g constant profile upon entry. As previously described, a significant entry mass savings was realized using constant-g profile over the constant bank angle profile; additional mass savings was seen with increased g-loading, such that a mass optimized trajectory that was flown to hold a constant 3g's during entry would have a lower entry mass than a trajectory that was optimized to hold 1g. However, the mass savings was not found to be linear with g-load and payload, and it dramatically increased from 20- to 70-t payloads (i.e., going from a 1g to a 2g trajectory for the 20-t case only resulted in an approximate 400-kg mass reduction where a 70-t payload case of a similar trajectory type yielded an entry mass reduction of close to 10,000 kg). There was not a significant difference between the entry mass that was saved going from the 2g to the 3g trajectory, thus the 2g case was baselined for the study. The 2g case provides more timeline margin over the 3g case, and would also be more tolerant to atmospheric dust loading and seasonal variation in atmospheric density profiles. These results are shown in table 5-22, with details on system and component masses for 20-, 40-, and 70-t payloads (here assuming no terminal descent delta-V margins). The 70-t entry mass had a ballistic coefficient that was too large to maintain a 3g entry profile and successfully land (i.e., the vehicle impacted the surface prior to decelerating to appropriate engine initiation conditions).

**Table 5-22.** EDL system sensitivities to Payload Mass and Trajectory Loading

| 24 hour orbit, 10x30 m, 40 mt payload | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|
| Payload Mass ➔ | | 20 mt | | | 40 mt | | | 70 mt | | |
| T/W eng = 80 lbf/lbm, T/Wsys = 3 g's | | 1 g | 2 g's | 3 g's | 1 g | 2 g's | 3 g's | 1 g | 2 g's | 3 g's |
| Orbit Mass: | kg | **74870** | **74467** | **74062** | **111299** | **109800** | **109257** | **173532** | **165393** | *Did not converge* |
| Deorbit Propellant: | kg | 271 | 300 | 331 | 411 | 454 | 506 | 653 | 698 | |
| Entry Mass: | kg | 74599 | 74168 | 73731 | 110888 | 109347 | 108752 | 172878 | 164695 | |
| Aeroshell Structure: | kg | 22500 | 22500 | 22500 | 22500 | 22500 | 22500 | 22500 | 22500 | |
| TPS: | kg | 15472 | 15472 | 15472 | 18178 | 18178 | 18178 | 22237 | 22237 | |
| RCS Dry Mass: | kg | 671 | 668 | 664 | 998 | 985 | 980 | 1556 | 1483 | |
| RCS Prop: | kg | 823 | 818 | 814 | 1223 | 1206 | 1200 | 1907 | 1817 | |
| Terminal Descent Prop: | kg | 4794 | 4448 | 4098 | 11341 | 10086 | 9595 | 28232 | 21504 | |
| Landed Mass: | kg | 30339 | 30261 | 30183 | 56648 | 56391 | 56298 | 96446 | 95154 | |
| Dry Descent Stage: | kg | 10339 | 10261 | 10183 | 16648 | 16391 | 16298 | 26446 | 25154 | |
| Payload Mass: | kg | 20000 | 20000 | 20000 | 40000 | 40000 | 40000 | 70000 | 70000 | |
| Deorbit ΔV | m/s | 13 | 15 | 16 | 13 | 15 | 17 | 14 | 15 | |
| Ballistic coefficient | kg/m2 | 321 | 319 | 317 | 477 | 471 | 468 | 744 | 708 | |
| Descent ΔV | m/s | 531 | 496 | 461 | 660 | 595 | 569 | 929 | 737 | |
| max stag heat rate | W/cm2 | 11 | 14 | 17 | 13.49 | 17.72 | 20.64 | 17 | 22 | |
| Total stag heat load | J/cm2 | 2562 | 1937 | 1646 | 3123 | 2351 | 1999 | 4309 | 2896 | |
| altitude engine initiation | m | 1333 | 966 | 1028 | 1744 | 1350 | 1501 | 2300 | 1431 | |
| Mach @ engine initiation | M | 2.03 | 1.89 | 1.74 | 2.56 | 2.29 | 2.18 | 3.71 | 2.88 | |
| time of flight | sec | 698 | 468 | 407 | 695 | 486 | 428 | 917 | 476 | |
| time at constant g's | sec | 335 | 150 | 85 | 306 | 134 | 73 | 134 | 124 | |

T/W of the engines was also considered as a parameter in the design space when it was recognized that previous large lander studies had assumed RL10 derivatives with an engine T/W ratio of 40 $lb_f/lb_m$. Acknowledging that a human mission Mars would likely benefit from and ultimately require more efficient engines, the parametric analysis also considered RD-180 derivatives engine with a T/W ratio of 80 $lb_f/lb_m$. The results of the parametric study indicate an approximate 6% entry mass savings for the 40-t entry mass using the larger T/W. The sensitivity of the baseline 40-t payload system to the engine T/W ratio is shown in table 5-23 below.

**Table 5-23.** Entry Mass Sensitivity to Engine Thrust to Weight

| From 24-hour orbit, 10×30-m, 40-t PL, 2g entry, T/W$_{sys}$ = 3g | | | |
|---|---|---|---|
| **T/W of the engines (lb$_f$/lb$_m$)** | | **40** | **80** |
| **Orbit Mass** | kg | **116,311** | **109,800** |
| Deorbit Propellant | kg | 481 | 454 |
| **Entry Mass** | kg | **115,830** | **109,347** |
| Aeroshell Structure | kg | 22,500 | 22,500 |
| TPS | kg | 18,178 | 18,178 |
| RCS Dry Mass | kg | 1,043 | 985 |
| RCS Prop | kg | 1,278 | 1,206 |
| Terminal Descent Prop | kg | 11,357 | 10,086 |
| **Landed Mass** | kg | **61,475** | **56,391** |
| Dry Descent Stage | kg | 21,475 | 16,391 |
| **Payload Mass** | kg | **40,000** | **40,000** |
| Deorbit DV | m/s | 15 | 15 |
| Ballistic coefficient | kg/m$^2$ | 498 | 471 |
| Descent DV | m/s | 613 | 595 |
| Max heat rate | W/cm$^2$ | 18 | 18 |
| Total heat load | J/cm$^2$ | 2,432 | 2,351 |
| Altitude engine initiation | m | 1,410 | 1,350 |
| Mach at engine initiation | M | 2.36 | 2.29 |
| Time of flight | sec | 491 | 486 |
| Time at constant g's | sec | 134 | 134 |

Sensitivities to terminal descent propellant delta-V margins were also considered in the parametric trade space, recognizing that additional terminal descent propellant for precision landing will likely be required to ensure that all jettisoned mass (e.g., aeroshell) impacts the surface at a safe distance from any pre-deployed landed infrastructure. A study was performed on the baseline case that considered margins of 20%, 50%, and 100% of the nominal delta-V that was  required to land. The results are presented in table 5-24. A 30% delta-V margin was selected as the reference baseline.

**Table 5-24.** Mass Sensitivity to Terminal Descent Propellant Delta-V Margin

| From 24-hour orbit, 10×30-m, 40-t PL, 2g entry, $T/W_{sys}$ = 3g, $T/W_{eng}$ = 80 | | | | | |
|---|---|---|---|---|---|
| **DV margin** | | **0%** | **20%** | **50%** | **100%** |
| **Orbit Mass** | kg | **109,800** | **113,096** | **118,689** | **130,403** |
| Deorbit Propellant | kg | 454 | 467 | 490 | 539 |
| **Entry Mass** | kg | **109,347** | **112,629** | **118,199** | **129,865** |
| Aeroshell Structure | kg | 22,500 | 22,500 | 22,500 | 22,500 |
| TPS | kg | 18,178 | 18,178 | 18,178 | 18,178 |
| RCS Dry Mass | kg | 985 | 1,014 | 1,064 | 1,170 |
| RCS Prop | kg | 1,206 | 1,243 | 1,304 | 1,433 |
| Terminal Descent Prop | kg | 10,086 | 12,737 | 17,230 | 26,628 |
| Pinpoint Landing Prop | kg | | 2,018 | 5,386 | 12,340 |
| **Landed Mass** | kg | **56,391** | **56,957** | **57,922** | **59,956** |
| Dry Descent Stage | kg | 16,391 | 16,957 | 17,922 | 19,956 |
| **Payload Mass** | kg | **40,000** | **40,000** | **40,000** | **40,000** |
| Deorbit DV | m/s | 15 | 15 | 15 | 15 |
| Ballistic coefficient | $kg/m^2$ | 471 | 484 | 508 | 559 |
| Descent DV | m/s | 595 | 605 | 620 | 654 |
| Max heat rate | $W/cm^2$ | 18 | 18 | 18 | 19 |
| Total heat load | $J/cm^2$ | 2,351 | 2,391 | 2,458 | 2,595 |
| Altitude engine initiation | m | 1,350 | 1,344 | 1,332 | 1,335 |
| Mach at engine initiation | M | 2.29 | 2.33 | 2.39 | 2.53 |
| Time of flight | sec | 486 | 486 | 485 | 483 |
| Time at constant g's | sec | 134 | 134 | 134 | 134 |

### 5.6.5 Conclusions and recommendations

The reference mission architecture parametric models were developed based on these parametric trades and sensitivity studies. Linear scaling laws were used to resize systems and masses, where appropriate, to provide overall architecture-level mass sensitivities for the various architecture trade options

Many technologies that could provide significant mass and/or EDL timeline savings were not considered for this study because of a lack of technology maturity. It is recommended that the following technologies be matured to the point that more-detailed EDL trade studies and technology assessments can be performed. These areas include:

1. *Aeroshell shape*. The ellipsled was chosen for this study as it was the most mature design available. Other shapes (e.g., biconic) may offer advantages in mass, controllability, packaging, etc.
2. *Aeroshell TPS*. This was a major mass driver for this study. Dual-use ablative TPS has not been demonstrated in flight.
3. *Inflatables for aerocapture and entry*. These would reduce the ballistic coefficient, allow for higher minimum aerocapture altitude, etc.
4. *Supersonic inflatable decelerators*. These would reduce the DV requirement from the engines and increase the descent timeline.
5. *Descent engines*. Data are showing that the proper design of the propulsive system could increase the base drag and, thus, effectively increase the $I_{sp}$, thereby reducing mass.

Additional discussion on technology roadmaps and options may be found in section 7 of this report.

## *5.7 Mars Transit Habitat*

The MTV consists of the TMI and TEI propulsion stages (whether nuclear or chemical), the CEV that serves the function of an ERV (for the final leg of the journey home), and a TransHab in which the crew lives for the round-trip between Earth and Mars. Whether the TransHab is constructed using rigid-body or inflatable technology will need to be determined by detailed engineering analysis, however, it is assumed that it will share as many systems as pragmatically possible with the Mars SHAB. The rationale behind maximizing the commonality between these two elements (one that operates in a zero-g environment and the other in a 1/3-g environment) is driven by the desire to lower development costs as well as to reduce the number of systems that astronauts would have to learn to operate and repair. An even more critical assumption is that the systems that will be comprising the TransHab (and SHAB) would be largely based on the hardware design and reliability experience that are gained by ISS operations, as well as long-duration surface habitat operations on the lunar surface (i.e., Lunar Outpost)which that would precede any Mars campaign.

For the analysis work that was completed in DRA 5.0, a parametric sizing tool was used that was developed at JSC. The mass estimates for the TransHab are essentially equal to the estimates that were produced in DRM 4.0, with a few minor changes in assumptions regarding consumables and spares for needed maintenance of the habitat. A summary of these estimates is included in table 5-25.

**Table 5-25.** Parametric Sizing of the Mars Transit Habitat

| Transit Habitat Mass Estimate | Mass (kg) | Stowed Vol. (M3) |
|---|---|---|
| **1.0 Power System** | **5,840** | **-** |
| Space PV Arrays | 1,050 | - |
| Regegerative Fuel Cells | 3,472 | - |
| Batteries | 124 | - |
| Power Management and Distribution | 1,189 | - |
| Margin | - | - |
| **2.0 Avionics** | **290** | **0.1** |
| Wireless Instrumentation | 80 | 0.0 |
| Command & Data Handling | 20 | 0.0 |
| Proximity Operations Communications | 72 | 0.0 |
| Guidance, Navigation, & Control | 45 | 0.0 |
| Displays & Controls | 70 | 0.1 |
| Margin | - | - |
| **3.0 Environmental Control & Life Support** | **3,950** | **19.1** |
| Air Revitalization | 817 | 7.3 |
| Water Reclamation | 423 | 4.0 |
| Waste Management | 436 | 5.8 |
| Food Production | 120 | 1.9 |
| Integrated System Management | 20 | - |
| Fluids | 1,883 | - |
| Spares | 250 | - |
| Margin | - | - |
| **4.0 Thermal Management System** | **1,260** | **5.3** |
| Heat Exchangers | 40 | 0.1 |
| Cold Plates | 150 | 0.4 |
| Pumps | 120 | 0.9 |
| Plumbing and valves | 260 | - |
| Instruments and controls | 87 | - |
| Fluids | 43 | - |
| Radiators | 557 | 3.9 |
| Margin | - | - |
| **5.0 Crew Accommodations** | **4,210** | **29.7** |
| Galley | 186 | 0.1 |
| Wardroom | 30 | 0.1 |
| Personal Hygeine | 32 | 0.1 |
| Clothing | 273 | 1.3 |
| Personal Stowage | 150 | 4.5 |
| Housekeeping | 17 | 0.3 |
| Operational Supplies | 220 | 0.1 |
| Maintenance | 1,092 | 5.9 |
| Photography | - | - |
| Sleep Accommodations | 54 | 0.6 |
| Crew Health | 759 | 3.7 |
| Margin | - | - |
| Other Consumables | 1,392 | 12.9 |
| **6.0 EVA Systems** | **870** | **2.9** |
| Primary System | 473 | 1.7 |
| Spares | 119 | 0.9 |
| Consumables | 276 | 0.4 |
| Margin | - | - |
| **7.0 Structure** | **2,020** | **-** |
| Core Structure | 1,218 | - |
| Inflatable Structure | - | - |
| Docking Mechanism | 800 | - |
| Margin | - | - |
| Margin (30%) | 4,920 | 8.6 |
| Additional Spares | 4,180 | 1.4 |
| Crew | 560 | - |
| **Total Transit Habitat Mass (without food)** | **28,100** | **65.8** |
| | | |
| Food (Return Trip) | 2,650 | 7.9 |
| Food (Outbound Trip) | 2,650 | 7.9 |
| Food (Contingency) | 7,940 | 23.5 |
| **Total Consumable Mass** | **13,240** | **39.4** |
| | | |
| **Total TransHab Mass @ TMI** | **41,340** | |

The food that will be carried aboard the TransHab includes food that is needed for the round-trip journey as well as food that may be needed in the event that all or part of the surface mission is aborted and the crew is forced to loiter in Mars orbit aboard the TransHab until the opening of the TEI window. Any contingency food remaining would be jettisoned prior to the TEI burn that will lead to a return home. The food that was included aboard the TransHab and that was used as a reference for DRA 5.0 is shown in table 5-23.

To calculate the complete mass of the MTV, the two totals in table 5-23 were added to 10,000 kg for the ERV capsule (based on the current Orion Block 1 command module mass), and an additional 5,500 kg were added to account for maintenance spares and any further propulsion module that may be necessary for Earth targeting of the ERV. This brings the total mass of the MTV to approximately 51.3 t. This mass was used to size the nuclear and chemical propulsion systems.

## *5.8 Mars Ascent/Descent Vehicle*

The MAV that was used for the DRA 5.0 reference studies nominally transports a crew of six between the surface of Mars and the MTV, which has been loitering in Mars orbit for the duration of the surface mission. Given the assumption that ISRU technologies are used for ascent oxidizer production, the MAV is pre-deployed to the surface of Mars during the opportunity prior to the crew’s departure. It is only after the MAV is verified as fully fueled that the crew is committed on its journey via the MTV. If ISRU technology is unavailable for use by the time of the human mission to Mars, a DAV that was fully fueled (heavier) would be pre-deployed to Mars orbit instead of to the surface, and the crew would both descend from and ascend to the MTV. A parametric tool was used for the DRA 5.0 design activities to size the lander, which was very similar (at a subsystem level) to the lander that was used in the *Dual Lander Study (1999)*. This vehicle however, consisted of a two-stage ascent vehicle design with one $LO_2/CH_4$ pump-fed engine on the second stage and four $LO_2/CH_4$ pump-fed engines on the first and descent stages. The $CH_4$ fuel is brought from Earth, but the $O_2$ is created using ISRU technology. This allows for a significantly lighter landed mass, which propagates back through the architecture to result in substantially reduced masses in LEO. Table 5-26 below summarizes the parametric vehicle sizing.

**Table 5-26.** Summary of Mars Ascent/Descent Vehicle Sizing

| | | |
|---|---|---|
| Ascent Stage 2 | 9,330 kg | Includes $CH_4$ brought from Earth |
| Ascent Stage 1 | 12,156 kg | Includes $CH_4$ brought from Earth |
| ISRU Equipment and Power Plant | 11,280 kg | |
| Descent Stage | 21,297 kg | |
| **Total** | **54,062 kg** | |

The ISRU equipment and power plant that are listed in the mass summary are further described in sections 6.2 and 6.3, respectively. The engine characteristics of the $LO_2/CH_4$ pump-fed engines that were used on the vehicle are as follows:

| | |
|---|---|
| 30,000 $lb_f$ | Maximum Thrust per Engine (1 N = 0.2248 $lb_f$) |
| 900 psi | Engine Chamber Pressure |
| 2.5 | Engine Chamber Contraction Ratio |
| 2.5 | Engine Chamber L/D |
| 200 | Nozzle Area Ratio |

## *5.9 Landing Plume Effects*

When a 40-t spacecraft lands on the regolith of Mars, the blast effects will be significantly greater than the experience of Apollo or Viking due to the deep fluidization and cratering of the soil that will occur. There are three major concerns: (1) regolith material will be ejected into high trajectories and may damage surrounding hardware far away; (2) rocks will strike the landing spacecraft at high velocity, and (3) the fluidized soil will collapse into a broad crater after the engines are cut off, possibly upsetting the stability of the landed spacecraft. These effects can be mitigated with reasonable effort, but it will require a keep-out blast zone of about 1-km radius around the landing site along with an increase of about 2 t in lander mass to account for shielding and wider landing gear. The alternative method to mitigate the plume is to robotically clear and stabilize the landing site prior to the arrival of the lander, thus preserving the lower lander mass (i.e., enabling a greater landed payload mass by about 2 t per lander) and reducing the size of the keep-out zone around the lander.

### 5.9.1 Summary and recommendations

The predictions and recommendations for a 40-t spacecraft on Mars are described in summary in this section. The next section of the report will then explain in detail how these predictions were obtained.

The engine exhaust plume from a 40-t lander on Mars will blow dust, sand, gravel, and even rocks up to about 7 cm in diameter at high velocity. These ejecta will cause significant damage to any hardware that is already placed on the martian surface within the blast radius. However, the blast radius is modest, extending out to approximately 1 km. The largest debris is accelerated by the plume to lower velocities and, thus, falls closer to the landing site; and the smallest particles are attenuated by the martian atmosphere, also falling closer to the landing. Thus, maintaining the distance of about 1 km between the landing site and any existing surface assets will completely solve this problem for all sizes of debris.

A second concern arises because the exhaust from the large engines will form deep, narrow craters that are directly beneath each of their nozzles, and these craters will redirect the supersonic jet of gas with sand and rocks up toward the landing spacecraft. This has been demonstrated in large-scale engine tests in sand and clay (Alexander, et al, 1966) [25], small-scale experiments (Metzger, 2007) [26], numerical simulations (Liever, et al, 2007) [27], and soil dynamics analysis (see section 5.10.2.3), so there is no question that this will occur. It did not occur in the Apollo and Viking missions because the thrust was lower and/or because the lunar regolith had higher shear strength and less permeability than martian soil. These variables have been taken into consideration in this report. An example of a small-scale test is provided in figure 5-55. The impact of debris striking the lander will be sufficient to cause damage to the lander, possibly resulting in LOM and LOC, and therefore must be prevented. Of special concern is damage to the engine nozzles, because with a multiple-engine lander the debris that is ejected by one engine will be aimed directly at the other engines. One mitigation approach is to add shielding to the spacecraft to block the debris. This will increase the mass of the lander and, therefore, reduce the mass of the payload by approximately 1 t.

[25] Alexander, J. D., W. M. Roberds, and R. F. Scott. *Soil Erosion by Landing Rockets*. NASA Contract NAS9-4825 Final Report. (Hayes International Corp., Birmingham, Alabama, 1966.)
[26] Metzger, Philip T., et al., “Jet-induced cratering of a granular surface with application to lunar spaceports,” *J. Aerospace Engineering* (accepted for publication, 2007).
[27] Liever, Peter, et al.. *Numerical Simulation of Rocket Exhaust Interaction With Lunar Soil.* NASA Contract NNK07MA36C Bi-Monthly Report (Kennedy Space Center, March 22, 2007).

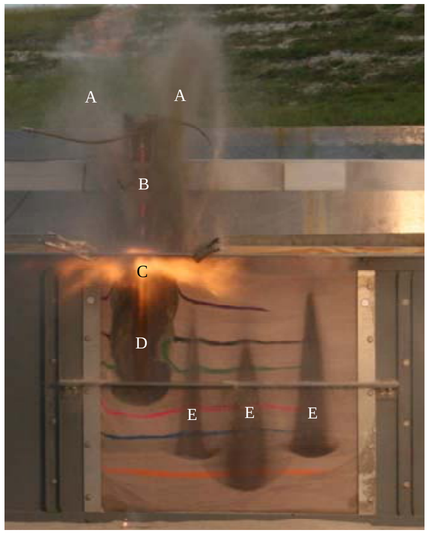



**Figure 5-55.** Plume exhaust test.

A third concern is the stability of the soil after the engines shut off. Experiments show that as the engines cut off, the resulting hole collapses into a broad, shallow crater as shown in figure 5-56.

**Figure 5-56.** Residual crater after solid motor firing is complete.

Current estimates show that the width of the collapsed zone in martian soil will be at least as wide as the lander itself (~10 m). The equations tend to under-predict the cratering effects and so the crater may actually be wider; we are unable to make more accurate predictions at the present. In any case, there is a real danger that the lander could be unstable after engine shutoff. One way to mitigate this problem is to increase the width of the landing gear to extend beyond the zone of unstable soil. This wider landing gear will increase the lander mass and, therefore, decrease the mass of the payload by another estimated 1 t, as estimated in section 5.10.10, in addition to the 1 t of shielding mass.

An alternative method to mitigate these plume effects is to prepare a landing site robotically (by the dual-use of an ISRU excavator for $H_2O$ extraction from martian soil or site/resource surveying rover, for example) so that no soil or rocks will be blown by the plume. This could be done by sending the rover that has site-preparation capabilities one flight opportunity prior to sending an unfueled ascent lander, which would be two flight opportunities prior to sending humans. A landing site would then be prepared for each of the next two landings by moving loose surface material followed by compaction and microwave sintering of the soil to create an in-situ landing pad. A Mars architecture that includes robotic site preparation would thus allow the lander mass to be reduced and the payload mass to be increased by about 2 t per landing (a rough order of magnitude), all other things being equal. Landing site preparation of this sort would also require the capability for precision landing to make use of the prepared site.

Because of the significant risk of damage to the landing spacecraft by debris impacts, and the added risk of lander instability from the deep crater and collapse zone that will form beneath it, the following steps are recommended to be completed prior to sending humans to Mars:

- The numerical models that are currently being developed at low-funding levels should be accelerated and advanced to higher fidelity within the next 2 to 3 years. This will enable better estimations of the plume effects both on the moon and on Mars.
- Concepts should be developed for the preparation of landing sites and the mitigation of blast effects on Mars. These should try, as far as possible, to make dual use of site survey and/or ISRU hardware such as excavators. These concepts should be developed early in the buildup to the lunar program so that prototypes can be field-tested using excavators, lunar landers, and other assets on the surface of the moon.
- Additional small-scale engine hot-firings should be performed into soil beds to improve our understanding of the soil/gas dynamics and to calibrate the numerical models.
- A robotic lander on a precursor mission should be instrumented to video capture and analyze the plume effects on Mars.
- Full-scale engine hot-firing tests should eventually be performed in terrestrial soils to identify any unexpected flow regime changes that may exist at larger scales.

### 5.9.2 Detailed analysis

#### *5.9.2.1 Background review*

The list below provides potential concerns for retrorocket plumes interacting with the martian regolith. This study is focused only on the first three items on that list, as these are the items most likely to affect mission architecture.

1. Ejecta hits surrounding hardware
2. Ejecta hits landing spacecraft
3. Regolith instability after landing
4. Loss of visibility during landing
5. Spoofing of landing sensors
6. Dust deposition on hardware
7. Crew exposed to plume chemicals in soil

NASA has extensive experience predicting and controlling the blast effects in launch and landing environments, both terrestrially (Schmalzer, et al, 1998 [28]) and in the prior lunar and martian programs (Roberts, 1963 [29], Land, et al, 1966[30], Christensen, et al, 1967[31], Hutton, 1968[32], Mason, 1970[33], O'Brien, et al, 1970[34], Cour-Palais, 1972[35], Jaffe, 1972[36]). The numerical tools that are needed to predict the blast in planetary regoliths are at a low state of

[28] Schmalzer, Paul A., et al.. *Monitoring Direct Effects of Delta, Atlas, and Titan Launches from Cape Canaveral Air Station*. NASA TM-207912 (1998).
[29] Roberts, Leonard, "The Action of a Hypersonic Jet on a Dust Layer," IAS Paper No. 63-50 (New York: Institute of Aerospace Sciences 31st Annual Meeting, 1963).
[30] Land, Norman S., and Harland F. Scholl. *Scaled Lunar Module Jet Erosion Experiments.* NASA Technical Note D-5051 (Hampton, Va.: Langley Research Center, 1966).
[31] Christensen, E. M., et al. "Surveyor V: Lunar Surface Mechanical Properties," *Science* **158**, pp. 637-40 (1967).
[32] Hutton, Robert E. *Comparison of Soil Erosion Theory with Scaled LM Jet Erosion Tests.* NASA-CR-66704 (1968).
[33] Mason, Curtis C., "Comparison of Actual versus Predicted Lunar Surface Erosion Caused by Apollo 11 Descent Engine," *Geological Society of America Bulletin* **81**, 1807-12 (1970).
[34] O'Brien, B.J., S.C. Freden, and J.R. Bates, "Degradation of Apollo 11 Deployed Instruments because of Lunar Module Ascent Effects," *J. Applied Physics* **41**(11), 4538-4541, (October, 1970).
[35] Cour-Palais, B. G., "Part E. Results of Examination of the Returned Surveyor 3 Samples for Particulate Impacts," *Analysis of Surveyor 3 material and photographs returned by Apollo 12*, NASA SP-284, (Washington D. C.: NASA, 1972), pp 154-67.
[36] Jaffe, L. D., "Part I. Blowing of Lunar Soil by Apollo 12: Surveyor 3 Evidence," *Analysis of Surveyor 3 material and photographs returned by Apollo 12*, NASA SP-284, (Washington D. C.: NASA, 1972), pp 94-6.

fidelity, however, since modeling capabilities have been developed in an era when planetary landings with retrorockets have not occurred. The tools are being developed now, but in this study it was necessary to rely on rudimentary approximations for some aspects of the blast effects.

The interaction with exhaust plumes with loose regolith material was first studied in the era leading up to the Apollo and Viking programs (Foreman, 1967[37], Hutton, 1968[38], Clark, 1970[39], Ko, 1971[40], Romine, 1973[41], Hutton, et al, 1980[42]) and again more recently (Phillips, et al, 1992[43], Katzan, et al, 1991[44], Metzger, 2006[45], Metzger, 2007[46]). Also, large particulates (rocks and boulders) are sometimes liberated and blown from refractory concrete of terrestrial launch pads. This effect produces significant damage to the surrounding hardware (fences, lighting fixtures) with each space shuttle launch. The interaction of an exhaust plume with a planetary regolith will vary dramatically from one environment to the next, depending upon the characteristics of the plume, the ambient atmosphere, the mechanical properties of the regolith, and gravity. Research has identified five primary phenomena that may occur in varying degrees. These are: (1) viscous erosion  (2) diffused gas eruption (Scott, et al, 1968[47]) , (3) bearing capacity failure, (4) diffusion-driven shearing, and (5) mechanical shock. The specific conditions of the planet and plume will govern which of these phenomena is/are predominant; it therefore would be a mistake to directly compare the blast effects on the moon or Earth with those that will occur on Mars. The comparison must take into account the differences in the physics.

For lunar landings, viscous erosion is the primary phenomenon wherein a high velocity flow of dust, sand, and possibly small gravel moves beneath the standoff shockwave of the plume in a nearly horizontal direction. The lifting of material appears to be attributable to aerodynamic forces on the grains without much saltation due to the limited radial extent of the dense plume. In Apollo 12, this flow of material produced significant surface erosion and pitting on the Surveyor III spacecraft that was about 180 m away. The high-velocity fines scoured a thin layer of material from all exposed surfaces. The larger particles peppered the surface with micro-craters (on the order of 30 μ in diameter). Soil was blown into the crevices of the Surveyor, which would have threatened mechanical jamming if the hardware had been functional. If gravel-sized particles that were blown by the plume had randomly struck the Surveyor, more significant damage would have occurred. The entrained dust particles obscured the view of the surface during landing, particularly on Apollo missions 12 and 15.

The more violent cratering phenomena do not occur easily in the lunar environment because of the high mechanical strength of the unique lunar soil (Carrier, et al, 1991[48]) as well as its low permeability resisting gas diffusion, and because the exhaust plumes expand widely in the lunar vacuum so that the stagnation pressure is not focused on a narrow patch of soil. Diffused gas eruption may have occurred at least one time during the Surveyor V mission, however, when the vernier engines were fired as a test after landing. There were some large blasts of soil in the final moments of some landings (particularly Apollo 15) that may have been due to enhanced viscous scouring at low

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[37] Foreman, K. M., "The Interaction of a Retro-Rocket Exhaust Plume with the Martian Environment," Grumman Res. Dept. Memorandum RM-354 (Bethpage, NY: Grumman Aircraft Engineering Corp., 1967).
[38] Hutton, Robert E. *Mars Surface Soil Study.* NASA Contract AX 422060 Report (Pasadena, Calif.: Jet Propulsion Lab., 1968).
[39] Clark, Leonard V. *Effect of Retrorocket Cant Angle on Ground Erosion – a Scaled Viking Study.* NASA TM X-2075 (Hampton, Va.: Langley Research Center, October 1970).
[40] Ko, Hon-Yim. *Soil Properties Study.* Viking Project Report VER-181 (Martin Marietta Corp, Oct. 15, 1971), compiles work under subcontract including "Effects of Pore Pressure on Cratering Due to Rocket Exhaust-Gas Impingement" (October, 1970).
[41] Romine, G. L., T. D. Reisert, and J. Gliozzi. *Site Alteration Effects from Rocket Exhaust Impingement During a Simulated Viking Mars Landing. Part I – Nozzle Development and Physical Site Alteration.* NASA CR-2252 (Denver Co.: Martin Marietta Corporation, July 1973).
[42] Hutton, R. E., H. J. Moore, R. F. Scott, R. W. Shorthill, and C. R. Spitzer, "Surface Erosion Caused on Mars from Viking Descent Engine Plume," *The Moon and the Planets* **23**, 293-305 (1980).
[43] Phillips, Paul. G., Charles H. Simonds, and William R. Stump, "Lunar Base Launch and Landing Facility Conceptual Design," *Proc. of the Second Conference on Lunar Bases and Space Activities of the 21st Century*, v.1 (NASA Johnson Space Center), Sept. 1, 1992, pp. 139-151.
[44] Katzan, Cynthia M. and Jonathan L. Edwards. *Lunar Dust Transport and Potential Interactions With Power System Components.* NASA Contractor Report 4404 (Sverdrup Technology, Inc., Brook Park, OH, 1991), pp. 8-22.
[45] Metzger, Philip T., et al., "Prediction and Mitigation of Plume-Induced Spray of Soil during Lunar Landings," NASA Lunar Architecture Team report (August 2006).
[46] Metzger, Philip T., et al., "Cratering of Soil by Impinging Jets of Gas, with Application to Landing Rockets on Planetary Surfaces," *Proceedings of the 18th Engineering Mechanics Division Conference* (Blacksburg, Virginia), June 3-6, 2007.
[47] Scott, Ronald F., and Hon-Yim Ko, "Transient Rocket-Engine Gas Flow in Soil," *AIAA Journal* **6**(2), 258-64 (1968).
[48] Carrier, W. David, III, Gary R. Olhoeft and Wendell Mendell, "Physical Properties of the Lunar Surface," in *Lunar Sourcebook, A User's Guide to the Moon*, G. H. Heiken, D.T. Vaniman and B.M. French, eds., (Cambridge University Press, Melbourne, Australia, 1991), pp. 475 – 594.

altitude or the removal of soil that was mechanically fractured by the contact probes of the lander. A crater of approximately 444 L volume was noted by the Apollo 14 crew.

For terrestrial launches on prepared concrete pads, mechanical shock may be the primary mechanism that fractures and liberates surface material, which is then blown away by the plume. This is primarily a launch effect because, in landings, the standoff shock is able to form more gradually as the spacecraft descends. Terrestrial experiments in loose soil beds have demonstrated cases in which diffused gas eruption, bearing capacity failure, and/or diffusion-driven shearing are the predominant phenomena. These occur more readily when there is an ambient atmosphere to focus the plume and create a high stagnation pressure with steep pressure gradients around the edges.

For martian landings during the Viking program, it was determined that bearing-capacity failure would occur and produce scientifically unacceptable disturbance in the soil sampling area. The spacecraft engines were re-designed so that the nozzle of each was replaced by a “showerhead” design with 18 tiny nozzles. This enhanced the turbulent mixing of the exhaust jets with the ambient atmosphere so that the jets were extincted at a shorter range. Nonetheless, shallow craters were confirmed to occur under the engines within view of the cameras; at the Viking Lander 2 site, there was a cluster of small craters corresponding to each of the individual nozzles. NASA is currently conducting a series of additional tests and simulations to better understand the conditions for onset of deep cratering and to scale the magnitude of these effects.

#### *5.9.2.2 Martian conditions*

On Mars, the atmospheric drag force and gravity are both intermediate to the lunar and terrestrial cases. The martian atmosphere is sufficiently rarefied that the Knudsen number that is relative to small dust-sized particles will be on the order of unity and, thus, the gas flow around these particles will not be well-described by Navier-Stokes equations. It will be necessary to account for this rarefaction in calculating the coefficient of drag for smaller particles. The $CO_2$ atmosphere has significant bulk viscosity that is negligible in the terrestrial atmosphere; indeed, to date, the effects of bulk viscosity for martian ballistics has not been well-characterized and are therefore neglected in this study. It is expected that bulk viscosity will only reduce the distance that particulates will travel, which means that if bulk viscosity is significant, this study represents a worse case.

The atmosphere on Mars will collimate the engine plumes so that interaction with soil will be much closer to the terrestrial case (experiments in sand beds in an atmosphere) than the lunar case (exhaust plume expanding into a vacuum). In the lunar case, a very broad, bowl-shaped shockwave forms over the regolith and shields the surface from the direct impingement of the plume. This shock is so broad, and the stagnation pressure beneath it is spread so widely, that deep cratering has not occurred in lunar landings. However, with an atmosphere, the standoff shock will be only as wide as the collimated plume with an abrupt drop in stagnation pressure around its circumference. This will be far more likely to penetrate the soil and create the type of deep craters that have been seen in terrestrial testing. Hence, the effects on Mars will dramatically differ from those on the moon.

The regolith on Mars (Squyres, 2004[49], Soderblom, 2004[50], Arvidson, 2004[51]) is probably very diverse but appears to be more akin to terrestrial than lunar regolith because it is looser, less cohesive, more porous, and overall more easily excavated by an impinging jet than the lunar regolith. In the absence of an atmosphere to regulate temperature, the extreme diurnal thermal cycling of the lunar regolith produces strong lunar quakes in the crust; over geologic time, these have shaken down and compacted the soil to a very dense state. On Mars, as on Earth, the temperature of the soil is moderated by the atmosphere so that heat is not directly radiated away to space, and the soil will not be as compacted by thermal cycling and diurnal shaking as on the moon. Further, the active Aeolian processes contribute to the relatively loose state of the soil at the surface, and its prior hydrological processes may have contributed to geological sorting of the particle sizes in some regions, enhancing the permeability and porosity of the soil. The particle shapes of weathered martian soil are probably more rounded like terrestrial particles than

[49] Squyres, S.W., et al. (2004), ”The Opportunity Rover’s Athena Science Investigation at Meridiani Planum, Mars,” *Science* **306** (5702), 1698-1702.
[50] Soderblom, L.A., et al. (2004), ”Soils of Eagle Crater and Meridiani Planum at the Opportunity Rover Landing Site,” *Science* **306** (5702), 1723-1726.
[51] Arvidson, R.E., et al. (2004), ”Localization and Physical Property Experiments Conducted by Opportunity at Meridiani Planum,” *Science* **306** (5702), 1730-1733.

the sharp, interlocking shapes of the unweathered lunar particles. All of these factors suggest that the soil is much weaker on Mars than on the moon, and so cratering effects are expected to be much greater for martian landings.

The mass and thrust of launching and landing spacecraft on Mars will also be intermediate to the conditions of the moon and Earth. With a larger crew and longer surface stay, the spacecraft will have a much greater landed mass than the Apollo lunar modules had. In addition, because of the greater surface gravity, the thrust to land these spacecraft must be much greater than in the lunar case.

#### *5.9.2.3 Deep cratering on Mars*

To characterize the martian plume effects, the principal question is whether the deep cratering of bearing-capacity failure and/or diffusion-driven shearing will occur, thus producing the deep, narrow crater characteristic of those processes. If so, the gas that is exiting the crater will eject rocks and soil into the vertical direction and then, as the crater erodes and broadens, into a wide range of elevation angles. There are presently two methods to predict the onset of deep cratering, and both predict that deep cratering will occur for human-scaled landers on Mars.

The first method is the purely empirical limit of 0.3 kPa stagnation pressure on the soil, which was determined during the Viking tests. This limit is still applied by the space science community in assessing the retrorocket plumes for martian robotic landers. While the low mass of the robotic landers has made it possible for them to stay below this limit, the heavy human-scaled landers must necessarily exceed it. For a 40-t lander that has four engines that use nominal engine conditions, the pressure is calculated to be 47 kPa, more than 150 times the limit. For a 60-t lander, this increases to 71 kPa, or about 240 times the limit. This implies that deep cratering will undoubtedly occur. However, the uncritical use of this empirical limit contains at least two flaws. First, it is obvious that a soil can easily support much more pressure without cratering than this limit suggests, as long as the pressure tails off very gradually over a long radial distance. It is not just the pressure, however, but the drop off of pressure around the edges that causes the soil to fail. The particular characteristics of a plume will determine how abruptly or gradually the stagnation pressure tails off. Second, it should be noted that setting a limit that is based on pressure assumes that bearing-capacity failure alone is the mechanism of cratering. If gas diffusion into the soil plays a role, the limit should consider the permeability of the soil, not just the stagnation pressure and its gradient. However, for soils that do not differ too greatly from those pressures and gradients that were used in the Viking study, and for nozzle exit diameters and plume conditions that do not differ too greatly from those tests, the empirical limit is the best method that is available to predict cratering.

The second method to predict deep cratering is an analytical theory that was developed by Alexander, et al. To date, this is the only theory or model that attempts to predict the onset and depth of cratering, so it is worth using for comparison with prior work. Nevertheless, it is necessary to point out that it contains three known flaws, which means that its predictions may not be valid. These three flaws are detailed as follows:

- First, it assumes that the stagnant gas that is under the impinging plume can play two roles simultaneously: (1) applying mechanical pressure at the surface of the soil, and (2) diffusing fully into the soil to reach a steady-state flow condition that maximally weakens the soil as in Terzaghi's effective stress hypothesis. This is wrong, however, because the net mechanical pressure that is applied to the surface of the soil (delta-pressure between the top and bottom sides of the top layer of sand grains) vanishes as the gas reaches steady-state diffusion into the soil. Thus, in the theory, the surface pressure driving the bearing-capacity failure should have been reduced to a degree that is commensurate with the diffusion.
- Second, Terzaghi's effective stress hypothesis deals with essentially static fluid in the soil, the pressure of which relieves some of the hydrostatic portion of the stress in the mechanical skeleton of the soil. However, in this case, the fluid is highly dynamic and its aerodynamic drag through the soil produces a distributed body force that actually contributes to the stress in the mechanical skeleton of the soil rather than relieving it. Thus, the pressure at the surface of the soil is reduced as diffusion proceeds, and is replaced by a body drag force throughout the bulk of the soil; this transitions the cratering effects from a case of bearing-capacity failure to one of diffusion-driven shearing. In the latter, the soil moves in a direction that is perpendicular to the former; therefore, the Alexander, et al. theory is inadequate to describe both cases. Unfortunately, it is difficult to create an analytical model of diffusion-driven shearing equivalent that achieves the elegance of the theory by Alexander, et al.

- Third, the Alexander, et al. theory assumes a geometry for soil motion that is unrealistically restricted to a narrow cylinder around the crater, resulting in too much predicted soil resistance. Recent experiments with a supersonic rocket exhaust in sand against a glass window have shown that the shearing geometry is much wider than this.

In summary, the Alexander, et al. theory without gas diffusion will under-predict the cratering because it neglects both diffusion-driven shearing and the lower-resistance shearing geometry of the soil; but the theory with gas diffusion is conceptually wrong, and it is unknown whether it will over- or under-predict the cratering.

With this in mind, the Alexander, et al. theory has been applied as follows: In the Viking era, Ko applied the theory with gas diffusion for the original design of a Viking lander engine (prior to the multi-nozzle engine) in martian conditions and demonstrated that bioconcentration factor (BCF) would occur from a descent height of 5 m down, producing a crater as deep as 60 cm. A much larger engine for a 40-t lander would therefore be expected nominally to produce a deeper crater. The present study has performed the calculations for such a case, but without the gas diffusion, to obtain an assuredly low-end prediction. These calculations were performed using soil cohesion and friction angle inferred from recent data that were collected at Meridiani Planum by the Opportunity rover  The calculations predict that the crater will be 2.0 m deep, which is the approximate, inferred depth to martian bedrock in the Meridiani Planum region. A sensitivity study was performed on the various parameters in the theory, and it was shown that, under all conceivable conditions, this cratering will occur. To demonstrate that this 2-m prediction is an under-prediction of the depth of the crater, the theory was also applied to the recent terrestrial test conditions of 50-N solid rocket motors firing into a deep quartz sand bed, and the theory without gas diffusion predicts only a 5-cm-deep crater compared with the actual craters that repeatedly formed to 55 cm depth. Thus, the theory without gas diffusion grossly under-predicts, by a factor of 11, in these test conditions. If the theory were updated to properly include gas diffusion leading to diffusion-driven shearing, and to use a more realistic shearing geometry, the prediction for a 40-t lander would undoubtedly be deeper than 2 m, although we cannot presently say how deep.

An integrated numerical model is being developed under a NASA contract that should be capable of predicting all of these effects. In the meantime, we can conclude with reasonable confidence that narrow cratering deeper than 2 m will occur beneath the large landers that are needed for human exploration. Simulation of the ballistics will therefore assume typical conditions for gas exiting a narrow crater as the worst situation that will occur during a nominal landing. As the crater grows and widens (as demonstrated in tests), the exiting gas velocities will lessen. The ballistics that are calculated from the narrow exit condition will therefore represent the worst case.

#### *5.9.2.4 Martian ballistics compared to terrestrial ballistics*

Ballistics equations were developed for the martian environment. For aerodynamic drag on a particle-neglecting rarefaction, we have used the Schiller and Nauman (Schiller and Naumann, 1933[52]) correlation

$$C_D = \left(\frac{24}{\mathrm{Re}}\right)\left(1 + 0.15\,\mathrm{Re}^{0.687}\right)$$

where the Reynolds number Re is based on a particle diameter. To account for rarefaction, the following factor by Carlson and Hoglund (Carlson and Hoglund, 1964[53]) is applied to the coefficient of drag:

$$\frac{1}{1 + \mathrm{Kn}\left[3.82 + 1.28\exp\left(-\mathrm{Kn}^{-1}\right)\right]}$$

where the Knudsen number Kn is based on a particle diameter. This method appears to under-predict the drag on the smallest particles at high Reynolds and Knudsen numbers, but these small particles have low terminal velocity and low mass and so are not a concern for damage calculations.

[52] Schiller, L. and A. Naumann (1933), "Uber die grundlegenden Berechungen bei der Schwerkraftaufbereitung." *Z. Deut. Ing.* **77**, 318-320, cited in [30].

[53] Carlson, D.J. and Hoglund, R.F. (1964), "Particle Drag and Heat Transfer in Rocket Nozzles," *AIAA Journal* **2**(11), 1980-1984, cited in [30].

Analysis of the ballistics as shown in figure 5-57 indicates that large silica particles will fall 3.5 times faster on Mars than on Earth, but small silica particles will fall 3.15 times faster on Earth than on Mars. The transitional regime is in the range from 10 µ to 1 mm, with 157-µ particles (fine sand) having the same terminal velocity of 1.33 m/s on either planet. Of course, these martian calculations will vary seasonally and with altitude. The calculations here used an atmospheric temperature of 193 K and a pressure of 6.2 mbar for specificity.

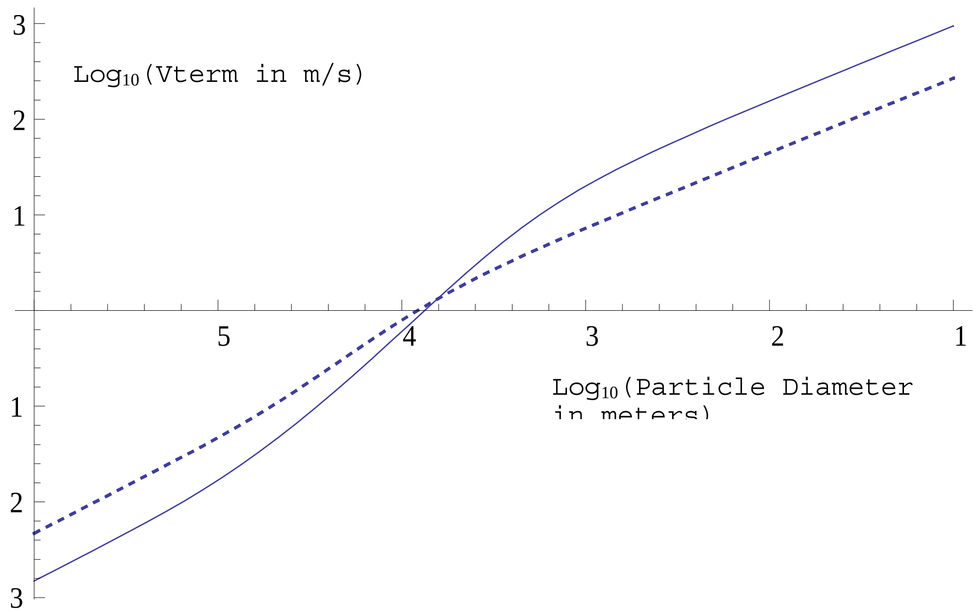



**Figure 5-57.** Terminal velocity as a function of particle size for Earth (dashed) and Mars (solid).

To develop a "sense" for martian ballistics, figure 5-58 shows a 630-µ particle (medium sand) that has been ejected at a 25-degree angle at a range of velocities for both martian and terrestrial conditions. On Mars, the particle travels more than twice as far as it does on Earth. For comparison, figure 5-59 shows a 100-µ particle (fine sand) with the same conditions. On Mars, the particle travels less than half as far as it does on Earth for the highest velocity, but farther on Mars for lower velocities.

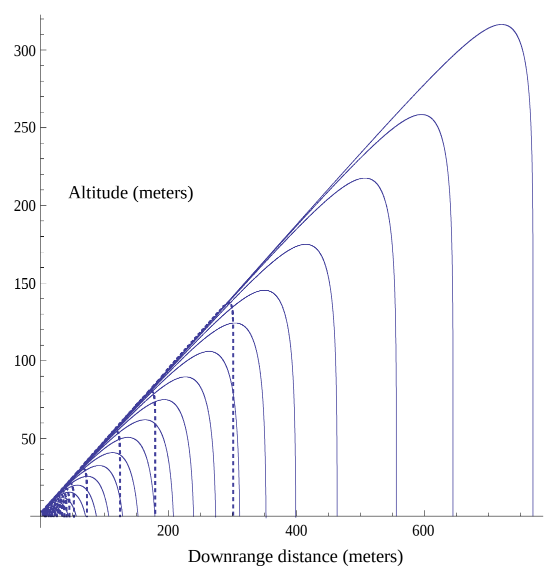



**Figure 5-58.** Trajectories for a 630-µ particle ejected at 25 degrees elevation for a variety of velocities.

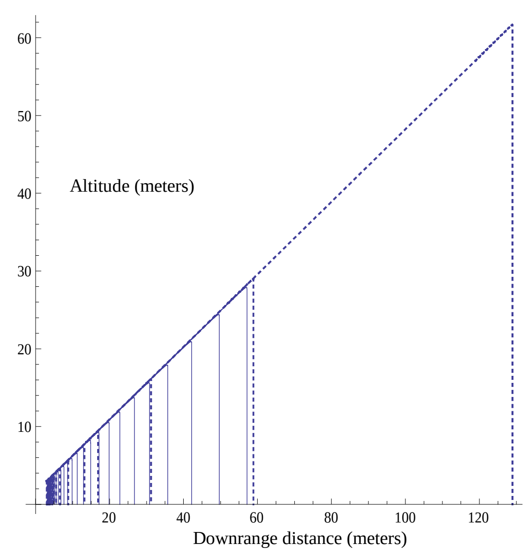



**Figure 5-59**. Trajectories for a 100-µ particle ejected at 25 degrees elevation for a variety of velocities.

### *5.9.2.5 Plume and crater model*

The martian atmosphere is sufficiently dense that standard CFD software can accurately model the plume. A number of plume simulations have been performed, and an example is shown in figure 5-60.

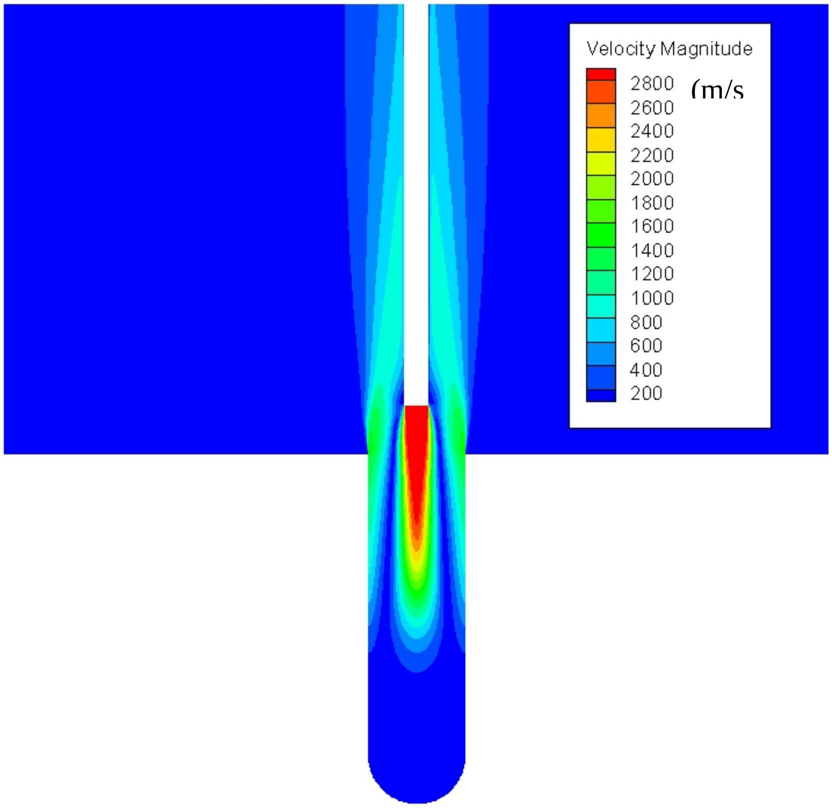



**Figure 5-60.** Computational fluid dynamics simulation of rocket exhaust plume.

The ratio of the engine nozzle diameter to crater diameter was chosen as typical from terrestrial experiments. With the geometry that is selected here, the engine exhaust gas exits the crater in a jet around the circumference of the crater, coaxial to the engine exhaust, and with maximum velocity near the lip where entrainment of particulates occurs. This agrees with test results, in which a vertical jet of entrained material leaves the crater. Since the cloud of small particulates that is blown from the crater will be stopped by drag forces at a close range to the lander, the blast zone radius will be determined by larger particles (gravel and rocks), which travel farthest. This damaging material will fall sparsely throughout the large blast zone, and the probability of being hit will depend on the quantity of each particle size that is ejected into each trajectory angle. Since even a single unacceptably high impact of large material must be prevented, we can define the blast radius as the maximum distance at which any unacceptable impacts may occur. Therefore, we assume that the particles are accelerated by the gas velocity that is shown in figure 5-60, but with exit angles that vary from horizontal to vertical. The gas jet velocity exiting the crater is on the order of 1,000 m/s over a distance of 5 m for a lunar-module scale engine, and 1,300 m/s over 5 m for a nominal engine on a 40-t lander. These values shall be used in the ballistics calculations.

### *5.9.2.6 Blast zone predictions*

Typical trajectories of ejected particles are shown in figure 5-61; these have been calculated with a 1,000-m/s coaxial jet from the crater.

Particles that are in the 3-mm size range (coarse sand to fine gravel) travel the farthest (700 m) and hit with the highest velocity (43 m/s). The heavier particles (as shown with the symmetrically arched trajectories) do not travel as far because the plume is not able to accelerate them to as high an initial velocity as the 3-mm particles. The lighter particles that have the more asymmetric trajectories have initially greater velocity leaving the plume but do not travel as far because they lose kinetic energy to the ambient atmosphere and fall straight down at their terminal velocities (as shown with the asymmetric trajectories). The 3-mm particles nominally set the blast radius to the 700 m to 1 km range. However, hardware designers may find that the momentum and energy of these particles is not too severe and may be successfully shielded. Therefore, it may be possible to set a smaller blast radius. Although the 3-mm particles have the highest impact velocity, the heavier particles falling at smaller radii have greater impact momentum and energy, as shown in figures 5-62, 5-63, and 5-64.

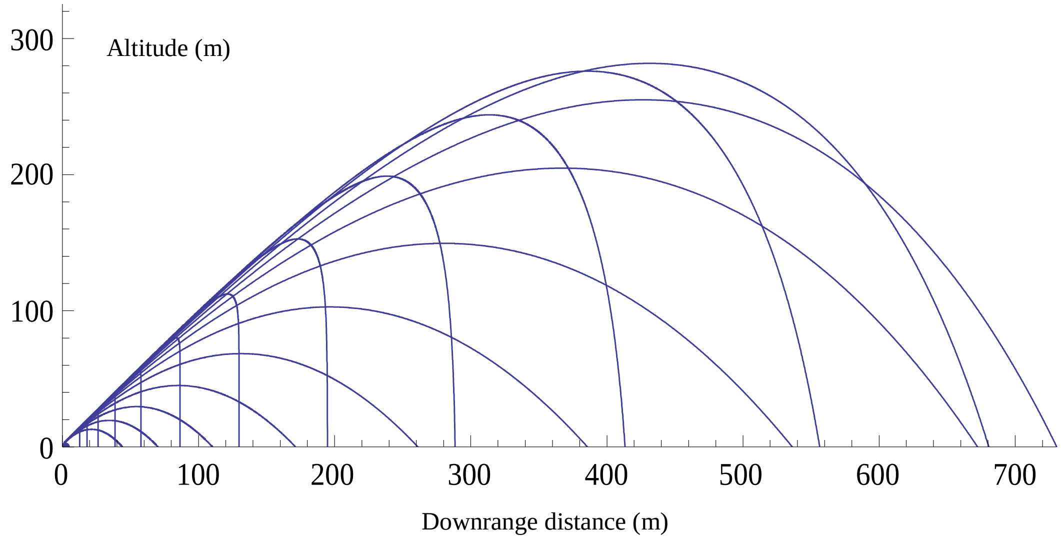



**Figure 5-61.** Particle trajectories ejected from a crater at 45 degrees for 23 different particle sizes.

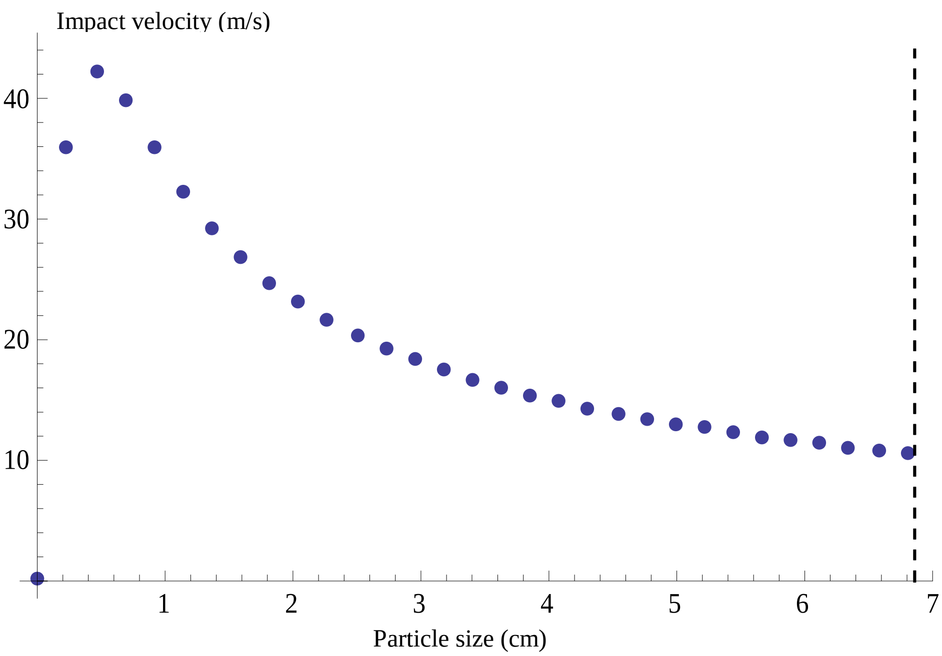



**Figure 5-62.** Impact velocity vs. particle size.

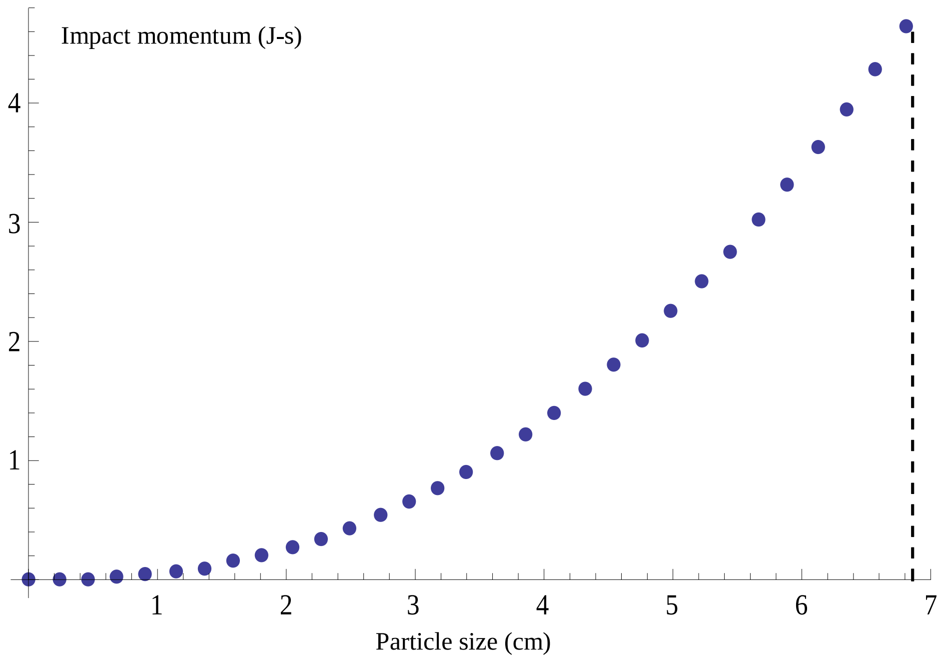



**Figure 5-63.** Impact momentum vs. particle size.

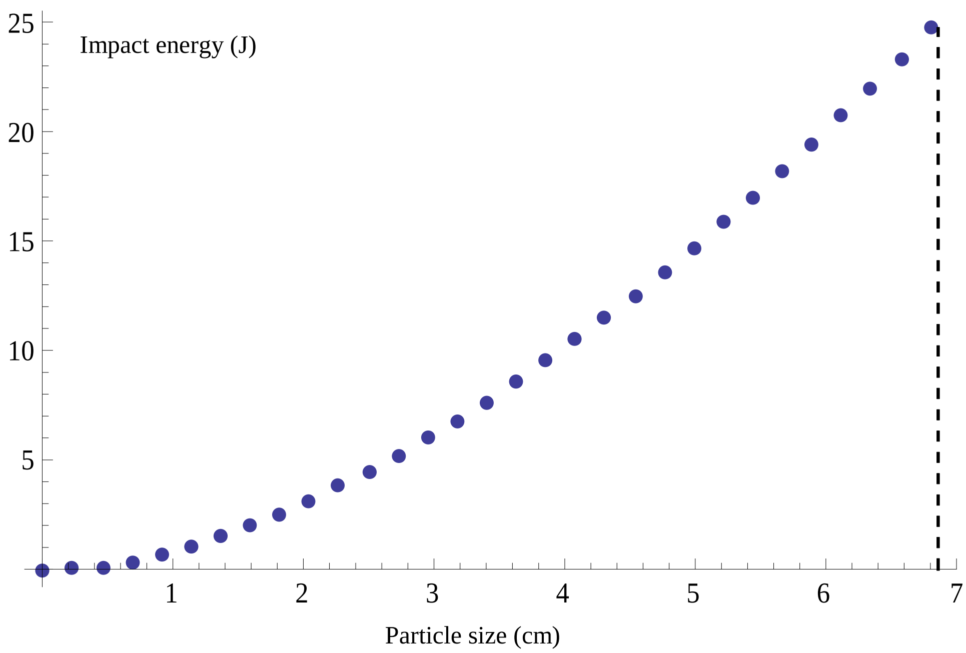



**Figure 5-64.** Impact energy vs. particle size.

Impact momentum and energy as a function of distance are shown in figures 5-65 and 5-66, respectively.

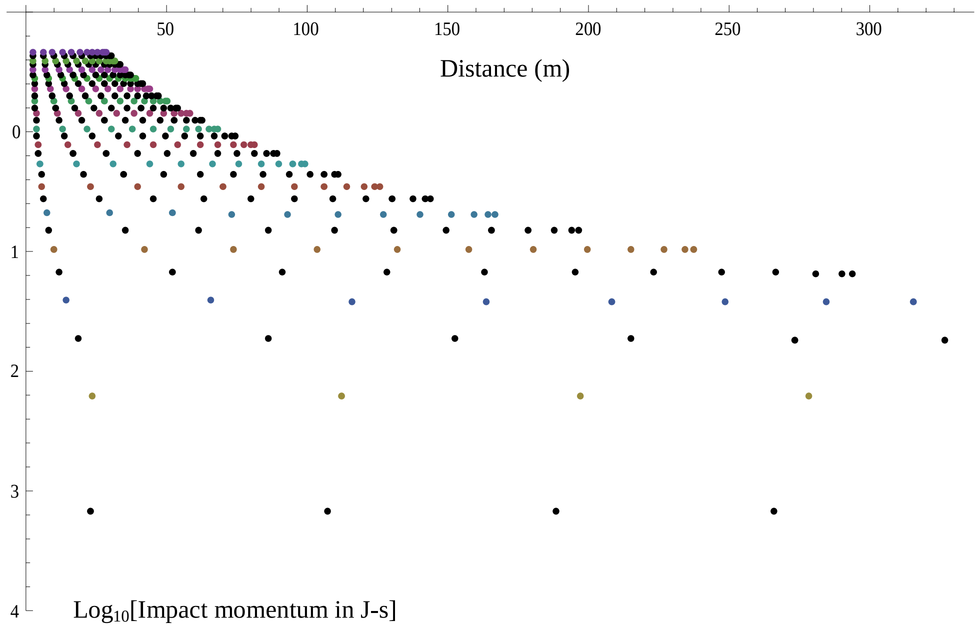



**Figure 5-65.** Semilog scatter plot of impact momentum vs. distance.

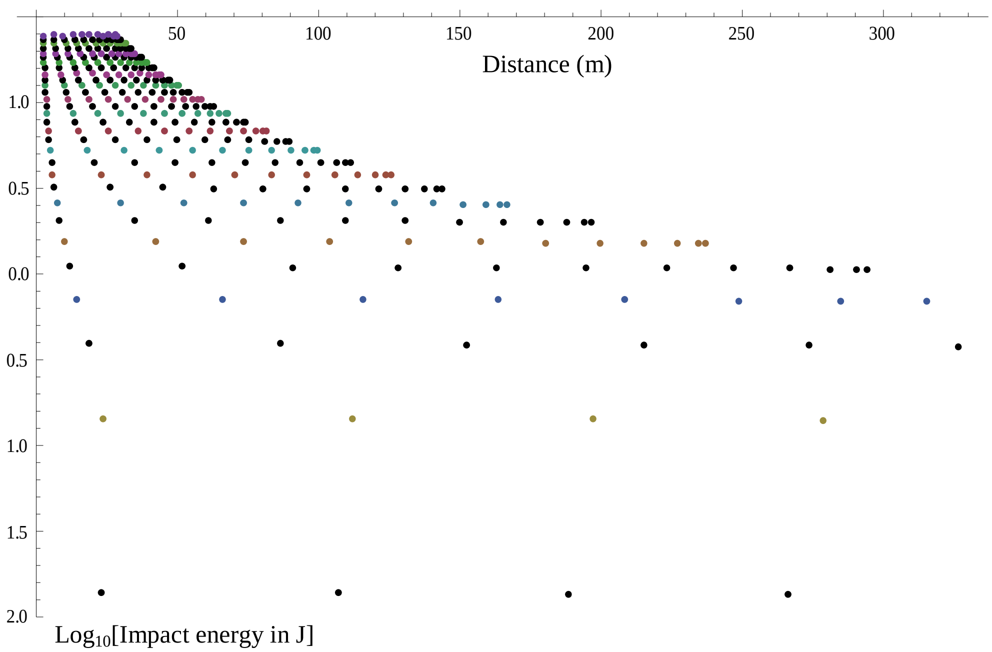



**Figure 5-66.** Semilog scatter plot of impact energy vs. distance.

### *5.9.2.7 Ejecta impingements on the lander*

Ejecta will not only strike hardware in the surrounding area but, because of the vertical ejection angles from the narrow crater, will strike the bottom of the lander itself. Impact velocities, momentum, and energy as a function of particle size are shown in figures 5-67, 5-68, and 5-69, respectively. Generally, these increase as the lander descends to lower altitudes. Shielding that is sufficient to withstand these momenta and energies will be required to protect the bottom of the lander. The shielding must also protect the nozzles, since eject that is blown by one engine may strike the nozzle of another engine.

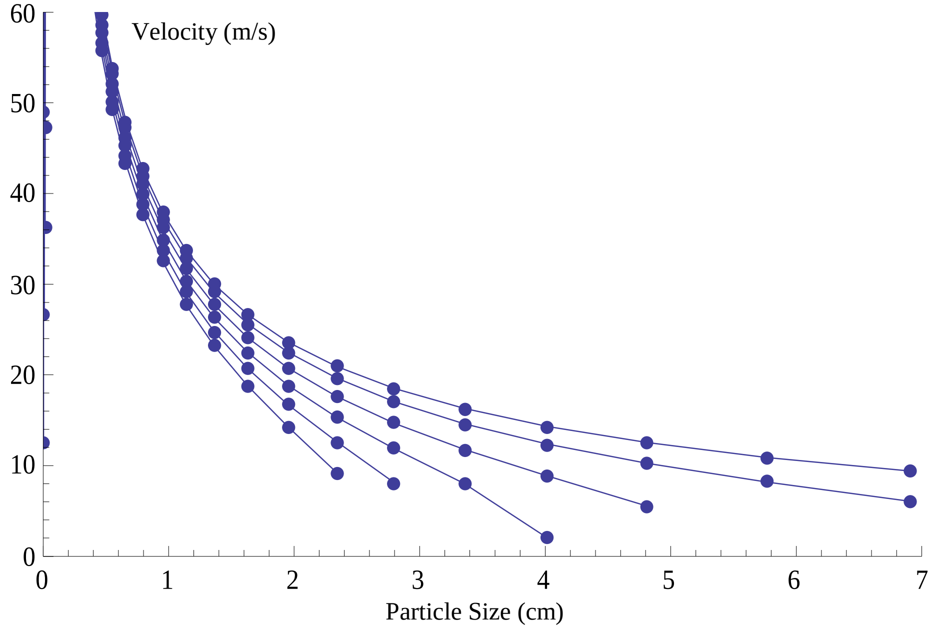



**Figure 5-67.** Impact velocity on the bottom of the lander as a function of particle size.

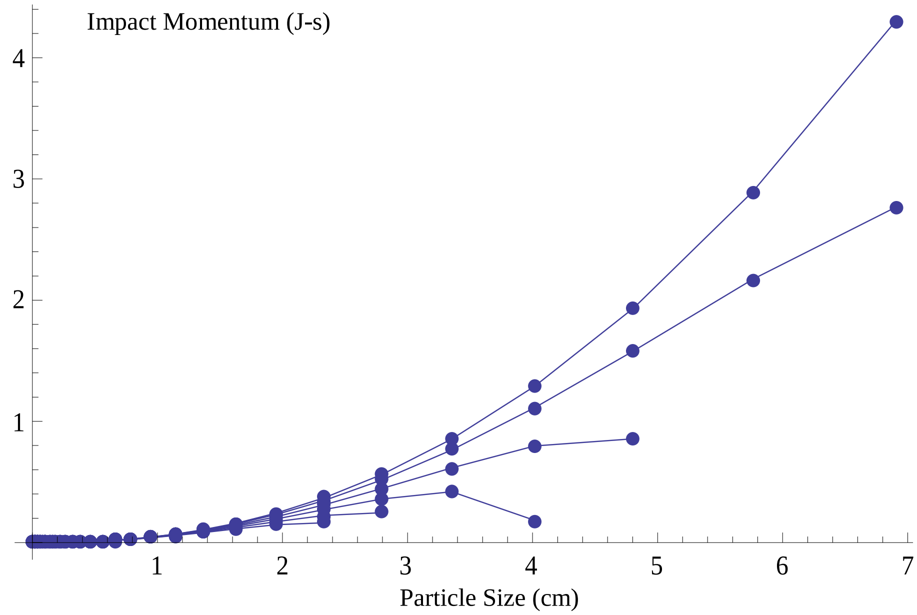



**Figure 5-68.** Impact momentum on the bottom of the lander as a function of particle size.

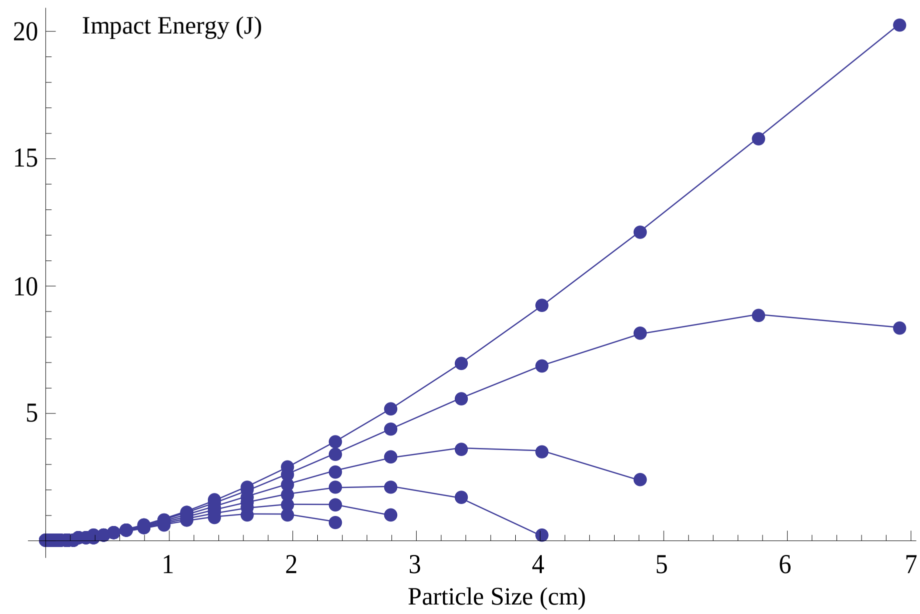



**Figure 5-69.** Impact energy on the bottom of the lander as a function of particle size.

#### *5.9.2.8 Regolith stability after landing*

Experiments have shown that the deep, narrow crater that is beneath the plume will maintain its shape and slowly broaden as long as the engine continues firing; but when the engine cuts off, the crater collapses into a cone-shaped "residual crater" as shown in figure 5-56. In experiments, the volume of the residual crater has been verified to equal the volume of the deep, narrow cylindrical crater that existed prior to engine cutoff. For noncohesive or only slightly cohesive soils, such as the upper layers of soil that were measured by the MERs, the crater will collapse into a cone at the angle of repose, which will be about 20 degrees. Thus, the residual crater will be very broad and may extend well past the footpads of the lander. Until development of the modeling tools is completed, we are not able to predict this quantitatively. However, if we assume for order-of-magnitude that the crater is of the diameter that is shown in figure 5-60 and has a depth of only 2 m to martian bedrock with very little additional erosion to widen it after reaching bedrock, the residual crater will have a width of 7.3 m. With four engines clustered toward the center of the lander, calculations show that the single residual crater resulting from the collapse of four closely spaced holes will have a width of 11.6 m. For a lander with a 10-m diameter, the legs can be configured so that this crater will not reach to its footpads and, thus, the lander should be stable after landing. However, if the bedrock is deeper and the crater is excavated more deeply than 2 m, or if the erosion removes significantly more soil, the residual crater may well reach to the lander footpads. Also, if the bedrock is much shallower than 2 m and the soil is stripped away, exposing bare bedrock, that might present an unexpectedly uneven landing surface.

#### *5.9.2.9 Additional lander mass for mitigating plume effects*

A rough estimate may be made for the additional mass that would be required for plume effects mitigation. This includes (1) wider landing gear to avoid the unstable regolith and the formation of a broad crater beneath the lander, and (2) shielding on the lander base-plate to protect the engine nozzles and other items from the impact of flying rocks, gravel, and sand. This is for the case where there is no prepared landing site on Mars, which means that the soil would not have been modified in anticipation of the arrival of the lander. The goal here is only to compute order of magnitude to determine whether plume effects mitigation will have a significant effect on Mars architecture.

The landing gear estimate is made by comparison with the mass of the Apollo-style lunar landing gear (four legs with crushable honeycomb). Scaling up the mass of the gear proportionately to the mass of the lander and doubling to account for the greater strength that is needed for the greater martian gravity, the landing gear on a Mars lander would be roughly 2,400 kg, not accounting for plume effects. To avoid regolith instability, the gear may need to extend an additional 2.5 m in the radial direction, or 7.5 m from the center of the lander rather than the nominal 5 m (assuming the lander is 10 m in diameter). So the mass of the gear would be increased proportionately by 50%

(order of magnitude), which is 1,200 kg additional mass. As a result, the payload mass that could be safely landed on Mars would need to be reduced by 1,200 kg.

For the shielding, we assume the equivalent mass of an enhanced Whipple shield, which is used on the ISS today, across the bottom of the lander. This comes to approximately 1,000 kg. This mass may be reduced by using a lower-mass shield, but additional mass may be required on the upper portions of the lander due to rocks that may be thrown vertically and fall onto the lander from above. (This possibility cannot be ruled out in the complex fluid dynamics of the blast zone with an uneven landing surface.)

Based upon these estimates, for landing on an unprepared surface on Mars, the plume impingement and cratering effects would require a wider landing gear and a shielding on the lander with a total mass delta increase of 1,200 kg + 1,000 kg = 2,200 kg.

For a Mars architecture that incorporates robotic landing site preparation prior to human arrival (e.g., leveling, compacting, and sintering to stabilize a landing zone), it would be possible to use a smaller landing gear and omit the extra blast shielding. Thus, the landed payload mass would be higher by about 2,200 kg, all other things being equal. These site-preparation roles may be included in the design of an ISRU excavator that is designed to extract ice from the martian regolith. For example, the excavator may level a landing site and then compact it by removing loose surface material followed by tamping. The excavator may also be equipped with a microwave transmitter to glassify the surface of the soil to a desired depth of penetration, creating an in-situ landing pad.

It should be noted that even with the additional landing gear and shielding, a Mars landing on an unprepared site may be significantly more risky than the alternative with surface preparation. For example, even with wider landing gear, it is possible that a slight horizontal translation during final descent could put the gear into a crater that had begun to form while the lander was higher up. More specific design work will be required in future to ensure that the lander is protected during landing and stable after landing.

#### *5.9.2.10 Conclusions*

A minimum 1-km blast radius should be sufficient to protect hardware around the landing vehicle. This radius may be reduced so that hardware can land closer together if the hardware is shielded to protect against the impacts of ejecta. The ejecta will have increasingly higher momentum and energy at closer distances. The bottom of the landing spacecraft will be subjected to significant ejecta strikes due to the presence of a narrow crater that will form under each engine, redirecting the exhaust jets back up toward the lander. Therefore, the bottom of the lander will need to have its nozzles and other features adequately shielded. The ground under the lander will present some localized instability at engine cutoff as the excavated hole under each engine collapses. The extent of this instability for cohesionless soil is predicted to be, at a minimum, the same diameter as the lander. Thus, specific design work to assure lander stability will be required in future. These are all order-of-magnitude estimates because the numerical tools that are required for better predictions are still being developed. Questions about the visibility during landing, the spoofing of landing sensors, the deposition of dust on hardware after landing, or the chemical contamination of soil around the lander have not been addressed.

# 6 SURFACE SYSTEMS

Candidate surface sites will be selected based on the best possible data that are available at the time of the selection, the operational difficulties that are associated with the site, and the collective merit of the science and exploration questions that can be addressed at the site. Data that are available for site selection will include remotely gathered data sets plus data from any landed mission(s) in the vicinity plus interpretive analyses that are based on these data.

Figure 6-1 illustrates a notional series of traverses to features of interest at the junction of the Isidis Planatia and Syrtis Major regions. No particular preference is being given to this site; it is included here to illustrate some general features of a human exploration mission and the resulting implications for operations at such a site.

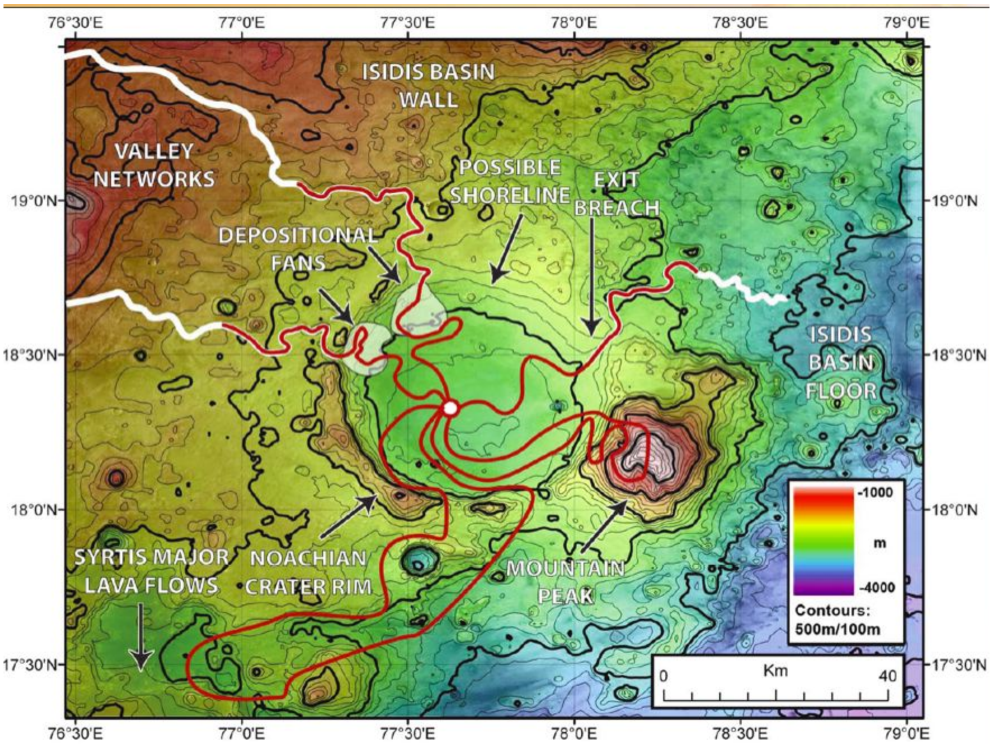


**Figure 6-1.** Notional traverses in the region located near the junction of the Isidis Planatia and Syrtis Major.

From an operational perspective, this location has a relatively broad, relatively flat, centrally located area where the cargo elements can land in relative safety. However, this places these systems and the crew at large distances from the features that are of interest to the crew and the science teams. The scale at the lower right of the figure indicates that these features of interest are beyond what is currently considered a reasonable walking range for the crew (determined by the distance that a crew member can walk during one charge of power and breathing gases in his/her portable life support system (PLSS); i.e., ~20 km total). Although sites with much more closely spaced features of interest are certainly possible, they are usually found at the expense of a relatively safe landing site. Thus, a nominal set of traverses for any of the first three human Mars missions is likely to be on the order of 100 km radial distance from the landing site. Therefore, based on several notional sites, including the one that is shown in figure 6-1, these traverses could be much longer than a simple 200-km round-trip.

One feature of interest is not illustrated here – the subsurface. Understanding the vertical structure of the site will also be of interest, indicating that a drilling capability will be included for each mission and site. The ability to move a drill from location to location will also be desirable.

Three possible approaches to satisfying this desired combination of horizontal and vertical exploration were created during this Reference Architecture assessment. These three options, which were given the working titles of “mobile home, “commuter,” and “telecommuter,” were constructed to focus on different approaches to accomplish these two exploration “directions.” It is recognized other combinations and permutations of these basic functions could also satisfy these high-level goals; but given the time and resource constraints of this Reference Architecture assessment, only these three options were examined. An overview of each will be discussed in the next several paragraphs, and the resulting implications to various surface systems will be discussed in the following subsections.

The “mobile home” surface mission scenario assumes that surface exploration by the crew will be primarily a mobile operation. Thus, this scenario assumes the use of two (for mutual support) large, capable, pressurized rovers for extended traverses that would spend between 2 and 4 weeks away from the landing site (see figure 6-2). These rovers will have space and resources allocated for on-board science experiments. The landing site is assumed to have infrastructure elements that are not needed for the extended traverses, such as an ISRU plant (making $O_2$ (probably), $CH_4$ (probably), $H_2O$, and any buffer gases that are residual from processing the martian atmosphere for these other commodities) and a large power plant. The processing capacity of this ISRU plant is TBD and dependent to a certain degree on the assumed implementation for the rover power source, which is assumed to be nuclear. The landing site will also be the “pantry” for food and other basic maintenance and repair capabilities; the landing site would have minimal crew habitation capabilities. With this division of functions among the surface systems, it is assumed that the crew will make a number of traverses away from the landing site, but return periodically to resupply and refit the rovers before deploying on the next traverse (this will be discussed in more detail below).

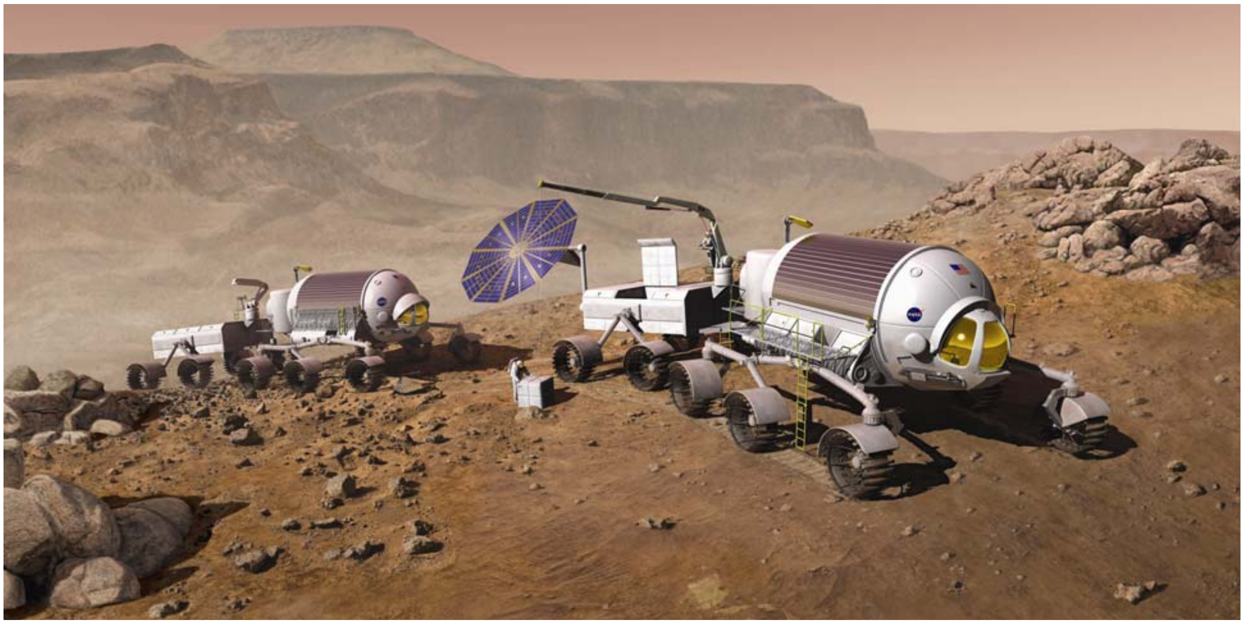


**Figure 6-2.** A notional design for the “mobile home” large pressurized rovers (Rawlings 2007[1]).

Because the rovers would be designed for mutual support, each rover would be sized for a nominal crew of three but would be able to accommodate all six crew members in a contingency situation. Thus, the majority of the habitation functions would be replicated in each rover. However, while both rovers will have space and resources allocated for on-board science experiments, it is assumed that these experiments would not be replicated in both rovers. Each rover would have an airlock to support EVA activity, but it is also assumed that the rovers would be able to routinely dock together, thereby allowing the crew to transfer between vehicles without the need for an EVA (e.g., at night when all traverse and EVA activities have concluded). This will allow the entire crew to use any of the assets in either vehicle on a regular basis.

[1] Drawing courtesy of Rawlings, 2007

In addition to the internal science experiments that are mentioned above, the pressurized rovers would also bring along two small robotic rovers; two unpressurized, small (comparable to the Apollo LRV) rovers to carry EVA crews; and a drill. The two robotic rovers can be teleoperated from the pressurized rover or be given a set of instructions and allowed to carry out these instructions in an automated fashion. The unpressurized rovers will allow the EVA crews to move relatively quickly between sites within walk-back range of the pressurized rovers once the latter have stopped for extended operation at a given location. Note that it is assumed that the pressurized rovers will not be very nimble and, thus, will serve as a “base camp” from which local traverses will be staged. A notional traverse in this scenario would be for the entire crew to deploy in the two pressurized rovers for up to 1 month before returning to the landing site where they will then spend several weeks doing maintenance and repair on the rovers and EVA equipment, plus restocking the rovers with expendables from the “pantry” before deploying on the next traverse. The timeline in figure 6-3 illustrates a mixture of 2- and 4-week traverses, separated by a 2-week refit/replanning period following a traverse and a 4-week refit/replanning period following every fourth or fifth consecutive traverse to allow for more substantial repairs, in-situ analysis, or replanning for the next sequence of traverse. The crew will have the capability to drill to shallow depths – 10’s of meters – at more than one site, typically once or twice during each traverse.

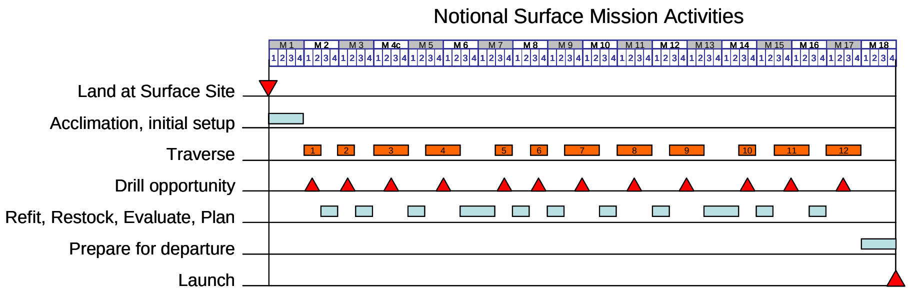



**Figure 6-3.** A notional timeline for the “mobile home” scenario.

With the limited resources that were available for this study, a very preliminary estimate was made of the mass for each of the surface system elements that were used in this “mobile home” scenario and their distribution between the two cargo elements that were used to deliver them to Mars. Table 6-1 provides a summary of these masses and their distribution.

**Table 6-1.** Mass Summary for Surface Systems for the “Mobile Home” Scenario

| | | **Hab Lander** | **DAV Lander** |
|---|---|---|---|
| **Manifested Item** | **Quantity** | **CBE Mass (kg)** | **CBE Mass (kg)** |
| Crew Consumables | | 1,500 | 4,500 |
| Science | | 0 | 1,000 |
| Robotic Rovers | 2 | 0 | 500 |
| Drill | 1 | 0 | 250 |
| Unpressurized Rover | 2 | 500 | 0 |
| Pressurized Rover | 2 | 27,800 | 0 |
| Pressurized Rover spares | | 539 | 1,617 |
| Pressurized Rover growth | | 2,777 | 0 |
| Pressurized Rover power | 2 | 4,608 | 0 |
| Traverse Cache | | 0 | 0 |
| Logistics/Repair Module | 1 | 0 | 4,500 |
| Stationary Power System | 2 | 7,300 | 7,300 |
| ISRU Plant | 1 | 0 | 1,305 |
| Ascent stage 1 (no $LO_2$) | | 0 | 12,156 |
| Ascent stage 2 (no $LO_2$) | | 0 | 9,330 |
| 30-day temp hab | | 0 | 0 |
| Descent Stage (wet) | | 24,300 | 24,300 |
| Aeroshell | | 44,200 | 44,200 |
| **Total IMLEO Mass** | | 113,600 | 11,100 |

The "commuter" surface mission scenario, which was adopted as the nominal scenario for this Reference Architecture as depicted in figure 6-4, assumed a centrally located, monolithic habitat; two small pressurized rovers; and two unpressurized rovers that were roughly equivalent to the Apollo LRV. Power for these systems would be supplied by a nuclear power plant that would be previously deployed with the DAV and used to make a portion of the ascent propellant. Although the traverses would be a significant feature of the exploration strategy that is used in this scenario, these traverses would be constrained by the capability of the small pressurized rover. In this scenario, these rovers are assumed to have a modest capability, notionally a crew of two, a 100-km total distance before being resupplied, and no more than 1-week duration. Thus, on-board habitation capabilities would be minimal in these rovers. However, these rovers are assumed to be nimble enough to place the crew in close proximity to features of interest (i.e., close enough to view from inside the rover or within easy EVA walking distance of the rover). Not all of the crew would deploy on a traverse, so there would always be some portion of the crew in residence at the habitat. The pressurized rovers would carry (or tow) equipment that would have the capability to drill to moderate depths – 100's of meters – at the terminal end of several traverses.

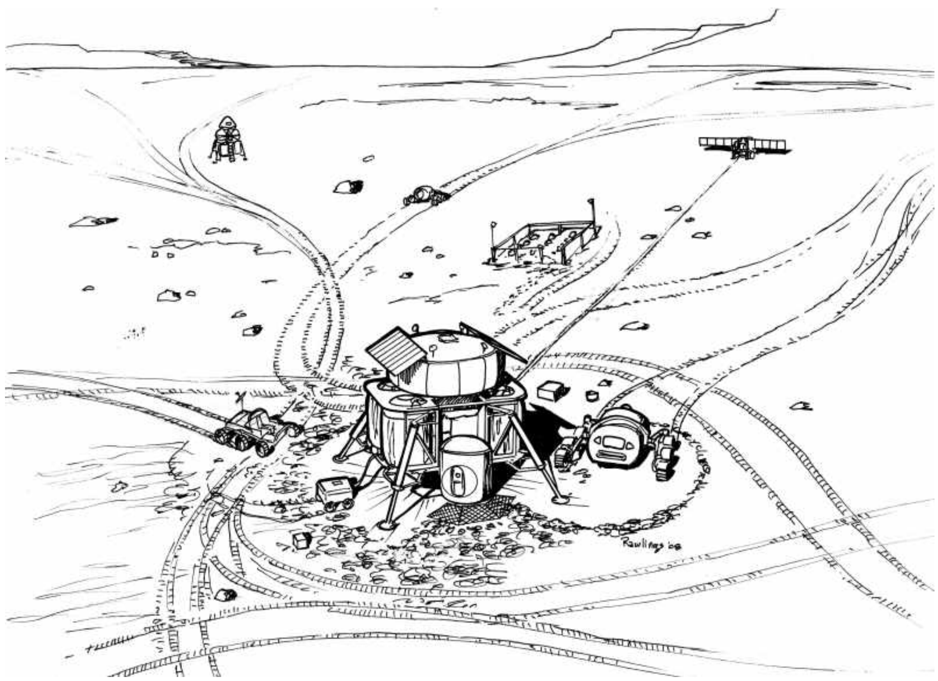



**Figure 6-4.** Notional view of the surface systems used in the "commuter" scenario (Rawlings, 2007).

The primary habitat would have space and resources allocated for on-board science experiments. The pressurized rovers would carry only the minimal scientific equipment that was deemed essential for field work (in addition to the previously mentioned drill); samples would be returned to the primary habitat and its on-board laboratory for any extensive analysis.

One approach to accomplishing the desired long traverses would be to use the pressurized rovers (or possibly the robotic rovers) to preposition supplies in caches along the proposed route of travel prior to the "full-duration" traverse. Thus, a typical traverse would begin with the crew (or robotic rovers) traveling out a nominal distance (~15 km, or EVA walk-back distance) and establishing a cache of commodities for life support and power (possibly emergency habitation) before returning to the habitat. Some amount of exploration-related activities may be accomplished during this cache-deployment phase, but the primary purpose is route reconnaissance and cache establishment. The crew then makes another traverse, establishing a second cache a like distance beyond the first cache. This process continues until all caches in this chain are built up sufficiently for the crew, in the two pressurized rovers, to make the entire round-trip traverse in the time duration that is needed to accomplish traverse objectives. The amount of time that is required to set up and retrieve these supply caches would depend on the specific conditions for a traverse. However, the timeline in figure 6-5 illustrates how much can be accomplished if approximately 2 weeks are allocated for establishing this string of caches and another 2 weeks to retrieve them. In addition, not all traverses will be long enough to require this type of support. A mixture of cache-supported and -unsupported traverses has been illustrated. Finally, some amount of time would be required in which to repair and restock the pressurized rovers after each traverse, as well as to conduct any local experiments and plan for the next traverse. A notional 2 weeks between short traverses and 4 weeks between long traverses has been illustrated in figure 6-5.

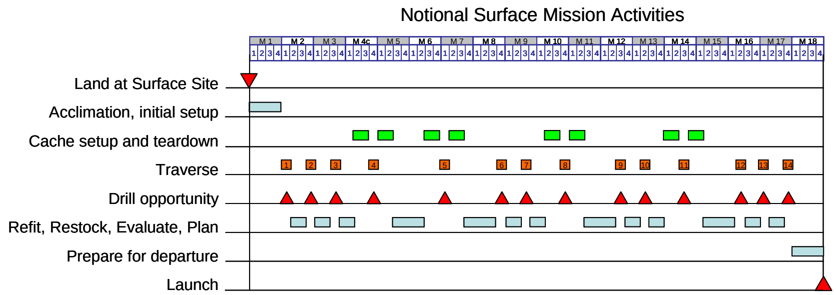



**Figure 6-5.** A notional timeline for the "commuter" scenario.

With the limited resources that were available for this study, a very preliminary estimate was made of the mass for each of the surface system elements and their distribution between the two cargo elements that were used to deliver them to Mars. Table 6-2 provides a summary of these masses and their distribution.

**Table 6-2.** Mass Summary for Surface Systems for the "Commuter" Scenario

| | | **Hab Lander** | **DAV Lander** |
|---|---|---|---|
| **Manifested Item** | **Quantity** | **CBE Mass (kg)** | **CBE Mass (kg)** |
| Crew Consumables | | 1,500 | 4,500 |
| Science | | 0 | 1,000 |
| Robotic Rovers | 2 | 0 | 500 |
| Drill | 1 | 0 | 1,000 |
| Unpressurized Rover | 2 | 500 | 0 |
| Pressurized Rover | 2 | 8,000 | 0 |
| Pressurized Rover spares | | (included above) | 0 |
| Pressurized Rover growth | | 1,600 | 0 |
| Pressurized Rover power | 2 | 0 | 1,000 |
| Traverse Cache | | 0 | 1,000 |
| Habitat | 1 | 16,500 | 0 |
| Hab growth | | 5,000 | 0 |
| Hab spares | | (included above) | 0 |
| Stationary Power System | 2 | 7,300 | 7,300 |
| ISRU Plant | 1 | 0 | 1,305 |
| Ascent stage 1 (no $LO_2$) | | 0 | 12,156 |
| Ascent stage 2 (no $LO_2$) | | 0 | 9,330 |
| 30-day temp hab | | 0 | 0 |
| Descent Stage (wet) | | 23,300 | 23,300 |
| Aeroshell | | 43,700 | 43,700 |
| **Total IMLEO Mass** | | 107,400 | 106,100 |

In the last case, the "telecommuter" scenario, it is assumed that the crew would be based in a centrally located, monolithic habitat and that only unpressurized (lunar rover equivalents) rovers will be used for EVAs. This implies traverses by the crew of no more than walk-back distances (~15 km radial distance). The long-range traverses would be handled by very capable robotic rovers (notionally a considerably improved MSL rover) that are teleoperated (or possibly supervised) by the surface crew members from their habitat (see figure 6-6). Because of the assumed pre-positioning of surface cargo, there is an opportunity to deploy these rovers independently from the large surface habitat (but during the same atmospheric entry event) to sites that are distant from the habitat landing site. In this situation, there would be as many as 2 years available for these rovers to carry out long-distances traverses, guided from Earth-based operators, with an ultimate destination of the habitat landing site. After the crew arrives at the habitat, these robotic rovers can be deployed on other traverses under the guidance of the surface crew.

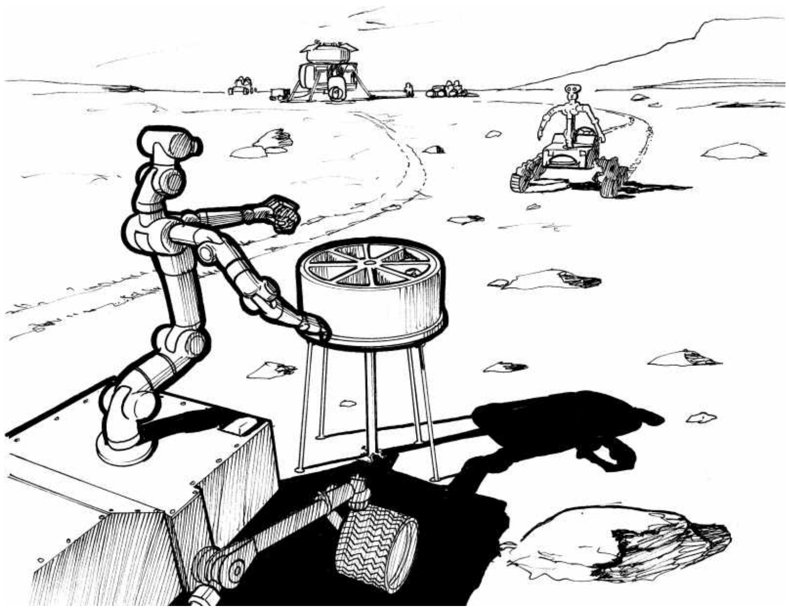


**Figure 6-6.** A notional image of teleoperated rovers for the "telecommuter" scenario (Rawlings 2007).

The timeline in figure 6-7 illustrates what can be accomplished with month-long robotic traverse by two rovers, which would be separated by 2-week refit/restock periods at the end of each traverse, with an extended refit/restock period after a sequence of three traverse to allow for more extensive repairs (if necessary) and to evaluate the data that were collected thus far as the foundation for the next sequence of traverses.

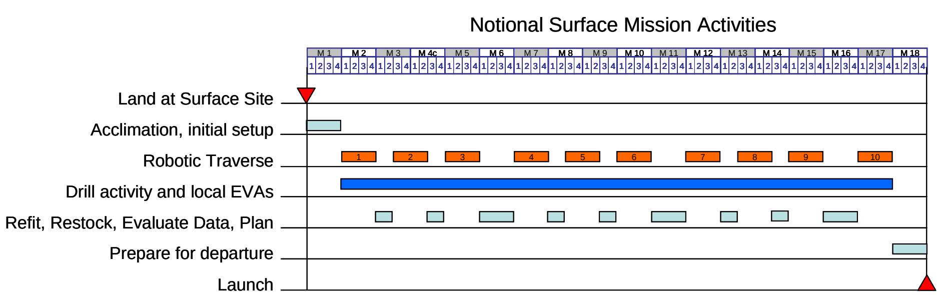



**Figure 6-7.** A notional timeline for the "telecommuter" scenario.

The primary habitat would have space and resources allocated for on-board science experiments. The unpressurized rovers and EVA crews would carry only the minimal scientific equipment that was deemed essential for field work (in addition to the previously mentioned drill); samples would be returned to the primary habitat and its on-board laboratory for any extensive analysis. The robotic rovers would carry a more extensive suite of instruments that was suitable for long-range and long-duration traverses, but would have the capability to acquire and return samples to the primary habitat for further analysis and, possibly, for return to Earth.

The human crew's primary EVA job will be to set up a drill that is capable of very deep drilling (thousands of meters?), if desired, and for operating that device. The ISRU plant may provide drilling fluid (e.g., liquid $CO_2$?) for this device.

As in the “mobile home” scenario, there would be an ISRU plant at the landing site/habitat site that would be making the same kinds of commodities. This ISRU plant and, subsequently, the habitat would be served by a large (assumed to be nuclear) power plant. The habitat would serve as the pantry and maintenance/repair facility as described above.

With the limited resources that were available for this study, a very preliminary estimate was made of the mass for each of the surface system elements and their distribution between the two cargo elements that would be used to deliver them to Mars. Table 6-3 provides a summary of these masses and their distribution.

**Table 6-3.** Mass Summary for Surface Systems for the “Telecommuter” Scenario

| | | Hab Lander | DAV Lander |
|---|---|---|---|
| **Manifested Item** | **Quantity** | **CBE Mass (kg)** | **CBE Mass (kg)** |
| Crew Consumables | | 3,000 | 3,000 |
| Science | | 0 | 1,000 |
| Robotic Rovers | 2 | 0 | 2,000 |
| Drill | 1 | 1,000 | 0 |
| Unpressurized Rover | 3 | 750 | 0 |
| Habitat | 1 | 16,500 | 0 |
| Hab growth | | 5,000 | 0 |
| Hab spares | | (included above) | 0 |
| Stationary Power System | 2 | 7,300 | 7,300 |
| ISRU Plant | 1 | 0 | 1,305 |
| Ascent stage 1 (no $LO_2$) | | 0 | 12,156 |
| Ascent stage 2 (no $LO_2$) | | 0 | 9,330 |
| 30-day temp hab | | 0 | 0 |
| Descent Stage (wet) | | 23,300 | 23,300 |
| Aeroshell | | 43,300 | 43,300 |
| **Total IMLEO Mass** | | 100,100 | 102,700 |

## *6.1 Surface Habitats*

The objective of the Mars habitat analysis was to estimate mass and power for three configuration options. The three options were identified as mobile home, commuter, and telecommuter. Each option offered a different approach to surface mobility, and was selected to represent an extreme or mid-point in the trade-off analysis range.

### 6.1.1 Approach

The first step in the approach was to establish GR&As. This defined the excursion range, crew size, and other attributes for each of the options. Next, an MEL that had been created for recent lunar habitat studies was used as a point of departure for the Mars options. This was a logical starting point because space habitats share similar subsystems, and the MEL incorporated the latest detailed input from subsystem specialists. Each of the subsystems was examined to determine the mass and power changes that would be required to accommodate the Mars habitat options (see figure 6-8). For this work, changes were the product of engineering judgment rather than analysis.

### 6.1.2 Three habitat options

#### *6.1.2.1 Mobile home*

The mobile home option featured two identical pressurized rovers, each of which was sized for three astronauts. They would initially land close to one another and establish a base. From the base, the rovers would explore together on a nominal 30-sol mission. Each rover would provide its own power and thermal control and, in case of an emergency, be provisioned to accommodate the other three crew members during a retreat to the base. The base was not intended to be habitable, but more a resource cache for resupplying the rovers.

#### *6.1.2.2 Commuter*

The commuter option had a habitable base that remained on the lander and used two small pressurized rovers for exploration excursions.

#### *6.1.2.3 Telecommuter*
At the other end of the spectrum, the telecommuter option had a habitable base that remained on the lander and unpressurized rovers that the crew would use for tele-exploration.

### 6.1.3 Ground rules and assumptions
Some GR&As are common across all of the habitat options, and others are particular to a configuration. The GR&As were divided into two classes. One included the commuter and telecommuter options because they had habitats that was sized for a crew of six and remained on the lander. The other class was the mobile home configuration because it did not have a habitable base and was sized for a crew of three.

It was also assumed that the habitats would arrive on an automated mission preceding the arrival of the crew. Figure 6-8 shows the G&RAs for both classes.

| Ground Rule Assumption | Notes |
|---|---|
| Power required for continuous thermal control en route, before habitation and during crew occupancy (~4 years) | Maintain seals, equipment, and consumables within thermal specifications; coolant in-place for transit and surface operations |
| One airlock (A/L) per Mars habitat (MH), 2 A/Ls for commuter (C) and telecommuter (TC) | Crew safety, dual ingress/egress |
| 75% of habitat crew $H_2O$ and $O_2$ provided by ISRU | Orbiting habitat has 100% $H_2O$, closed-loop ECLSS |
| 100% of SPE radiation protection $H_2O$ provided by ISRU | $H_2O$ for shelter available on crew arrival (no additional $H_2O$ from Earth) |
| Habitat internal pressure same as LAT2 | 8 psi |
| Structural loads same as LAT2 | Mars entry/landing no greater than Ares launch loads |
| **MOBILE HOME** | |
| For MH, add 25% mass of mobility system for on-vehicle spares | Mobility intensive campaign, rough terrain, long excursions, 500-sol mission |
| MH uses fuel cell power source | Travel by night, PV charge by day |
| MH uses partially closed ECLSS | 30-day excursion below crossover |
| Each MH accommodates crew of three, six under emergency conditions | Permanent accommodations for three, temporary (emergency) accommodations for six |
| MH central station (base) accommodates crew of three | If one rover disabled, three crew in rover and three at base |
| No EVA cooling water provided from Earth for surface operations | Assumes new technology for cooling ($CO_2$), ISRU provided $H_2O$, or tethered operations |
| Provide habitat subsystem health prior to Earth departure | Assumes Earth control and monitoring of habitat |
| SPE radiation protection same as LAT2 habitat | Mars environment more benign than moon, but crew stay is longer and lunar approach accommodates worst-case site selection; additional mass for six crew members |
| Dust management same as moon | Will be different, but not enough to affect mass or volume for this assessment |
| Redundant habitable volumes; each accommodates all crew members in emergency situation | Infrequency return opportunities require survival until launch opportunity |
| Food and clothing brought from Earth; 10% food margin for emergency | Accommodates distribution due to separate pressure volumes |
| 250 EVAs per crew member (C and TC) | EVA every other day for 500 days |
| Fixed-base power provided lander | Mass for power is not book-kept by the habitat |
| Subsystems designed for maintenance | 500-day stay requires a design for continued operation while inspecting, servicing, or repairing subsystems |

**Figure 6-8.** Ground rules and assumptions.

### 6.1.4 Reference master equipment list
Habitat subsystems were defined and sized for a number of configuration options during the recent LAT2 study (see figures 6-9 and 6-10). This data base served as a starting point for the Mars habitats.

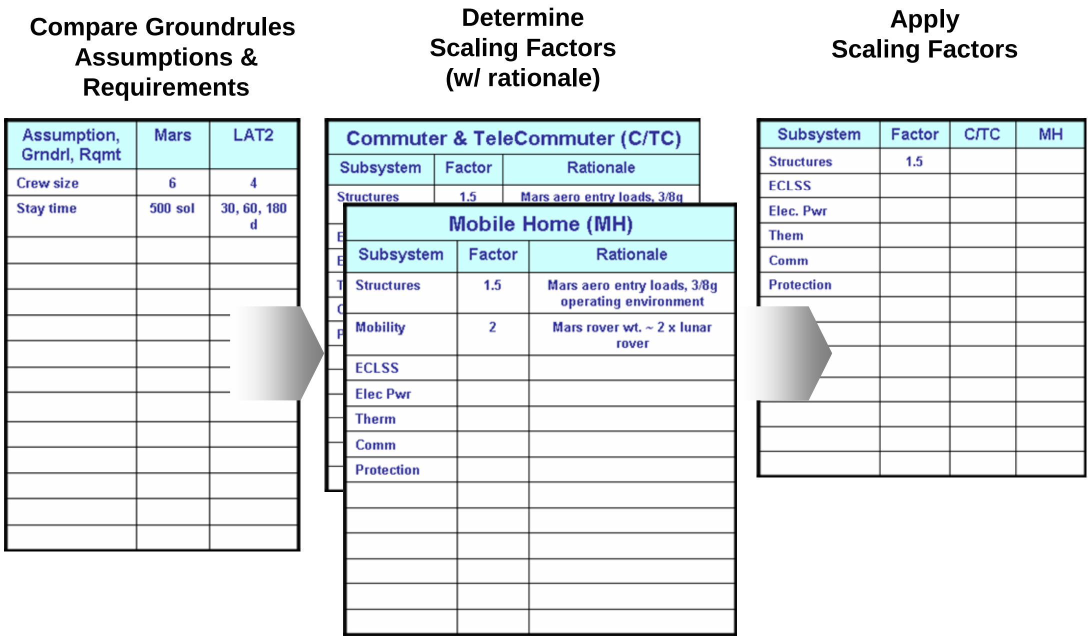
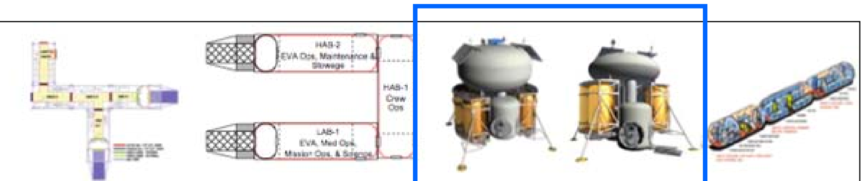



**Figure 6-9.** Mars habitat sizing approach.



| Parameter | Option 2 & 6<br>3 x 5 mini-Habs<br>Al-Li Hard Shell | Option 2<br>3 x 10 (deployed)<br>Hybrid Hab<br>Mid Section Expandable | Option 3<br>Concept #1<br>Monolithic<br>Habitat | Option 3<br>Concept #2<br>Monolithic<br>Habitat | Option 4<br>Mobile Habitat |
|---|---|---|---|---|---|
| # of Elements | 5 | 3 | 1 | 1 | 3 |
| Pressurized Volume | 137 $m^3$ | 189 $m^3$ | 198 $m^3$ | 154 $m^3$ | 165 $m^3$ |
| Floor Area | ~ 2 m x 4.75 m = 9.5 $m^2$<br>Total of 5 = 47.5 $m^2$ | ~ 2 m x 9.75 m = 19.5 $m^2$<br>Total of 3 = 97.5 $m^2$ | 41.17 $m^2$ | 33.45 $m^2$ | 3 m x 8.5 m = 25.5 $m^2$<br>Total of 3 = 76.5 $m^2$ |
| Mass | 5 units = 20375 kg | 3 units = 18384 kg | 12975 kg | 12280 kg | |
| Power | 12.4 kW | 12.6 kW | 10.3 kW | 10.3 kW | 9.8 kW |
| # of Available Ports | 3 | 4 | 3 | 2 | 2 |
| Outfitting | Pre-Integrated | Some Outfitting Required | Some Outfitting Required | Some Outfitting Required | |

**Figure 6-10.** Habitat options studied by Lunar Architecture Team 2.

Lunar habitats accommodated a crew of four and varied from an assembly of small modules to a one-shot delivery to a "train" of smaller mobile homes (see figure 6-11). Lunar habitat option 3 (figure 6-12) was designed as a habitat on a lander and therefore was used for the commuter and telecommuter classes. The Mars mobile home started with the lunar option 4 mobile habitat then incorporated relevant data from past pressurized rover studies, adjusting the subsystem distribution to accommodate crew size to ensure the autonomy of the Mars concept. An example of using the LAT 2 structural subsystem and scaling it to a Mars habitat is shown in figure 6-13. Another important difference with the mobile home concept was the addition of on-board electrical power for mobility and for thermal control. The result was a trailered power cart.

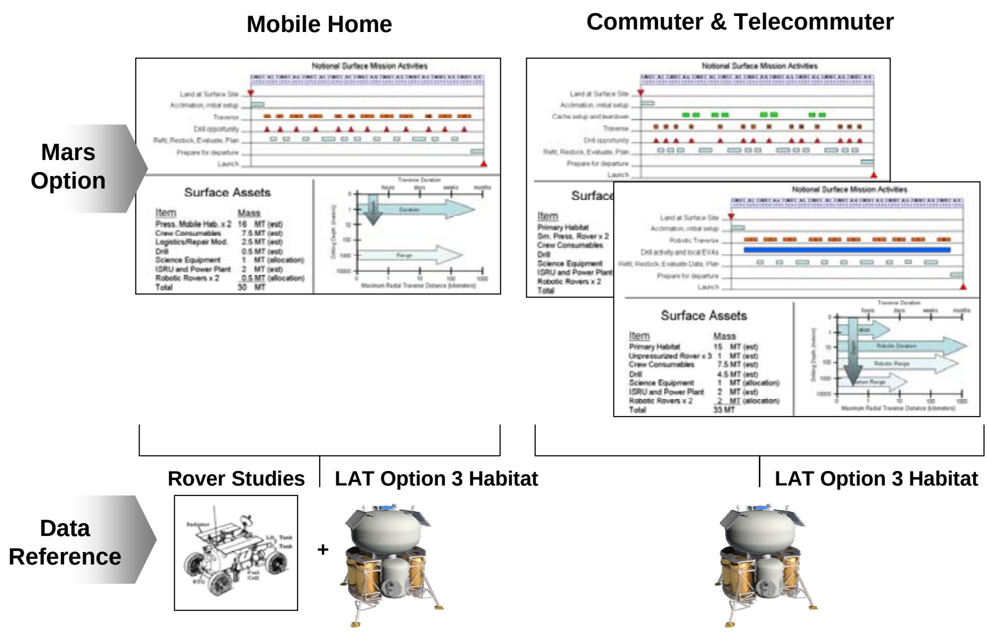
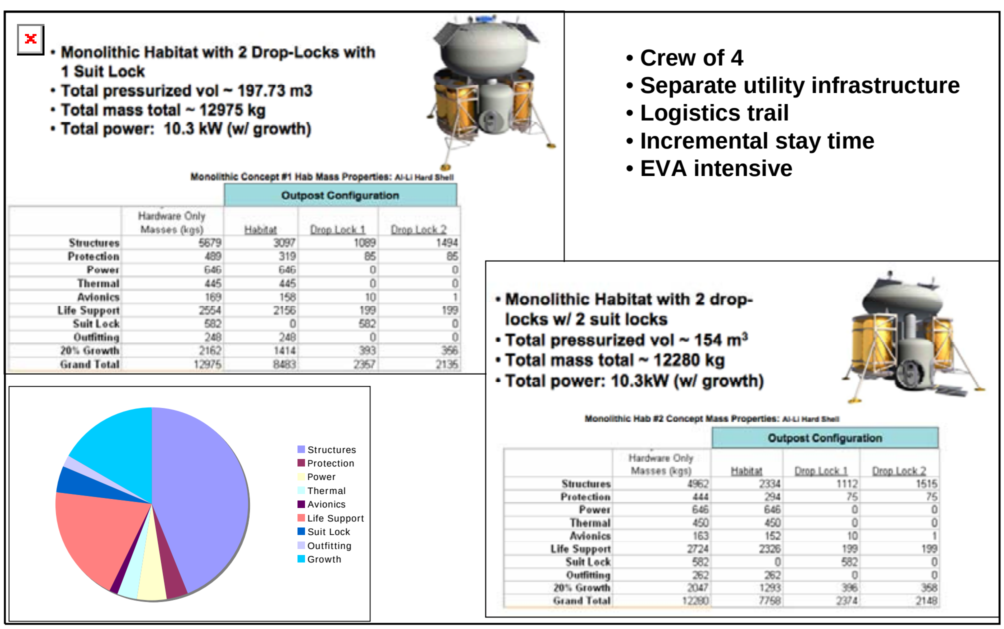



**Figure 6-11.** Mars Habitats Drawn from Lunar Architecture Team options.




Monolithic Concept #1 Hab Mass Properties: Al-Li Hard Shell

| | Hardware Only Masses (kgs) | Outpost Configuration: Habitat | Drop Lock 1 | Drop Lock 2 |
|---|---|---|---|---|
| Structures | 5679 | 3097 | 1089 | 1494 |
| Protection | 489 | 319 | 85 | 85 |
| Power | 646 | 646 | 0 | 0 |
| Thermal | 445 | 445 | 0 | 0 |
| Avionics | 169 | 158 | 10 | 1 |
| Life Support | 2554 | 2156 | 199 | 199 |
| Suit Lock | 582 | 0 | 582 | 0 |
| Outfitting | 248 | 248 | 0 | 0 |
| 20% Growth | 2162 | 1414 | 393 | 356 |
| Grand Total | 12975 | 8483 | 2357 | 2135 |

Monolithic Hab #2 Concept Mass Properties: Al-Li Hard Shell

| | Hardware Only Masses (kgs) | Outpost Configuration: Habitat | Drop Lock 1 | Drop Lock 2 |
|---|---|---|---|---|
| Structures | 4962 | 2334 | 1112 | 1515 |
| Protection | 444 | 294 | 75 | 75 |
| Power | 646 | 646 | 0 | 0 |
| Thermal | 450 | 450 | 0 | 0 |
| Avionics | 163 | 152 | 10 | 1 |
| Life Support | 2724 | 2326 | 199 | 199 |
| Suit Lock | 582 | 0 | 582 | 0 |
| Outfitting | 262 | 262 | 0 | 0 |
| 20% Growth | 2047 | 1293 | 396 | 358 |
| Grand Total | 12280 | 7758 | 2374 | 2148 |

**Figure 6-12.** Lunar habitats on the lander from Lunar Architecture Team 2.

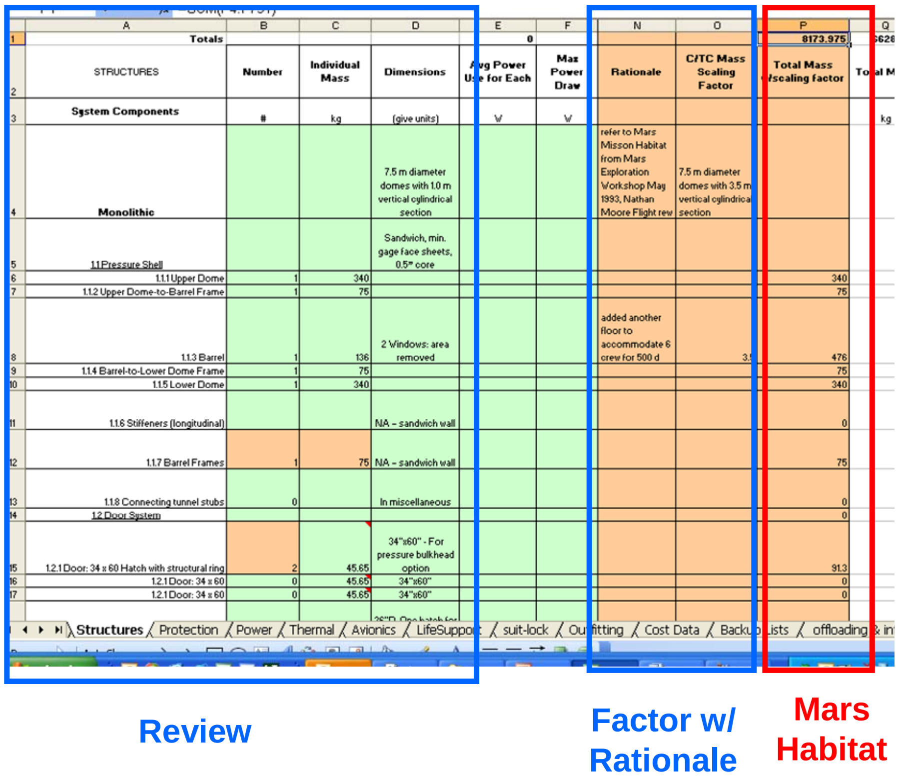



**Figure 6-13.** Example of scaling process.

### 6.1.5 Summary mass and power

Because a mission to Mars is a long journey without an opportunity for logistics resupply, each subsystem is determined to be a spares factor of additional mass that is to be delivered with the habitat. For totals, a 20% concept design factor was added. Table 6-4 shows the summary mass and power for the two classes of habitats. The commuter and telecommuter is approximately 30 t using 12.1 kW electrical power. Because there are two identical mobile home rovers, the 20% concept design factor was only applied to the first unit. At this, the total delivered mass is a little over 37 t with each drawing 13.6 kW average power. Table 6-5 shows the individual subsystem breakout for the Mars options.

**Table 6-4.** Summary of Mars Habitat Mass and Power

| Surface Habitat | Mass (kg) | Spares (kg) | Power (kW) |
|---|---|---|---|
| Commuter and telecommuter | 29,447 | 989 | 12.1 |
| | | | |
| Mobile home w/power<br>Second mobile home | 20,392<br>16,994 | 1,442<br>1,201 | 13.6<br>13.6 |
| **Total (2)** | **37,386** | **2,643** | **13.6 each** |

**Table 6-5.** Mars Habitat Subsystem Mass and Power

| | C & TC Subsystem & Fluids Breakdown | C & TC Subsystem Total Power (Watts) | Spares | Rationale |
|---|---|---|---|---|
| Structures | 8174 | 0 | 82 | LAT Option 3 scaled to crew of 6 with internal bulkhead; spares are based on 1% of total mass |
| Mobility System | 0 | 0 | 0 | None |
| Protection | 863 | 0 | 9 | SPE structure scaled to crew of 6 (water provided by ISRU not included in mass) Passive thermal and micro meteoroid scaled to surface area; spares are based on 1% of total |
| Power | 599 | 248 | 130 | |
| Power Generation | 0 | 0 | 0 | |
| Thermal | 785 | 349 | 39 | LAT Option 3 cold plates and pumps were scaled based on power loads; LAT Option 3 radiator was scaled based on best engineering judgment but requires further investigation; doesn't consider power generation power loads; assumed minimal spares (5%), not assessed at an individual equipment level |
| Avionics | 222 | 696 | 61 | LAT Option 3 scaled to crew of 6 with Mars mission factors |
| Life Support | 2767 | 3555 | 138 | assumed minimal spares (10%), not assessed at an individual equipment level |
| Suit Locks | 964 | 210 | 48 | assumed minimal spares (5%), not assessed at an individual equipment level |
| Outfitting | 8966 | 4298 | 198 | Scaled to a crew of 6 |
| Science Equip | 1200 | 700 | 120 | 10% spares |
| Sub total | 24539 | 10055 | 824 | |
| Growth | 4908 | 2011 | 165 | |
| 20% Grand Total | **29447** | **12067** | **989** | Spares location TBD |

| | Single MH Subsystem & Fluids Breakdown | Subsystem Total Power (Watts) | Spares | Rationale |
|---|---|---|---|---|
| Structures | 5100 | 0 | 51.00 | Boeing ACSS Pressurized Rover parametric for structure |
| Mobility System | 4420 | 5000 | 442 | Mass scaled LAT 2 mobility chassis E to 16600 kg less 3000 kg using pressure vessel structure, power from LAT2 chassis workbook 5kw avg, 20 kw peak |
| Protection | 431 | 0 | 4 | SPE scaled to crew of 3 structure only (water provided by ISRU) Passive thermal and micro meteoroid scaled to surface area; spares based on 1% of total mass |
| Power | 699 | 328 | 105 | Batteries required for supplemental power during mobility |
| Power Generation | 1605 | 200 | 161 | |
| Thermal | 764 | 349 | 38 | LAT Option 3 cold plates and pumps were scaled based on power loads; LAT Option 3 radiator was scaled based on best engineering judgment but requires further investigation; doesn't consider power generation power loads; assumed minimal spares (5%), not assessed at an individual equipment level |
| Avionics | 346 | 1149 | 95 | LAT Option 3 scaled to crew of 6 with Mars mission factors. Includes GN&C avionics for rover |
| Life Support | 1621 | 1430 | 81 | assumed minimal spares (10%), not assessed at an individual equipment level |
| Suit Locks | 482 | 105 | 24 | One LAT suit lock; assumed minimal spares (5%), not assessed at an individual equipment level |
| Outfitting | 926 | 2360 | 140 | Crew of 3 for 30 days |
| Science Equip | 600 | 350 | 60 | 10% spares |
| Sub total | 16994 | 11271 | 1201 | |
| Growth | 3399 | 2254 | 240 | |
| 20% Grand Total | **20392** | **13525** | **1441** | Spares location TBD |
| Second MH | 16994 | | 1201 | No growth in 2nd rover |
| Two Rovers | **37386** | | **2642** | |

**15389** MH without Pwr Generation
**18466** MH with 20% growth
1056 Water for radiation protection
245 3 crew
245 3 Mars suits
**20012** Total suspended mass without spares

**24818** Allegro Bus GCWR (Spartan chassis)

## *6.2 In-situ Consumable Production*

### 6.2.1 Introduction

The following sections describe the subsystems that were designed and sized for varying amounts and types of consumables production. Taken together, they constitute complete ISRU systems for deployment on the surface of Mars. These subsystems were developed through the use of models, some of which necessitated translation from lunar to martian applications. Additionally, exploratory ISRU transportation architectures and campaign analyses results are described in section 6.2.

### 6.2.2 Ground rules and assumptions

GR&As are described in the following sections, where they differ from or enlarge upon those that were described in section 3.5.4.

### 6.2.3 In-situ resource utilization options concepts

The options concepts that are associated with the decision package are described in section 3.5. The exploratory architecture options concepts may be found in sections 6.2.7 and 12.7.

### 6.2.4 In-situ resource utilization options considered

The ISRU systems were developed to cover options that produced consumables for (1) ECLSS/EVA closure only ,and (2) all consumables, meaning production for ECLSS/EVA closure and propellant. Three types of systems were developed to cover the options: Atmospheric $CO_2$-based, surface $H_2O$-based, and atmospheric $CO_2$ and $H_2O$-based (a combination of both). It should be noted that for some of these systems to operate, some supplies must be delivered from Earth (e.g., $H_2$). Additionally, scenarios where operational time and soil $H_2O$content were varied, and the results thereof, can be seen in the appropriate subsystem sections in terms of changes in mass, power, and volume requirements.

#### *6.2.4.1 Atmospheric carbon dioxide-based options*
See details below in corresponding subsystem sections.

#### *6.2.4.2 Surface water-based options*
See details below in corresponding subsystem sections.

#### *6.2.4.3 Atmospheric carbon dioxide and surface water-based options*
See details below in corresponding subsystem sections.

### 6.2.5 System designs
The complete ISRU system designs were developed from a combination of three main subsystems. These subsystems include: ISRU plant, excavation subsystem, and $H_2O$ extraction subsystem.

#### *6.2.5.1 Approach*
Each subsystem was generated using separate models, and the complete system was integrated after each subsystem analysis was complete for the various options that were considered.

#### *6.2.5.2 Results*
The results of the modeling analyses can be seen below in the corresponding subsystem sections.

### 6.2.6 Subsystem designs
#### *6.2.6.1 In-situ resource utilization plant (atmospheric acquisition and related subsystems)*
The atmospheric acquisition ISRU plant is designed to convert Mars atmosphere combined with $H_2$ from $H_2O$ that is extracted in situ into $O_2$ and $CH_4$ for use as propellants for Mars ascent and to supply the ECLSS with consumables. The plant, as shown in figure 6-14, is made up of a Sabatier chemical reactor that converts $CO_2$ and $H_2$ into $H_2O$ and $CH_4$. $CO_2$ is obtained via a microchannel adsorption pump, and the $H_2$ is provided by electrolyzing the $H_2O$ that is extracted via the $H_2O$ extraction subsystem. The $H_2O$ that is generated via the Sabatier reaction is then run through an electrolyzer to obtain $H_2$ and $O_2$. The $H_2$ is recycled back to the Sabatier reactor, and the $O_2$ is liquefied and stored for use as a propellant or for ECLSS consumables. The $CH_4$ that is generated by the Sabatier reaction is liquefied and stored in the ascent vehicle propellant tank.

Other trades performed were to use Mars atmosphere to generate $O_2$ for the ECLSS alone and for $O_2$ production for ECLSS and ascent oxidizer. The system that performs $O_2$-only production, the solid oxide electrolysis system, is pictured in figure 6-15. This system operates by taking Mars atmosphere via a microchannel adsorption pump and sending it to a solid oxide electrolyzer that is made from stacks of Zirconia. The Zirconia is heated and the $O_2$ molecules are stripped out and passed downstream to the $O_2$ tank, where it is liquefied and stored for ECLSS and/or ascent propellant uses.

**Approach**
The approach that was taken for modeling of the atmospheric acquisition ISRU plant was to divide it into three subsystems: the atmospheric acquisition subsystem, the consumable generation subsystem, and the liquefaction subsystem. The atmospheric acquisition subsystem is made up of the following component models: filter, microchannel $CO_2$ adsorption pump, valves, flow controllers, buffer gas pump, and buffer gas tank. The consumable generation subsystem is made up of a Sabatier reactor, $H_2O$ electrolyzers, filters, and valves. The liquefaction subsystem is made up of cryocoolers for $CH_4$ and $O_2$, $H_2O$ dryers, filters, and valves. Since the plant is driven more by power than mass, redundancy is accomplished by the use of two separate ISRU plants, each of which is sized to generate the needed consumables.

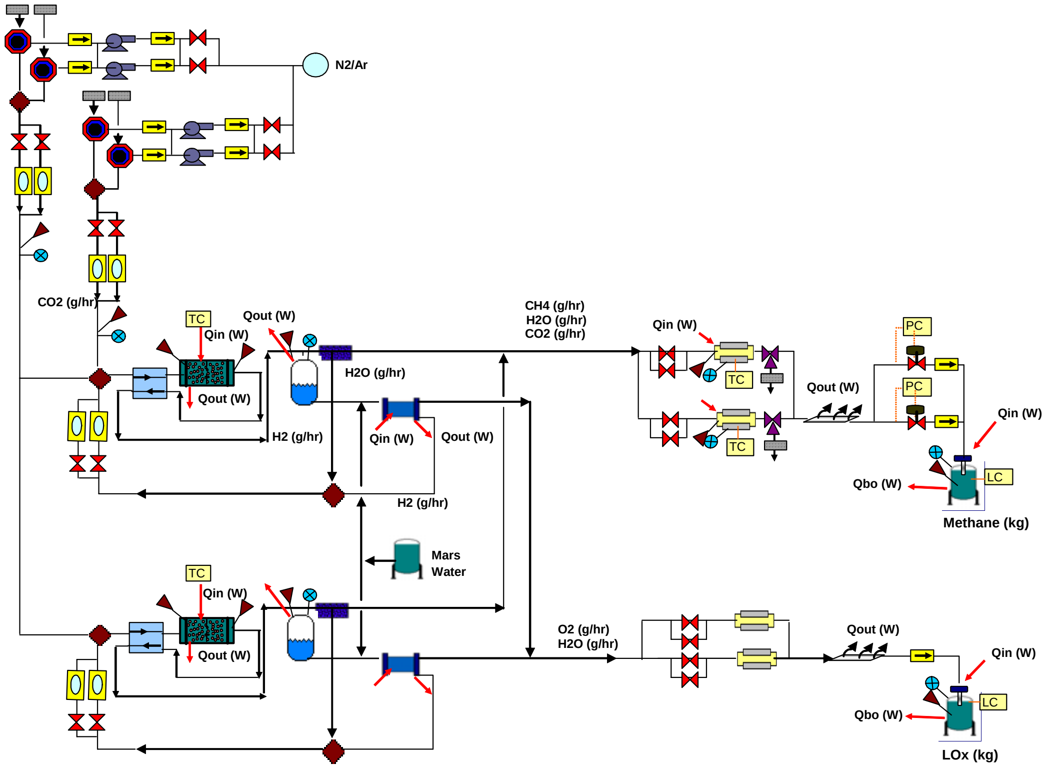



**Figure 6-14.** Schematic of the Sabatier water electrolysis system.

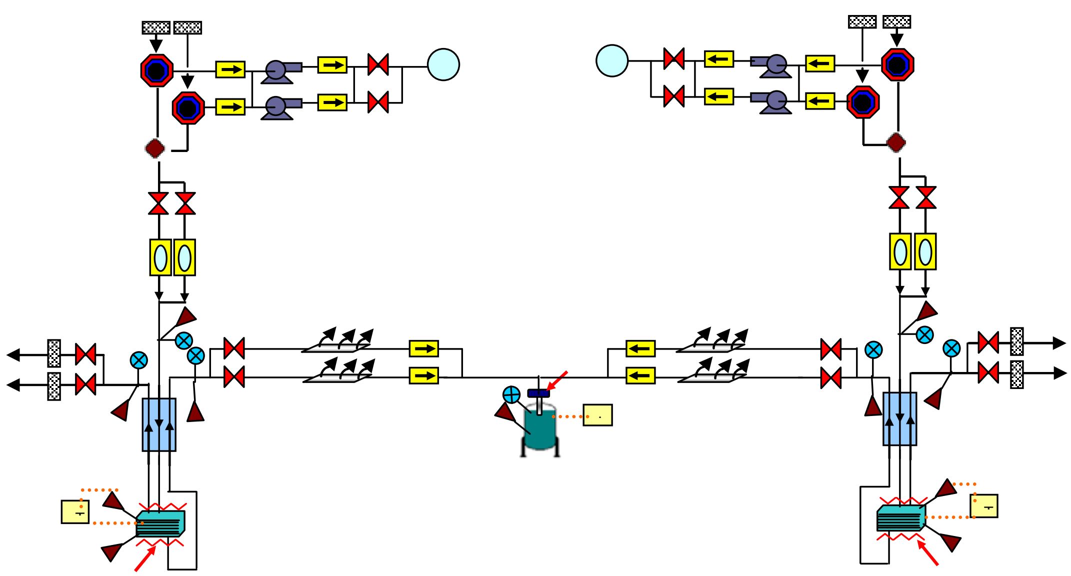


**Figure 6-15.** Schematic of the solid oxide electrolysis system.

**Results**

The atmospheric acquisition ISRU plant model provides results for mass, power, and volume for each subsystem in the plant. Results were based on the above plant producing all the necessary $O_2$ and $CH_4$ for an ascent vehicle as well as consumables for ECLSS, which consist of $H_2O$, $O_2$, and inert gases ($N_2$ and Ar, which are a by-product of the Mars atmosphere. The mass, power, and volume of the system and associated components are recorded in the table below. Results for the Sabatier reactor ISRU plants and solid oxide electrolysis plants are below in tables 6-6, 6-7, 6-8, and 6-9.

**Table 6-6.** Results of the Atmospheric Acquisition In-situ Resource Utilization Plant

| System Element/Description | Subcomponents | Sabatier Water Electrolysis 24-hour Production | | | |
|---|---|---|---|---|---|
| | | Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
| **ISRU System** | | | **479.12** | **0.80** | **24.98** |
| Atmospheric Acquisition Subsystem | | 2 | 288.12 | 0.61 | 3.655 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Microchannel $CO_2$ Adsorption Pump | 4 | 6.50 | 0.0015 | 3.652 |
| | Check Valve | 8 | 0.10 | | 0 |
| | Buffer Gas Pump | 4 | 1.23 | 0.002 | 0.003 |
| | Isolation Valve | 8 | 0.50 | | 0 |
| | Buffer Gas Tank | 1 | 250.00 | 0.6 | 0 |
| | Flow Controller | 4 | 0.50 | | 0 |
| Consumable Generation Subsystem | | 2 | 165.80 | 0.002 | 17.35 |
| | Sabatier Reactor | 2 | 3.50 | 0.001 | 0.55 |
| | Electrolyzer | 4 | 38.50 | 0.008 | 16.80 |
| | Isolation Valve | 8 | 0.50 | | 0 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Check Valve | 4 | 0.10 | | 0 |
| Liquefaction Subsystem | | 1 | 25.20 | 0.18 | 3.98 |
| | $CH_4$ Cooler | 2 | 3.00 | 0.04 | 1.24 |
| | $O_2$ Cryocooler | 2 | 5.00 | 0.05 | 2.68 |
| | Water Dryer | 4 | 2.00 | 0.003 | 0.06 |
| | Filter | 2 | 0.10 | | |
| | Isolation Valve | 8 | 0.50 | | |

**Table 6-7.** Solid Oxide Electrolysis for ECLSS Oxygen Operating at 8 Hours per Day

| System Element/Description | Subcomponents | Solid Oxide Electrolysis 8-hour Production | | | |
|---|---|---|---|---|---|
| | | Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
| **ISRU System** | | | **361.00** | **0.79** | **5.76** |
| Atmospheric Acquisition Subsystem | | 2 | 332.80 | 0.64 | 4.116 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Microchannel $CO_2$ Adsorption Pump | 4 | 15.20 | 0.0036 | 4.1 |
| | Check Valve | 8 | 0.10 | | 0 |
| | Buffer Gas Pump | 4 | 3.70 | 0.007 | 0.016 |
| | Isolation Valve | 8 | 0.50 | | 0 |
| | Buffer Gas Tank | 1 | 250.00 | 0.6 | 0 |
| | Flow Controller | 4 | 0.50 | | 0 |
| $O_2$ Generation Subsystem | | 2 | 15.00 | 0.06 | 0.62 |
| | Solid Oxide Electrolysis Stack | 2 | 5.10 | 0.03 | 0.62 |
| | Isolation Valve | 8 | 0.50 | 0.008 | 0 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Check Valve | 4 | 0.10 | | 0 |
| Liquefaction Subsystem | | 1 | 13.20 | 0.09 | 1.03 |
| | $H_2$ Cooler | 2 | 10.60 | 0.01 | 0.34 |
| | $O_2$ Cryocooler | 2 | 1.30 | 0.04 | 0.69 |

**Table 6-8.** Solid Oxide Electrolysis for ECLSS Oxygen Operating at 24 Hours per Day

| | Solid Oxide Electrolysis 24-hour Production |
|---|---|

| System Element/Description | Subcomponents | Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
|---|---|---|---|---|---|
| **ISRU System** | | | **305.92** | **0.71** | **2.18** |
| Atmospheric Acquisition Subsystem | | 2 | 284.92 | 0.61 | 1.371 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Microchannel $CO_2$ Adsorption Pump | 4 | 5.70 | 0.001 | 1.368 |
| | Check Valve | 8 | 0.10 | | 0 |
| | Buffer Gas Pump | 4 | 1.23 | 0.002 | 0.003 |
| | Isolation Valve | 8 | 0.50 | | 0 |
| | Buffer Gas Tank | 1 | 250.00 | 0.6 | 0 |
| | Flow Controller | 4 | 0.50 | | 0 |
| $O_2$ Generation Subsystem | | 2 | 9.40 | 0.04 | 0.21 |
| | Solid Oxide Electrolysis Stack | 2 | 2.30 | 0.02 | 0.21 |
| | Isolation Valve | 8 | 0.50 | 0.008 | 0 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Check Valve | 4 | 0.10 | | 0 |
| Liquefaction Subsystem | | 1 | 11.60 | 0.06 | 0.60 |
| | $H_2$ Cooler | 2 | 10.60 | 0.01 | 0.34 |
| | $O_2$ Cryocooler | 2 | 0.50 | 0.02 | 0.26 |

**Table 6-9.** Solid Oxide Electrolysis for ECLSS and Ascent Oxygen Operating at 24 Hours per Day

| | | Solid Oxide Electrolysis 24-hour Production | | | |
|---|---|---|---|---|---|
| **System Element/Description** | **Subcomponents** | **Quantity** | **Mass (kg)** | **Volume ($m^3$)** | **Power (kW)** |
| **ISRU System** | | | **554.92** | **0.84** | **23.69** |
| Atmospheric Acquisition Subsystem | | 2 | 492.12 | 0.66 | 17.863 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Microchannel $CO_2$ Adsorption Pump | 4 | 57.50 | 0.014 | 17.86 |
| | Check Valve | 8 | 0.10 | | 0 |
| | Buffer Gas Pump | 4 | 1.23 | 0.002 | 0.003 |
| | Isolation Valve | 8 | 0.50 | | 0 |
| | Buffer Gas Tank | 1 | 250.00 | 0.6 | 0 |
| | Flow Controller | 4 | 0.50 | | 0 |
| $O_2$ Generation Subsystem | | 2 | 38.80 | 0.11 | 2.59 |
| | Solid Oxide Electrolysis Stack | 2 | 17.00 | 0.05 | 2.59 |
| | Isolation Valve | 8 | 0.50 | 0.008 | 0 |
| | Filter/Frit | 4 | 0.10 | | 0 |
| | Check Valve | 4 | 0.10 | | 0 |
| Liquefaction Subsystem | | 1 | 24.00 | 0.07 | 3.24 |
| | $H_2$ Cooler | 2 | 10.60 | 0.01 | 0.34 |
| | $CH_4$ Cooler | 2 | 1.20 | 0.01 | 0.02 |
| | $O_2$ Cryocooler | 2 | 5.50 | 0.03 | 2.90 |

#### *6.2.6.2 Excavation subsystems*

**Approach**

The excavation system model was developed to perform studies on the lunar surface. These studies involved excavation and transportation of regolith to a processing plant. A vehicle is sized based on different excavation rates to the plant. Vehicle size is also determined by a variety of other inputs to the model that describe the vehicle and the excavation parameters. The model was developed for lunar studies but was modified for Mars studies by inputting properties for the soil and the environment on Mars.

The excavation system model was written in Excel with the input and output being performed in the spreadsheet and all calculations coded within Visual Basic. The excavation system model consists of separate force and mass modules. The force module determines the vehicle forces as well as vehicle and bucket dimensions. The mass module performs structural calculations to size the subsystem components on the vehicle (dump bin and digging tool). The mass module calculates the platform mass based on the regolith mass, excavator mass, and subsystem component mass. An iterative process between the force and the mass modules is performed until a solution is reached that satisfies the conditions in both modules. A flowchart of this process is shown in figure 6-16.

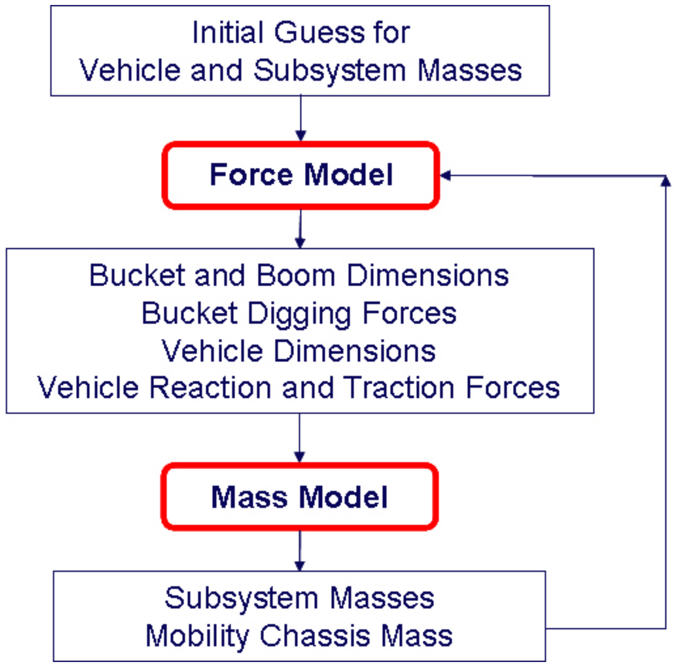



**Figure 6-16.** Excavation system flowchart.

**Results**

Mass, power, and volume results are presented in table 6-10 for a vehicle that is required to excavate a quantity of soil that is necessary to produce the quantities of water that are listed. For redundancy, two vehicles will be available to perform the task.

**Table 6-10.** Required Excavation System Summary

| System Element/Description | Subcomponents | 3% water – 8-hour ops – 2,146 kg $H_2O$ Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
|---|---|---|---|---|---|
| | | Solid Oxide Electrolysis 8-hour Production | | | |
| **Excavation Subsystem** | | 2 | 426 | 2.23 | 0.50 |
| | Excavator | 1 | 213 | 1.12 | 0.50 |
| | Dump Bin Mass | 1 | 8 | | |
| | Digging Tool Mass | 1 | 18 | | |
| | Platform Mass | 1 | 186 | | |

| System Element/Description | Subcomponents | 3% water – 24-hour ops – 2,146 kg $H_2O$ Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
|---|---|---|---|---|---|
| | | Solid Oxide Electrolysis 24-hour Production | | | |
| **Excavation Subsystem** | | 2 | 350 | 1.53 | 0.40 |
| | Excavator | 1 | 175 | 0.76 | 0.40 |
| | Dump Bin Mass | 1 | 3 | | |
| | Digging Tool Mass | 1 | 17 | | |
| | Platform Mass | 1 | 154 | | |

| System Element/Description | Subcomponents | 3% water – 24-hour ops – 16,788 kg $H_2O$ Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
|---|---|---|---|---|---|
| | | SWE 24-hour Production | | | |
| **Excavation Subsystem** | | 2 | 985 | 11.49 | 1.39 |
| | Excavator | 1 | 493 | 5.74 | 1.39 |
| | Dump Bin Mass | 1 | 46 | | |
| | Digging Tool Mass | 1 | 36 | | |
| | Platform Mass | 1 | 410 | | |

| System Element/Description | Subcomponents | 3% water – 24-hour ops – 16,788 kg $H_2O$ Quantity | Mass (kg) | Volume ($m^3$) | Power (kW) |
|---|---|---|---|---|---|
| | | SWE 24-hour Production | | | |
| **Excavation Subsystem** | | 2 | 587 | 4.25 | 0.73 |
| | Excavator | 1 | 293 | 2.12 | 0.73 |
| | Dump Bin Mass | 1 | 17 | | |
| | Digging Tool Mass | 1 | 22 | | |
| | Platform Mass | 1 | 254 | | |

#### *6.2.6.3 Water extraction subsystems*

The $H_2O$ extraction subsystem is used to extract adsorbed $H_2O$ from the martian soil. The subsystem consists of an inlet/ outlet hopper, an inlet/outlet auger, a reactor, gas clean up, and a condenser. A simple diagram of the subsystem is shown in figure 6-17. The inlet/outlet hopper and auger are used to receive soil from the excavator/hauler vehicle and transfer soil in and out of the reactor, respectively. Once the soil is in the reactor, it is heated to approximately 600 K. An inert gas flow fluidizes the soil to aid desorption of water. The inert- $H_2O$ gas stream is sent to a gas clean-up process to remove any contaminants that evolved during the process. The $H_2O$ is then collected on a condenser, which is actively cooled by a cryocooler.

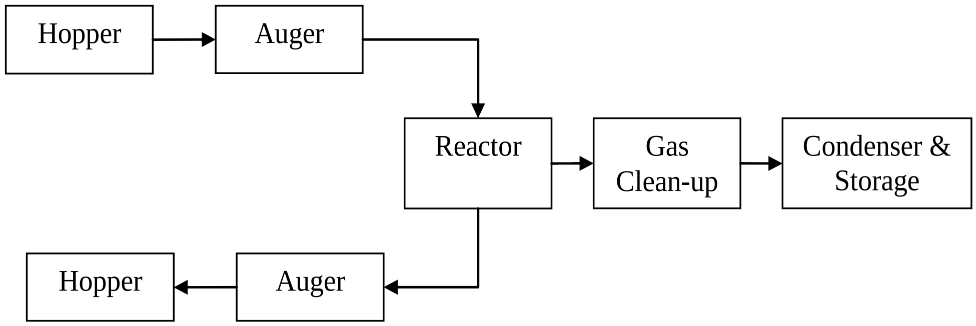



**Figure 6-17.** Diagram of the $H_2O$ extraction subsystem.

**Approach**
These models had been developed for lunar ISRU, and have been somewhat modified for Mars ISRU. The following is a brief description of the models that were used during the $H_2O$ extraction subsystem simulation:

- Hooper model
  - Simple cylindrical shape with conical exit
  - Mass and volume are determined by maximum amount of soil stored
- Auger model
  - Calculates time and power required to feed new batch of soil into reactor and remove spent soil out of the reactor
  - Can vary hub radius, flight radius, angle and total length of flights, rotational velocity, and fill ratio
  - Power, mass, and volume are estimated based on frictional forces (function of rotational velocity, fill ratio, wall friction, soil shear strength properties)
- Reactor model
  - Fluidized bed approach
  - Includes cyclones to separate gas from soil particles
  - Calculates inert flow rate that is required to fluidize soil based on reactor diameter, particle size distribution, and total mass of soil
  - Calculates amount of water extracted and time of reaction
  - Mass and volume are estimated based on reactor dimensions and pressure inside the reactor
  - Power to heat up the soil is estimated using lunar soil heat capacity data
- Gas clean-up model
  - Uses packed bed techniques to remove impurities from the reactor gas stream
  - Assumes two parallel beds: one is operating while the other is being regenerated, using some of the O2 that is produced
  - Calculates mass of adsorbent and structure as well as initial start-up power
- Condenser model
  - Calculate temperature and pressure of streams based on thermodynamics, flow rates of separated streams, and mixed streams
  - Mass based on flow rates, volumes, maximum operating pressure, and material properties (density, yield strength)
  - Active cooling provided by cryocooler

The simulation is initiated by providing the following inputs: amount of $H_2O$ that is required, time for processing, and $H_2O$ content in the soil. Then, the amount of soil per batch is calculated and sent to hoppers, auger, and reactor. The reactor provides a total gas flow and $H_2O$ extracted in the form of a flow rate output, which is then passed to the gas clean-up and then to the condenser. The reactor, gas clean-up, and condenser are sized based on flow rate and composition of the gas stream that is entering and exiting each component model.

## Results

The $H_2O$ extraction subsystem simulation provided mass, power, and volume estimates for total $H_2O$ production of 2,146 kg, 3,586 kg, and 16,788 kg within 300 days, with operation time of 8 or 24 hours/day, and a soil water content of 3% or 8% by weight. The result for each option is shown in tables 6-11, 6-12, 6-13, and 6-14.

**Table 6-11.** Summary of $H_2O$ Extraction System Assuming 3% Soil Water Content and 24-hr Operations

| | | **3% $H_2O$ – 24-hour ops – 2,146 kg** | | | | **3% $H_2O$ – 24-hour ops – 3,586 kg** | | | | **3% $H_2O$ – 24-hour ops – 16,788 kg** | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **System Element/Description** | **Subcomponents** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** |
| **$H_2O$ extraction System** | | **1** | **149.51** | **1.51** | **1.41** | **1** | **181.58** | **2.34** | **1.47** | **1** | **413.14** | **9.92** | **2.02** |
| | | | | | | | | | | | | | |
| Reactor Subsystem | | 1 | 88.75 | 1.44 | 0.09 | 1 | 118.14 | 2.26 | 0.09 | 1 | 325.22 | 9.82 | 0.09 |
| | Reactor Assembly | 1 | 88.75 | 1.44 | 0.09 | 1 | 118.14 | 2.26 | 0.09 | 1 | 325.22 | 9.82 | 0.09 |
| | Supply hopper | 1 | 25.85 | 0.53 | | 1 | 36.39 | 0.89 | | 1 | 101.79 | 4.15 | |
| | Feed auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Reactor Structure | 1 | 5.09 | 0.14 | | 1 | 8.50 | 0.24 | | 1 | 39.81 | 1.09 | |
| | Heater/thermal heat exchanger | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 |
| | Dust removal (cyclone) | 1 | 0.06 | 0.001 | | 1 | 0.06 | 0.001 | | 1 | 0.06 | 0.001 | |
| | Dump auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Dump hopper | 1 | 25.85 | 0.53 | | 1 | 36.39 | 0.89 | | 1 | 101.79 | 4.15 | |
| | Gate valves (2 inlet, 2 outlet) | 4 | 4.00 | | | 4 | 4.00 | | | 4 | 4.00 | | |
| | Insulation | 1 | 8.22 | 0.03 | | 1 | 13.12 | 0.05 | | 1 | 58.08 | 0.24 | |
| Gas clean-up Subsystem | | 1 | 25.23 | 0.009 | 0.00 | 1 | 26.22 | 0.010 | 0.00 | 1 | 35.32 | 0.013 | 0.00 |
| | Desulphurization Unit | 2 | 12.62 | 0.005 | | 2 | 13.11 | 0.005 | | 2 | 17.66 | 0.007 | |
| | Chamber/structure | 1 | 0.34 | 0.004 | 0.20 | 1 | 0.35 | 0.005 | 0.21 | 1 | 0.44 | 0.006 | 0.28 |
| | Adsorbent | 1 | 12.20 | | | 1 | 12.68 | | | 1 | 17.12 | | |
| | Insulation | 1 | 0.08 | 0.0003 | | 1 | 0.08 | 0.0003 | | 1 | 0.10 | 0.0004 | |
| Compressor, reactor loop | | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 |
| Condensers/Cryocooler Subsystem | | 1 | 35.45 | 0.06 | 1.24 | 1 | 37.13 | 0.06 | 1.30 | 1 | 52.54 | 0.09 | 1.86 |

**Table 6-12.** Summary of $H_2O$ Extraction System Assuming 8% Soil Water Content and 24-hr Operations

| | | **8% $H_2O$ – 24-hour ops – 2,146 kg** | | | | **8% $H_2O$ – 24-hour ops – 3,586 kg** | | | | **8% $H_2O$ – 24-hour ops – 16,788 kg** | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **System Element/Description** | **Subcomponents** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** |
| **$H_2O$ extraction System** | | **1** | **116.22** | **0.74** | **1.39** | **1** | **132.44** | **1.05** | **1.44** | **1** | **248.71** | **3.91** | **1.88** |
| | | | | | | | | | | | | | |
| Reactor Subsystem | | 1 | 56.24 | 0.67 | 0.09 | 1 | 70.31 | 0.98 | 0.09 | 1 | 166.87 | 3.81 | 0.09 |
| | Reactor Assembly | 1 | 56.24 | 0.67 | 0.09 | 1 | 70.31 | 0.98 | 0.09 | 1 | 166.87 | 3.81 | 0.09 |
| | Supply hopper | 1 | 13.47 | 0.20 | | 1 | 18.94 | 0.33 | | 1 | 52.92 | 1.55 | |
| | Feed auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Reactor Structure | 1 | 1.91 | 0.06 | | 1 | 3.19 | 0.09 | | 1 | 14.93 | 0.41 | |
| | Heater/thermal heat exchanger | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 |
| | Dust removal (cyclone) | 1 | 0.06 | 0.001 | | 1 | 0.06 | 0.001 | | 1 | 0.06 | 0.001 | |
| | Dump auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Dump hopper | 1 | 13.47 | 0.20 | | 1 | 18.94 | 0.33 | | 1 | 52.92 | 1.55 | |
| | Gate valves (2 inlet, 2 outlet) | 4 | 4.00 | | | 4 | 4.00 | | | 4 | 4.00 | | |
| | Insulation | 1 | 3.65 | 0.02 | | 1 | 5.49 | 0.02 | | 1 | 22.35 | 0.09 | |
| Gas clean-up Subsystem | | 1 | 24.94 | 0.009 | 0.00 | 1 | 25.74 | 0.010 | 0.00 | 1 | 33.07 | 0.012 | 0.00 |
| | Desulphurization Unit | 2 | 12.47 | 0.005 | | 2 | 12.87 | 0.005 | | 2 | 16.54 | 0.006 | |
| | Chamber/structure | 1 | 0.33 | 0.004 | 0.20 | 1 | 0.34 | 0.004 | 0.20 | 1 | 0.41 | 0.006 | 0.26 |
| | Adsorbent | 1 | 12.06 | | | 1 | 12.45 | | | 1 | 16.03 | | |
| | Insulation | 1 | 0.08 | 0.0003 | | 1 | 0.08 | 0.0003 | | 1 | 0.10 | 0.0004 | |
| Compressor, reactor loop | | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 |
| Condensers/Cryocooler Subsystem | | 1 | 34.96 | 0.06 | 1.22 | 1 | 36.31 | 0.06 | 1.27 | 1 | 48.70 | 0.08 | 1.72 |

**Table 6-13.** Summary of $H_2O$ Extraction System Assuming 3% Soil Water Content and 8-hr Operations

| | | 3% $H_2O$ – 8-hour ops – 2,146 kg | | | | 3% $H_2O$ – 8-hour ops – 3,586 kg | | | | 3% $H_2O$ – 8-hour ops – 16,788 kg | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **System Element/Description** | **Subcomponents** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** |
| **$H_2O$ extraction System** | | **1** | **182.27** | **1.85** | **1.59** | **1** | **236.32** | **2.91** | **1.77** | **1** | **669.25** | **12.52** | **3.43** |
| | | | | | | | | | | | | | |
| $H_2$ Reduction Reactor Subsystem | | 1 | 113.55 | 1.77 | 0.09 | 1 | 159.57 | 2.83 | 0.09 | 1 | 519.18 | 12.45 | 0.09 |
| | Reactor Assembly | 1 | 113.55 | 1.77 | 0.09 | 1 | 159.57 | 2.83 | 0.09 | 1 | 519.18 | 12.45 | 0.09 |
| | Supply hopper | 1 | 25.85 | 0.53 | | 1 | 36.39 | 0.89 | | 1 | 101.79 | 4.15 | |
| | Feed auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Reactor Structure | 1 | 15.27 | 0.42 | | 1 | 25.51 | 0.70 | | 1 | 119.42 | 3.25 | |
| | Heater/thermal heat exchanger | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 |
| | Dust removal (cyclone) | 1 | 0.06 | 0.001 | | 1 | 0.06 | 0.001 | | 1 | 0.08 | 0.001 | |
| | Dump auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Dump hopper | 1 | 25.85 | 0.53 | | 1 | 36.39 | 0.89 | | 1 | 101.79 | 4.15 | |
| | Gate valves (2 inlet, 2 outlet) | 4 | 4.00 | | | 4 | 4.00 | | | 4 | 4.00 | | |
| | Insulation | 1 | 22.83 | 0.09 | | 1 | 37.54 | 0.16 | | 1 | 172.41 | 0.71 | |
| Gas clean-up Subsystem | | 1 | 28.19 | 0.010 | 0.00 | 1 | 31.16 | 0.012 | 0.00 | 1 | 58.39 | 0.021 | 0.00 |
| | Desulphurization Unit | 2 | 14.09 | 0.005 | | 2 | 15.58 | 0.006 | | 2 | 29.20 | 0.011 | |
| | Chamber/structure | 1 | 0.37 | 0.005 | 0.22 | 1 | 0.040 | 0.005 | 0.25 | 1 | 0.65 | 0.010 | 0.46 |
| | Adsorbent | 1 | 13.64 | | | 1 | 15.10 | | | 1 | 28.41 | | |
| | Insulation | 1 | 0.09 | 0.0004 | | 1 | 0.09 | 0.0004 | | 1 | 0.14 | 0.0006 | |
| Compressor, reactor loop | | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 |
| Condensers/Cryocooler Subsystem | | 1 | 40.46 | 0.07 | 1.42 | 1 | 45.51 | 0.08 | 1.60 | 1 | 91.61 | 0.14 | 3.27 |

**Table 6-14.** Summary of $H_2O$ Extraction System Assuming 8% Soil Water Content and 8-hr Operations

| | | 8% $H_2O$ – 8-hour ops – 2,146 kg | | | | 8% $H_2O$ – 8-hour ops – 3,586 kg | | | | 8% $H_2O$ – 8-hour ops – 16,788 kg | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| **System Element/Description** | **Subcomponents** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** | **Quantity** | **Mass (kg)** | **Vol. ($m^3$)** | **Power (kW)** |
| **$H_2O$ extraction System** | | **1** | **131.93** | **0.87** | **1.53** | **1** | **158.69** | **1.27** | **1.68** | **1** | **371.48** | **4.95** | **3.02** |
| | | | | | | | | | | | | | |
| $H_2$ Reduction Reactor Subsystem | | 1 | 65.54 | 0.80 | 0.09 | 1 | 85.85 | 1.19 | 0.09 | 1 | 239.60 | 4.80 | 0.09 |
| | Reactor Assembly | 1 | 65.54 | 0.80 | 0.09 | 1 | 85.85 | 1.19 | 0.09 | 1 | 239.60 | 4.80 | 0.09 |
| | Supply hopper | 1 | 13.47 | 0.20 | | 1 | 18.94 | 0.33 | | 1 | 52.92 | 1.55 | |
| | Feed auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Reactor Structure | 1 | 5.72 | 0.16 | | 1 | 9.57 | 0.26 | | 1 | 44.78 | 1.22 | |
| | Heater/thermal heat exchanger | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 | 0 | 0.00 | | 0.00 |
| | Dust removal (cyclone) | 1 | 0.06 | 0.001 | | 1 | 0.06 | 0.001 | | 1 | 0.07 | 0.001 | |
| | Dump auger | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 | 1 | 3.84 | 0.10 | 0.05 |
| | Dump hopper | 1 | 13.47 | 0.20 | | 1 | 18.94 | 0.33 | | 1 | 52.92 | 1.55 | |
| | Gate valves (2 inlet, 2 outlet) | 4 | 4.00 | | | 4 | 4.00 | | | 4 | 4.00 | | |
| | Insulation | 1 | 9.13 | 0.04 | | 1 | 14.65 | 0.06 | | 1 | 65.22 | 0.27 | |
| Gas clean-up Subsystem | | 1 | 27.33 | 0.010 | 0.00 | 1 | 29.73 | 0.011 | 0.00 | 1 | 51.68 | 0.019 | 0.00 |
| | Desulphurization Unit | 2 | 13.66 | 0.005 | | 2 | 14.86 | 0.006 | | 2 | 25.84 | 0.010 | |
| | Chamber/structure | 1 | 0.36 | 0.005 | 0.22 | 1 | 0.38 | 0.005 | 0.24 | 1 | 0.59 | 0.009 | 0.41 |
| | Adsorbent | 1 | 13.22 | | | 1 | 14.39 | | | 1 | 25.12 | | |
| | Insulation | 1 | 0.08 | 0.0004 | | 1 | 0.09 | 0.0004 | | 1 | 0.13 | 0.0005 | |
| Compressor, reactor loop | | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 | 1 | 0.08 | | 0.08 |
| Condensers/Cryocooler Subsystem | | 1 | 38.99 | 0.07 | 1.37 | 1 | 43.04 | 0.07 | 1.51 | 1 | 80.12 | 0.13 | 2.85 |

### 6.2.7 Exploratory transportation and campaign architecture considerations

#### *6.2.7.1 Objectives and approach for campaign architectures analyses*

This study was intended to investigate possible ISRU architectures other than those that are included in the ISRU decision package. Involved were the following two interrelated tasks engaging the use of the Mars ISRU Architecture (MIA) Workbook that was developed at the NASA Jet Propulsion Laboratory (JPL):

- Developing and analyzing a variety of ISRU transportation architectures
- Analyzing a subset of ISRU transportation architectures in campaign scenarios

The objective for the first task was to identify and characterize a range of crewed Mars transportation architectures that exhibits a variety of levels of dependency on ISRU. The effects of varying ISRU systems and applications were examined as compared to the baseline architecture. A wide range of transportation architectures was investigated, from a basic case-like producing propellant for fueling the ascent vehicle on the surface of Mars to more exploratory cases such as using orbital depots with propellant tankers or placing an ISRU plant on Phobos. Propellant options were also varied ($LO_2/H_2$, NTP, $LO_2/CH_4$) as well as aerocapture/aerobraking options. The objective of the second task was to identify transportation architectures from the first task that would likely have an additional mass savings benefit when used in a campaign format and then use the MIA Workbook to complete analysis of the transportation architecture as a campaign. A campaign is defined as a set of three missions to a single site on Mars.

The general approach was to first identify ISRU-friendly martian transportation architectures and model each candidate architecture in the MIA Workbook. The FOMs, which were then calculated and evaluated for each architecture, consist of:

- *Delta MAV/MXH/MH/IMLEO Mass*: The change in the mass of the LEO stack between the baseline and the current architecture
- *Delta launches required per mission*: The change in the number of launches that is needed by the current architecture and the baseline
- Delta IMLEO mass for overall campaign architecture (three-mission set)

#### *6.2.7.2 Mars in-situ resource utilization architecture model description*

The MIA Workbook was produced by using MassTracker, a tool that was developed at JPL and that was designed to quickly model multiple architecture cases and give detailed mass information throughout the resulting architectures. Both standard and "exploratory" ISRU architectures were developed using this tool and fed into the MIA Workbook. The model was then used to compare the IMLEO mass and the number of launches that are required for each of these architectures.

#### *6.2.7.3 Transportation architecture analysis results*

The results of the single mission transportation architectures from the MIA workbook are shown in figure 6-18. From the figure it is apparent that the architecture requiring the least IMLEO mass to perform a single mission is the NTP version of surface ISRU fueling for ascent only. None of the chemical options provides a notable mass savings over the baseline (not enough to save a launch) in a single-mission scenario. The options for Phobos ISRU and the orbital depot (sized only for descent) architectures produce similar results to the simplest case of ISRU (propellant for ascent). Each of these architectures shows a mass savings over the baseline for a single mission, but it is unclear which one of the architectures would provide the most mass savings without a more detailed study of all three architectures.

In addition to looking at the transportation architectures, several other trades were preformed including the martian parking orbit (elliptical vs. circular) and using a pump-fed vs. a pressure-fed launch system.

#### *6.2.7.4 Assumptions for campaign analysis*

Several additional assumptions were made when completing the campaign analysis. Subsequent missions were assumed to land sufficiently close to the original site such that ISRU assets from the original mission could be used to fuel the following mission. Replacement parts would be brought on each of the subsequent missions to keep the ISRU system, the habitat, the power plant, etc. in good repair. Additionally, the MAVs were staged to decrease the amount of fuel that would be required to get to orbit.

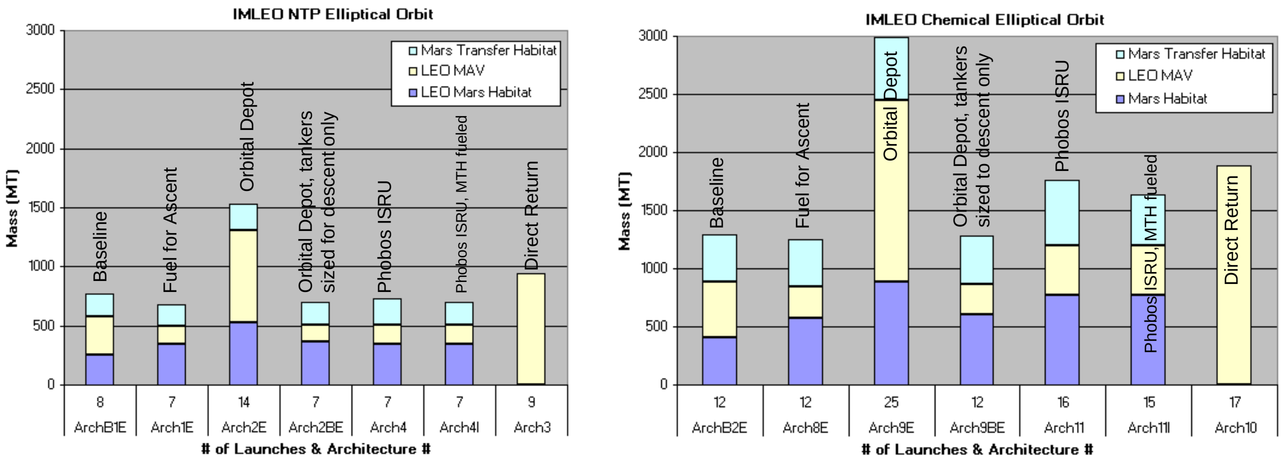



**Figure 6-18.** Selected results from analysis of ISRU transportation architectures for single missions.

The architectures that were analyzed in the campaigns (three missions) are as follows:

- A baseline model with no ISRU used as fuel (ArchB1E), a modest ISRU fueling approach where just the ascent stage is fueled on the surface of Mars (Arch1E)
- An ambitious approach in which the descent stages are turned into tankers that are used to fuel an orbiting depot and then to fuel the following descent stages (Arch2BE)
- Another ambitious approach in which an ISRU fueling station is set up on Phobos and used to fuel the descent stages and the TEI stages (Arch4I)
- A Mars direct return option was looked at as an ISRU comparison (Arch 3)

**Note:** Arch2BE, Arch4I, and Arch 3 all have fueling on the surface similar to Arch 1E.

#### *6.2.7.5 Campaign results and conclusions*
Figure 6-19 shows the results of the campaign analysis. Mass savings over one campaign to a single site range from approximately 475 t to approximately 492 t when comparing fueling ascent vehicle only, orbital depot, and Phobos ISRU architectures with non-ISRU baseline. The direct return option does not provide any mass savings over the baseline. The mass savings for those architectures that show improvement over the baseline are equivalent to five Ares V launches in all three cases. The mass savings that would be attained through the use of ISRU on Mars may be mission enabling.

#### *6.2.7.6 Future work*
Future work includes adding mobility to ISRU facility and habitat for moving to independent sites and investigation of architectures where hoppers and rovers are utilized. We would also like to add the ability to do analyses for multiple site campaigns (?). Additional FOMs that may be developed to enhance future analyses include:

- The probability for Loss of Mission (LOM) / Loss of Crew (LOC) over the various architectures
- Mission Life Cycle Cost

### *6.2.8 Conclusions and future work*
The work described in this section has significantly influenced the ISRU Decision Package. Based on this work and the current level of fidelity of the subsystems designed for the production of consumables, ISRU may be mission enabling. However, much work still needs to be done in model development and system testing before a complete assessment of ISRU on Mars can be attained.

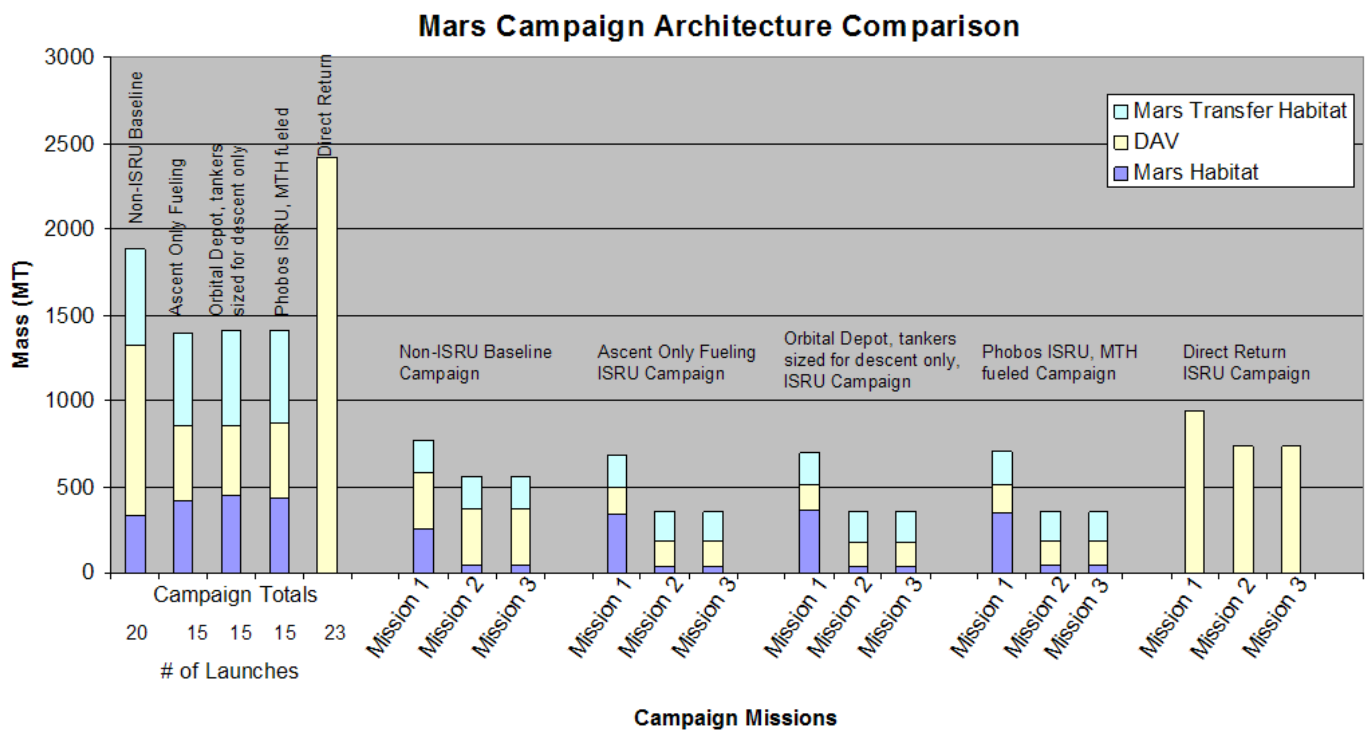



**Figure 6-19.** Results from transportation architectures analyzed as campaigns.

## *6.3 Surface Power Systems*

### 6.3.1 Power systems analysis

Architecture options Options 1, 2, and 3 were evaluated for their power requirements. Each of these architectures was then compared. Two power system technologies were considered as prime power sources: solar PVAs with energy storage, and nuclear fission. In addition to a prime power source, the architecture for Options 1 and 2 called for long-range crew mobility to expand the range of exploration beyond the immediate vicinity of the base, and power systems for these mobile systems were also analyzed.

### 6.3.2 Background: Mars environmental factors

The solar array design and sizing are heavily influenced by the martian dust environment, site location, and time of year. The effect of atmospheric dust on solar intensity at the surface is expressed in terms of the optical depth of the atmosphere. The total optical depth, tau ($\tau$), is a measure of the quantity of light that is removed from a beam by scattering ($\tau_s$) and absorption ($\tau_a$) from its path from the upper edge of the atmosphere to the planet surface. A tau of zero corresponds to no scattering or absorption; all of the incoming light reaches the surface. A significant amount of the sunlight is scattered by the dust; and of this, some reaches the surface while some is scattered back into space. Thus, although the direct solar intensity on the surface decreases with the amount of dust in the atmosphere, the actual intensity of illumination on the surface is a mixture of direct and scattered sunlight with a complicated dependence on the amount of dust in the atmosphere and the sun angle. Data from the MERs indicated that a nominal day on Mars has an optical depth of about 1.0 to 0.9 (Stella, et al, 2005[2])

Another important design consideration is the light that is blocked by dust and that settles on the array surface. Data from the Pathfinder rover Sojourner showed a 0.2% power loss per sol (1 sol = 1 martian day). The MERs also experienced a similar degradation rate. The rover Spirit had an estimated dust loss of approximately 30% by sol 300 and 40% by sol 400 (Landis, 2005[3]). Results of short-circuit solar cell tests on Opportunity confirm power losses due to dust accumulation. (Landis 2005) However, a major "clearing event" occurred on sol 418, restoring array power to 90% followed by a slow decline down to 70% after another 100 sols. The clearing event occurred when the rover was at a 22-degree tilt and atop a ridge, which seems

[2] Stella, Paul M., et al., Design and Performance of the MER (Mars Exploration Rovers) Solar Arrays, Proc. 31st IEEE Photovoltaic Specialists Conference, Jan. 3-7, Orlando, FL, 2005, pp. 626-630.
[3] Landis, G.A., Exploring Mars with Solar-Powered Rovers, Proc. 31st IEEE Photovoltaic Specialists Conference, Jan. 3-7, 2005, Orlando, FL, pp. 858-861

to suggest that in addition to increased wind speed the angle to the array surface plays some part in the cleaning effect. This surface feature may have contributed to a localized increase in wind speed or, possibly, turbulence that is not normally encountered in flatter terrains. It was felt that the rover tilt angle had a large part to play in the dust removal. Test results of past wind tunnel testing in a simulated Mars environment also showed greater dust removal at high "angles of attack" (Gaier 1990) [4].

A third major environmental concern is the frequency, duration, and severity of a Mars dust storm. Reviewing past observations by telescope, Viking and MER show dust storms to occur during northern hemisphere winter when Mars is closest to the sun in its orbit, at which time the temperature difference between the northern and southern hemispheres tends to be the greatest. These larger temperature differences create conditions for higher winds to occur and suspend fine surface dust in the atmosphere. The recent storm that occurred with the MERs has provided excellent data to observe PVA performance under varying levels of $\tau$. Figure 6-20 is a chart from Mark Lemmon, MER Science Team, of the daily values of $\tau$ during the 2007 dust storm (and compared to Viking observations). Correlating the dust data ($\tau$ = ~5) with reported array daily energy for Opportunity of 128 W-hr on July 17, 2007 and the early mission "clear day" energy of approximately 900 W-hr, the array provided about 14% of the maximum possible average power during the worst part of the storm. As a note, both MERs had 8 to 1.0 W thermal, radioisotope heater units (RHUs) – six for the battery and two for electronics. The heat from the RHUs helped the rovers survive during the dust storm by keeping circuits warm and preventing the battery from freezing.

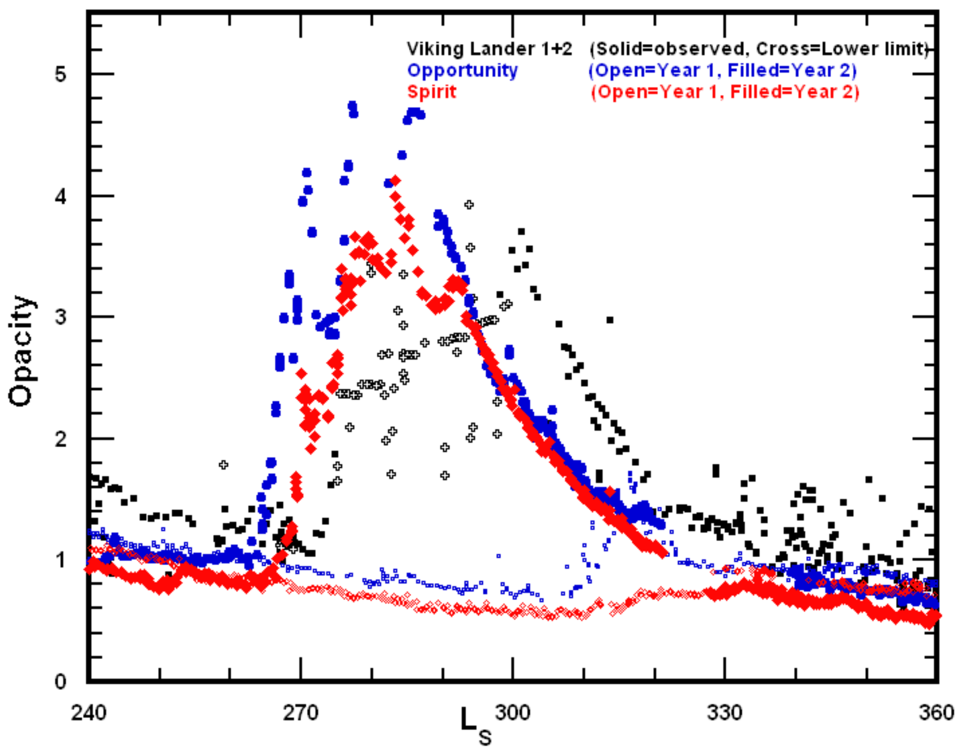



**Figure 6-20.** Opacity measurements for VL-1, VL-2, and the Mars exploration rover mission.

### 6.3.3 Solar power design human mission design

The current reference architecture calls for pre-deployment of mission assets via a cargo-only spacecraft prior to the crew Earth departure. Once the cargo vehicle has landed on the surface, the power system will be deployed and made operational to support the production of ascent propellants, habitat readiness, and other operations such as robotic rover recharging, maintaining logistics modules, and propellant maintenance. The power system is planned to be deployed and readied in 30–40 sols. Total production of the propellants and crew consumables $O_2$ cache must be completed prior to the crew departure. A total of 300 sols have been baselined to make $O_2$ for ascent vehicle propellant, in addition to a cache of $O_2$ for crew

[4] Gaier, J.R., et al., Aeolian Removal of Dust from Photovoltaic Surfaces on Mars. NASA TM-102507, 1990

consumables. This number is derived from the following: a time between the cargo launch and crew launch of approximately 760 sols, less approximately 310 sols for cargo vehicle trip time, less approximately 40 sols for power system setup, less approximately 50 sols dust storm and approximately 60 sols overall contingency.

The architecture that was established for our analysis is the long-stay option, where the crew will be on the surface for a duration from 500 to 550 days. The power system must operate continuously and reliably for more than 4 years. The solar power system must be designed to tolerate settled dust degradation and at least one dust storm. The MERs, with their low power needs and higher acceptable risks, easily resolved these issues by using oversized arrays and small amounts of isotope decay heat to keep from freezing. They also were able to tolerate the dust losses and still function at severely reduced performance until the right wind conditions cleared the arrays and, in addition, survived a significant dust storm.

The power system for a human mission is a mission-critical function. High reliability over the required lifetime will be accomplished by sufficient flight hardware testing in conjunction with component and system redundancy, as required.

The Mars environment poses a significant challenge to designing a solar powered system, as was previously discussed, particularly for the stringent reliability levels that are mandated by human missions. Table 6-15 shows the assumptions that were used for the analysis of the solar system design.

**Table 6-15.** Power System Design Guidelines

| | |
|---|---|
| Site Latitude | 30¼ North |
| Optical Depth Clear Day | 0.9–1.0 |
| Optical Depth Dust Storm | 4.0–5.0 |
| Length of Dust Storm | 50 sols |
| Nominal Dust Deposition Loss Rate | 0.20%/sol |
| Maximum Allowable Power Loss | 10% |
| Mission Duration | 550 sols |

Since dust accumulation on the arrays is a critical factor in sizing, it was assumed that some effective method of cleaning the arrays robotically every 40 to 50 sols will have been developed prior to a human mission, thus limiting loss due to settling dust to about 10% and keeping array areas more manageable. A robotic or automated method for cleaning the array is necessary because power is required during the pre-deploy phase prior to crew arrival. Due to the very large array area that would be required, significant power loss due to dust coverage, as that experienced with MER, is prohibitive and would not be practical to accommodate by over-sizing the array.

Latitude also becomes an important factor since the winter daylight period shortens as latitude increases. At 30º latitude, winter solstice daylight is about 10 hours duration with a 14.5-hour nighttime. At 60º latitude, the ratio is about 5 hours per day and 20 hours per night. This is problematic for solar power systems because the array area that is needed increases significantly to produce enough energy with less daylight and a longer nighttime period.

A brief description of each of the three mission scenarios evaluated is given below.

**OPTION 1 – MOBILE HOME**

In the Mobile Home scenario, the crew would live in two large, long-range rovers. These rovers would be required to provide all of the power that would be necessary to support the crew members during their stay, as well as providing the considerable energy that would be required for roving expeditions lasting up to 30 days, during which time the rovers would traverse as much as 200 km. No central habitat would be included in this scenario, although a central power supply that would be needed to support ISRU and other assets prior to crew arrival would be available to power the rovers at the landing site. It is assumed that the rovers would not be on a sortie during the dust storm season, and the rovers would be receiving power from the main system during this time.

**OPTION 2 – COMMUTER**

The Commuter scenario includes a central habitat in addition to two smaller, pressurized rovers. The central habitat would provide services to the full crew in between rover excursions, maintaining a minimum crew of two when both rovers are in the field. The rover sortie requirements were set at 100-km, round-trip distances accomplished in a 1-week period. As in the Mobile Home option, each pressurized rover carries its own power system, and Apollo-type rovers at the base would be recharged off the main power system. In this particular case, the crew has a safe haven to return to and does not have to rely solely on each rover power system for shelter and life support.

**OPTION 3 – TELECOMMUTER**

No pressurized rovers are included in the Telecommuter scenario. The habitat is included and power requirements are estimated to be the same as for the Commuter scenario, which was discussed above. This scenario also includes two long-range robotic rovers. These rovers are expected to be based on a Mars Science Lab-type design, and assumed to use their own dedicated RPSs in the low multi-kilowatt range.

Crew mobility will be limited to shorter distances from the habitat because only the Apollo LRV-type rovers are used. The power source for these rovers is assumed to be batteries or, possibly, fuel cells depending upon stored energy requirements. Range will be limited by suit power, rover speed, and permissible “walk-back” distance as with the Apollo mission. Speed for this rover is estimated at around 10 km/hour and in the 1–2 kW range. Rover distance might be extended if suit functions were powered off the rover rather than using the suit battery when the crew is driving the rover. Rover recharge was estimated at 1.5 kWe during daytime only, but night recharge could be considered. Daytime recharging might dictate additional rovers or spare batteries, with one being used while the spares are on charge.

### 6.3.4 Power requirements summary

The major power requirement is the production of $O_2$ for the ascent stage. The power level of 92 kWe for this ISRU phase is based on a production time of 8 hours/sol for 300 sols, and a requirement to supply 5 kWe during the night to maintain the propellants in a liquid form and keep the production plant equipment in a warm quiescent mode.

All three architecture options include robotic rovers to perform various tasks. In particular these would be used during the pre-deploy phase to move and set up equipment such as the ISRU plant, logistics module, and power system setup, and to perform power cable connections. It is anticipated that appropriate recharge stations would be either attached to the habitat or a power management distribution module for rover recharge. Details of the designs of these rovers were not assessed during this phase, but it was assumed that they would be battery-powered due to their short-range application. The ascent stage will also require power for “keep-alive” functions and propellant maintenance. An ISRU plant will produce $O_2$ for life support and EVA suit re-supply for all three options while $O_2$ production for ascent production remains an option.

Table 6-16 shows the power that is required for the various architecture elements for normal day and night operations and also for use during a dust storm. The habitat power estimate is scaled for Mars based on a monolithic habitat design for the lunar South Pole with a crew of four. It might be possible to reduce the night and dust storm habitat power by a reduction in the habitat power during a dust storm compared to normal operations power levels. This would make a significant difference in additional “dust storm” array area and the mass of an “all-fuel-cell” power option. The ISRU plant, which would be making ascent stage $O_2$ propellant, is clearly the dominant power requirement at 66 kWe operating nominally for 8 hours/day (22 kWe continuously). This strategy of limiting the operational time for the propellant production reduces the required array size. Any energy that is used at night has to be recharged during the day with additional power for electrochemical recharge inefficiency. For an efficiency of 50% and a 2:1 charge to discharge ratio (i.e., 8-hour charge/16-hour discharge), an array has to produce 25% more power for nighttime operation than if operated during daytime only.

**Table 6-16.** Estimated Power Requirements for Various Surface Elements

| Element | Day kWe | Night kWe | Dust Storm kWe | Notes |
|---|---|---|---|---|
| Habitat | 12 | 12 | 12 | |
| ISRU $O_2$ Propellant (Solar) | 66 | 2–3 | 2–3 | 8 hours/sol |
| ISRU $O_2$ Propellant (Nuclear) | 22 | 22 | 22 | |
| ISRU $O_2$ Consumables (Solar) | 5.7 | 0.5–1 | 0.5–1 | 8 hours/sol |
| ISRU $O_2$ Consumables (Nuclear) | 2 | 2 | 2 | |
| Logistics Module | 1.5 | 1.5 | 1.5 | Option 1 only |
| Ascent Stage | 1.5 | 1.5 | 1.5 | |
| Rover Recharge | 1.5 | 0 | 0 | |
| ISRU Crew $O_2$ Cache | 1.5 | 1.5 | 1.5 | Maintain only |
| Drill | 3 | 0 | 0 | Power from Rover |

Total rack-up of estimated power levels of the PV system option is shown in figure 6-21. Nominal total load power for the crew phase is approximately 20 kWe for both day and night operation. With ascent $O_2$ propellant production, the total daytime average power required is about 96 kWe. If only crew-consumable $O_2$ is produced, the total average day power is reduced to about 12 kWe. It should be noted that this is not peak power at noon delivered by the arrays but, rather, a time-averaged value.

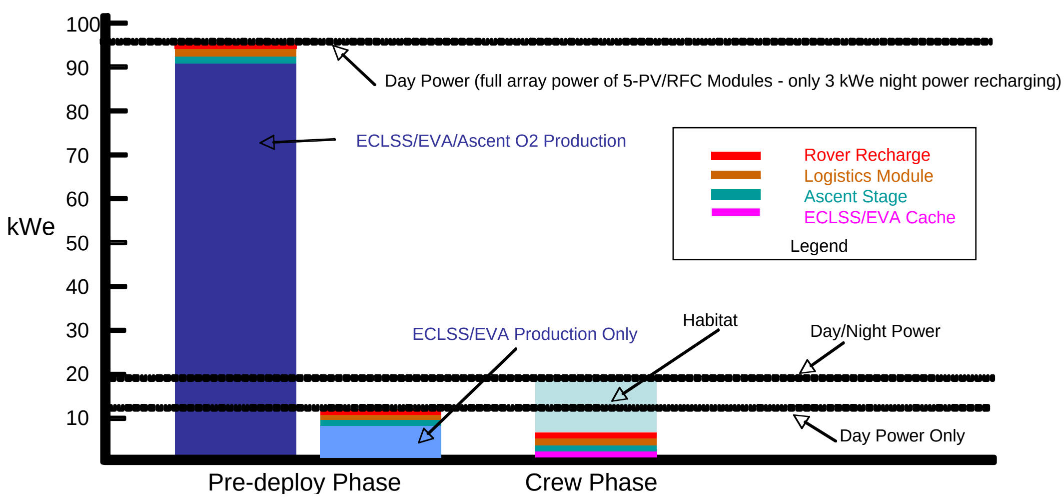



**Figure 6-21.** Power requirements based on the use of a solar power system.

Thus, in the consumable ISRU-only case, the habitat has the greatest power demand. A system sizing to meet this requirement will have ample power available for pre-deploy phase activities. However, if ISRU for ascent $O_2$ production were adopted, it would become the predominate load for power. The PV/regenerative fuel cell (RFC) system module size was selected at 5 kWe. As it turns out, if one additional unit is delivered, the total array area of all five modules is sufficient to provide the daytime power. In fact, the ratio of ascent ISRU power and habitat power level is such that the five PV/RFC modules could support continuous operations both day and night. The downside to doing this is that 2 years of RFC lifetime would be used, and additional electrolyzer and fuel cell stacks would be needed to maintain the reliability of the system during the crew phase. The impact of component lifetime and overall system reliability was not evaluated in this phase of the study.

As an option to the PV/RFC system, a nuclear power system can also be considered as the main power source for the base. Figure 6-22 shows the total power levels if nuclear power were the power technology that was chosen for the architecture.

Again, ISRU for ascent propellant production requires the greatest amount of power. Whereas the crew phase needs (mainly for the habitat) is the driver for the ISRU consumables case, because a nuclear power source produces power continuously without the need for energy storage, the peak power that is required is significantly reduced when compared with the solar/RFC system. For an 8-hour daylight/16-hour night period, the solar system produced a peak power output of 90 kWe while the nuclear system produced around 30 kWe. An item of note here is that the sizing for the nuclear system is valid at all latitudes while the solar case sized for this study is valid at 30ºN latitude or the equator.

Details of both systems will be given in the power system technology section.

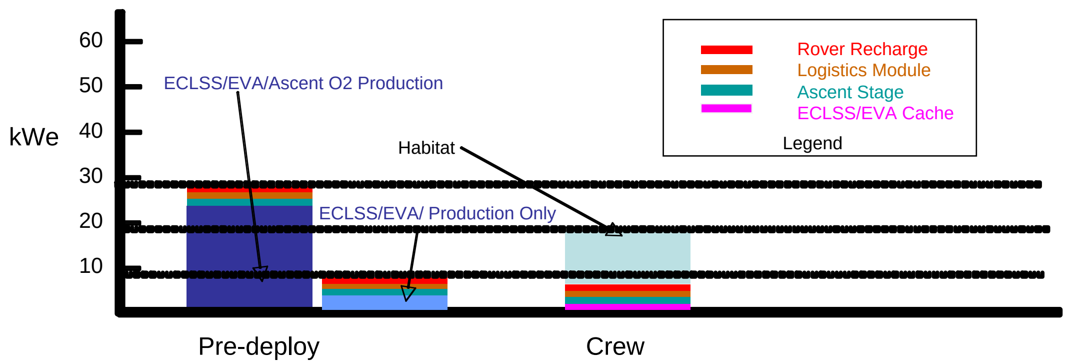



**Figure 6-22.** Power requirements based on the use of a nuclear power system.

Power system masses were estimated for both solar and nuclear power systems for each architecture option. The architecture options included base and habitat power with consumable $O_2$ production only and base and added power for ascent vehicle $O_2$ production. The results of these mass estimates are shown in table 6-17.

**Table 6-17.** Estimated Total Power System Masses

| Power System Mass Summary (t) | | | |
|---|---|---|---|
| | Architecture Study Options | | |
| | Mobile Home | Commuter | Telecommuter |
| Power System | | | |
| Solar – no ascent propellant | 3.5 | 22.5 | 22.5 |
| Solar – ascent propellant | 19.5 | 22.5 | 22.5 |
| Nuclear – no ascent propellant | 5.5 | 6.8 | 6.8 |
| Nuclear – ascent propellant | 7.8 | 7.8 | 7.8 |

- Includes 20% mass margin
- Solar mass includes dust storm array, no DIPS
- Commuter and Telecommuter – five 5/5-kWe PV/RFC modules – fifth unit to accommodate ascent $O_2$ propellant product day power
- Mobile Home option – custom-sized solar modules to accommodate higher day power
- Nuclear system mass is for a single unit
- Power system mass for rovers not included

The solar power system masses include an additional 8,000-kg mass for an additional array, which would be deployed in the event of a major dust storm. The array area required during a dust storm is approximately 4,300 $m^2$ (29% eff. cells) in addition to the array area of five PV/RFC modules. It is envisioned that the crew at the start of a dust storm would roll out the thin-film array. The arrays could be spooled on 8.5-m-wide by 100-m-long sections, in which case approximately five spools would be required. Since each spool would be about 1,500 kg, the spools could be emplaced with aide of the rovers that would be readied for future deployment. An all-fuel-cell option to supplement the power loss during the dust storm was assessed to supply the required energy, but it was two to three times heavier than the rollout array option.

This particular architecture calls for only one visit by a crew, and subsequent missions would be at another location. This means that all of the power system assets are only used once and require a lifetime of 4 years. A different power system strategy – i.e., technology selection, system sizing, back-up emergency power system selection, etc. – might be chosen based on a multiple-visit scenario with greater power level and an increased lifetime requirement.

The final five-module configuration of the solar power system is shown in figure 6-23. Each module consists of a 5-kWe RFC for nighttime power production and a PV array with 29% efficient solar cells with an area of 290 $m^2$ for both wings. The array panels are inclined 30 degrees to optimize the overall power profile by increasing output during early morning and late afternoon and reducing peak power at noon. Dimensions of the module are 1.5 m $\times$ 2.0 m $\times$ 3.0 m. Each array wing is 2.5 m high $\times$ 58 m long. Total capability of the five units is 25 kWe nighttime load and 25 kWe day power load to loads plus RFC recharge power. It is anticipated that each module would be offloaded from the cargo lander and set in place by robotic rovers. A robotic rover, which would be tele-operated from earth, assists deployment of the array wings. Support legs drop down and lock in place as the wing is pulled out. The array has a 0.5-m clearance off the ground, so a fairly flat area is needed since the total array span from end-to-end is almost 120 m. The array deployment system concept was not assessed in great detail, and has been identified as an area that needs future in-depth design study.

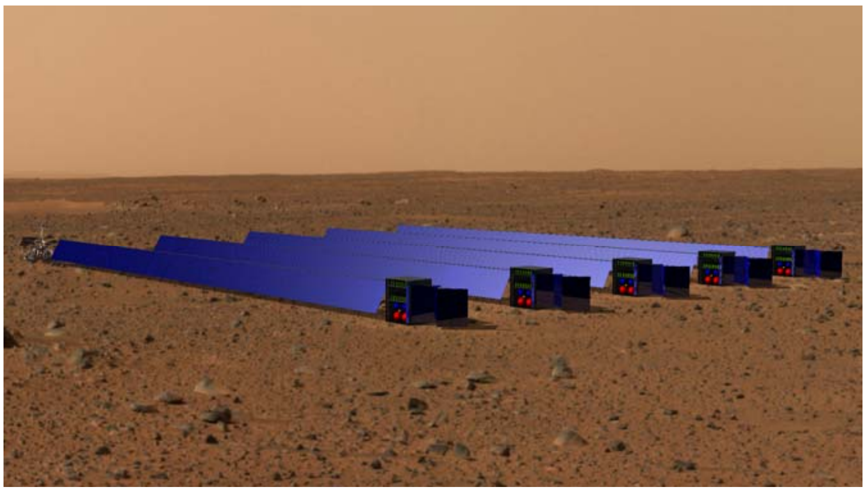


**Figure 6-23.** Solar photovoltaic/regenerative fuel cell system.

The nuclear power reactor concept that was used for this study is based on a lunar design that is capable of operating on the martian surface (Mason, et al, 2008) [5]. The low operating temperature of the reactor fuel enables the use of stainless steel, a material that is compatible with the predominately $CO_2$ atmosphere of Mars. The mass of the nuclear power system that was used for comparison was for a 30-kWe version of this design. The image in figure 6-24 shows the reactor in a stowed configuration as offloaded from the cargo bay and ready for emplacement with external power taken from a utility power cart that would have multiple functions. The power cart could be PV/RFC, battery powered, or powered by an RPS. For this study, it was assumed that a DIPS would be used for the power cart and could also be an option for powering the pressurized rovers. Plutonium-238 ($^{238}$Pu) isotope, which has fueled numerous deep space missions as well as Apollo and Viking, would be used with advanced power conversion technology to increase power output 3- to 4-fold as compared with thermoelectric devices that are currently used.

[5] Mason, Lee, et al., System Concepts for Affordable Fission Surface Power, 25th Symposium on Space Nuclear Power and Propulsion, Feb. 10-14, 2008, Albuquerque, NM

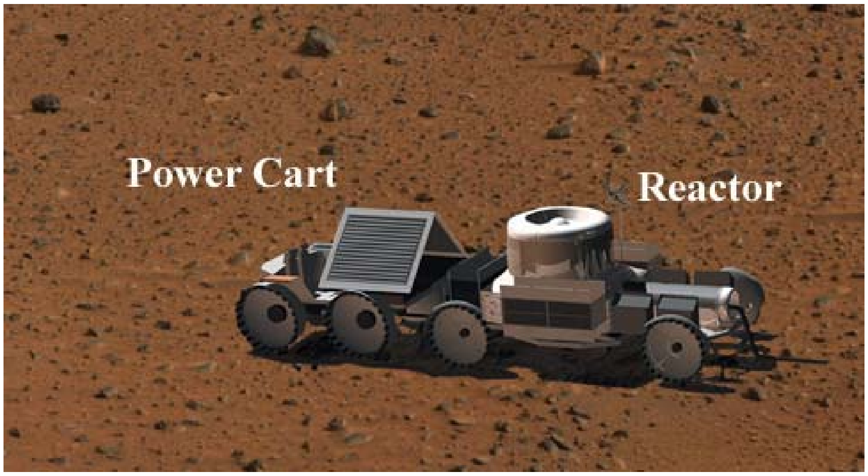



**Figure 6-24.** Reactor and power cart.

The advantage of this technology is that continuous (i.e., 24 hours a day, 7 days a week) power is available from this unit without the need for any recharging. It is envisioned that the DIPS cart would provide power to the reactor mobility chassis while it is being transferred to a location approximately 1 km from the landing site. The reactor has an external shield to protect the crew from radiation and adopted a guideline of 5 rem/year dose to the crew. Since the shield is a significant portion of the system mass, a shape shield is employed whereby the radiation is limited to 5 rem/yr (at 1 km) toward the habitat and 50 rem/yr (at 1 km) in all other directions. This creates a small exclusion zone and a limited pass-through zone for the base. One option to reduce or eliminate the exclusion zone and to save shield mass is to bury the reactor below grade where the soil provides additional protection, as has been suggested for lunar applications. However, the team felt that this option was risky due to numerous factors and has opted for the above-ground emplacement. If a second reactor were required for risk reduction, it would be possible to consider the crew assisting in burying and setting up a second nuclear power system, using power that is available from the first reactor unit.

With the above-ground option, the reactor would be driven about 1 km from the lander trailing the power cable. Once at the site, the mobile chassis would be aligned (orientate the shield), leveled, and secured by jacks. The DIPS cart, which would be outfitted with appropriate equipment, would assist in the deployment of the radiators if needed. The power cart would be driven back to the landing site and the reactor would be started. It was assumed that the total time to perform this is 30 to 40 sols.

### 6.3.5 Pressurized rovers for the Mobile Home and the Telecommuter Options

In addition to the main base power system, power system options were looked at for powering the pressurized rovers.

The Mobile Home option has two large, pressurized rovers that will each house three members of the crew with the capability of supporting all six crew members for a short term. There is no habitat, but the rovers come back to the landing site to get re-supplied with consumables; e.g., $O_2$, $H_2O$, food, etc. Each sortie is planned to travel 200 km in 30 days, with possibly 10 to 15 total sorties per mission.

The guideline from the science team was the desire to minimize distance traveled and maximize field science time. For the basis of the power system analysis, it was assumed that half the time was spent roving and half the time was spent stationary. It was assumed that a trafficability factor of 30% (avoid rocks, steep grades, soft sand, etc.) would be used to capture an “odometer” distance that the rover speed would be based on; thus, a total of 260 km would actually be traversed during the sortie.

The Commuter option has two smaller rovers that would house a crew of two and traverse 100 km (130 km total) in 15 days. This option has a habitat that the crews will return to and in which they would stay in between sorties.

Many scenarios exist for exploration during each sortie. Since there were no operating timelines from the science team, the following assumptions were used to evaluate the different power system options: Drive time was 5 hours each day, which dictated a speed of 3 km/hour to cover the total distance in the time allocated, and driving was only during sunlight.

Three power system options were evaluated for both the large and the small rover; these are summarized in table 6-18. The options included: PV/battery, PV/battery with DIPS augmentation, and fuel cell only. The significant drivers for both power and energy are the rover mass and drive speed. Drive power to achieve the 3 km/hour speed for the large and small rovers is 47 kWe and 25 kWe, respectively, as shown in table 6-18. It is a major challenge to meet the specified requirement of sortie distance in the allotted time. To keep the array area and battery mass to a minimum, recharging the system on as short a cycle as possible is needed. Therefore, for this analysis, we adopted the operation scenario of driving and stopping to do science and recharge on alternating days. Even with this strategy, the array size that would be required to recharge the batteries is 800 m$^2$, which must be deployed and stowed. If we assumed a 5-m-long rover and two 400-m$^2$ arrays, the crew would need to deploy each array about 80 m out from the rover. Adding a 5-kWe DIPS did not much impact the sizing due to the low ratio of load power to DIPS output. However, if the 30-day sortie were relaxed, speed could be reduced, and the resultant drive power reduces greatly. A speed of 0.5 km/hour brings the drive power close to the nominal crew power of 5 kWe. Array area and battery mass are reduced ,and now the addition of the DIPS allows a major reduction in array area and battery mass. One additional case was evaluated at 0.1 km/hour to reduce array area to a size that could be fixed on top of the rover, thereby eliminating the need for array deployment/stowage.

**Table 6-18.** Summary of Pressurized Rover Power Systems

| Speed km/hour | Array m$^2$ | Isotope Power kW | Drive Power kW | Battery kg | kW-hr | DIPS mass kg |
|---|---|---|---|---|---|---|
| **Large Rover (15,000 kg)** | | | | | | |
| 3 | 800 | 0 | 47 | 4,370 | 437 | 0 |
| 0.5 | 323 | 0 | 8 | 1,850 | 185 | 0 |
| 0.5 | 80 | 5 | 8 | 100 | 10 | 375 |
| 0.1 | 20 | 5 | 1.5 | 100 | 10 | 375 |
| **Small Rover (7,500 kg)** | | | | | | |
| 3 | 400 | 0 | 25 | 2,500 | 250 | 0 |
| 0.5 | 160 | 0 | 4.2 | 1,100 | 116 | 0 |
| 0.5 | 40 | 2.5 | 4.2 | 300 | 30 | 190 |
| 0.1 | 10 | 2.5 | 0.8 | 130 | 13 | 190 |

**Notes:**
Average slope 5 degrees
Recharge Time 1 sol
Drive Time 5 hours/sol
Array sized for winter solstice
Crew Power 5-kW day; 3.5-kW night; large rover, 3.4-kW day; small rover, 2.4-kWe night
Li-ion Battery, 100 W-h/kg, 70% DOD
Latitude 30
Fuel cell (FC) PEM, 70% eff., 2,000 psi

| **All Fuel Cell** | | | |
|---|---|---|---|
| Large Rover | 3,470 kg $O_2$, 433 kg $H_2$ | 9,590 kg total | 3 km/hour case |
| Small Rover | 975 kg $O_2$, 122 kg $H_2$ | 2,840 kg total | 3 km/hour case |

The small rovers have much less demanding power requirements than the large rovers, mainly due to the lower rover mass of 7,500 kg vs. 15,000 kg, not including the power system mass. It is still a challenge to meet the speed requirement, but the Commuter option seems much more plausible. Here again, a DIPS augmenting the array for power generation helps reduce the mass because it outputs power continuously and reduces the required battery capacity.

An FC-only option was assessed whereby the $O_2$ reactant could be produced by the ISRU plant during the pre-deploy phase. The $O_2$, $H_2$ and total FC mass estimates are shown based on accomplishing the full sortie roll-through (R/T) distance within the required duration.

Many options and combinations of such a hybrid system exist; therefore, since there was not enough time in which to come to closure on the exploration sortie operations scenario, we only investigated a portion of the trade space. Additional investigation of the use of differing DIPS power systems were assessed. We limited the power level to the minimum because of the cost and availability of the $^{238}Pu$ isotope. In fact, use of the DIPS mobile "Power Utility Cart" has many advantages. Since the DIPS supplies continuous power output, it has application to provide/augment power for many functions including to deploy the reactor or PV/RFC modules, power-assist the pressurized rovers, augment habitat night power, and provide habitat dust storm power.

The assumptions that were used for drive power calculation appear below.

A software package has been developed for the ISRU excavation system (Gallo, et al, 2008) [6]. This software models activities on the lunar surface that include excavation of regolith and transportation of the regolith to a processing plant. This software, which was written in Visual Basic with Microsoft Excel used for input and output, was used to simulate the travel of a rover on the surface of Mars. The code was initially written for activities on the lunar surface. The properties that were input to the code were adjusted to simulate those of the martian surface. These properties are used in the equations that model the interaction of the rover wheels with the martian soil. The values that were assumed for the martian surface are as follows:

| | |
|---|---|
| Mars Gravity (m/sec$^2$) | 3.72 |
| Regolith Density (kg/m$^3$) | 1,000 |
| Cohesion (Pa) | 10,600 |
| Modulus of Friction | 20,000 |
| Modulus of Cohesion | 306,800 |

Two runs were made for a large (15,000-kg) rover and a small (7,500-kg) rover. Each rover was assumed to have four wheels and an ability to drive up a 5-degree slope. The following rover dimensions were assumed in each analysis:

| | | |
|---|---|---|
| Vehicle Total Mass (kg) | 20,000 | 10,000 |
| Length (m) | 8.96 | 7.11 |
| Width (m) | 5.98 | 4.74 |
| Height (m) | 3.20 | 2.54 |
| Wheel Diameter (m) | 2.80 | 2.25 |

## *6.4 Surface Mobility Systems*

A key objective of the Mars surface mission is to get members of the crew into the field where they can interact as directly as possible with the planet that they have come to explore. This section will discuss one of the means by which this would be accomplished – the use of EVAs, assisted by pressurized and unpressurized rovers, to carry out field work in the vicinity of the surface base.

Although the list of specific field exploration activities will undoubtedly grow as landing sites with specific objectives are chosen and the means to accomplish them are defined, there are two examples that can serve to illustrate the range of these activities: field geology/mapping, and intensive field work at a specific site. Some of the key characteristics of each of these activities, as they apply to surface exploration, will be described in the following paragraphs. For Mars, astrobiology questions are likely to be equally important for the surface exploration mission. The specific investigations will be different from these geologic activities, but the functions that would be carried out during an EVA are likely to be similar, especially given that evidence of extinct organisms will be found in rocks and extant organisms are most probably endolithic. Thus, this section will be limited to geological activities as representative of the functions for both geologic and astrobiologic activities for these earlier human Mars mission.

The activities of a field geologist on the surface of Mars will differ greatly from EVA activities of the space shuttle and ISS eras. These differences will impact both the design and the use of EVA systems for surface activities. Some of these activities and the impacts that will result include the following :

[6] Gallo, Christopher, et al., Excavating Regolith on the Moon Using the ISRU Force/Mass System Model, Presentation at the Planetary and Terrestrial Mining Sciences Symposium, June 2008, Sudbury, Canad

> "Geologic field work involves collecting data about the spatial distribution of rock units and structures in order to develop an understanding of the geologic history and distribution of rock units in a particular region.
>
> "It is an oft-stated but correct maxim that the best field mappers are the ones who have seen the most rocks. Geologic field work on the planets, if it is to be worth the significant cost needed to get the geologists there, will require both EVA suits that will allow EVA crew to walk comfortably for hours at a time, and rovers that will allow the crew to see as much terrain as possible.
>
> "One distinction that needs to be emphasized is the difference between field mapping and pure sampling. A popular misconception is that geologists conduct field work purely for the purposes of sampling rock units. Sampling *is* an important part of field mapping, but sampling in the absence of the spatial information that field mapping provides leads to, at best, a limited understanding of the geology of a particular area. Having said that, the nature of the rock exposure in a given area can limit the amount of field mapping that can be done, and *can* drive field work efforts to conducting a sampling program that, with some ingenuity, can provide the basics for understanding the broad geologic context of a particular locality."

With this background, a typical field exploration campaign will begin with one or more questions regarding the geology in a particular region and the identification of specific surface features, based on maps and overhead photographs that offer the potential for answering these questions. Traverses are planned to visit these sites, typically grouping these sites together (into multiple traverses, if necessary) to meet the limitation of the equipment or environment (e.g., EVA suit duration limits, rover un-refueled range, crew constraints, local sunset, etc.). Depending on the anticipated difficulty of the planned traverse, the crew may choose to send a teleoperated robot to scout the route and send back imagery or other data for the crew to consider. (**Note:** These robot scouts are probably surface rovers, specifically the teleoperated rovers that were mentioned elsewhere in this document, but small aerial vehicles should not be discounted as options for this activity.) In addition, crew safety concerns when entering a region that is highly dissimilar from any explored before or an area with a high potential for biological activity may dictate the use of a rover in advance of the crew; this contingency will be discussed in a later section. The EVA crew walks, or rides if rovers are planned for the traverse, towards the first of these planned sites using visible landmarks and cues that are available through the surface navigation system. The crew stops at this planned site to make observations, record data (e.g., verbal notes to be transcribed later, imagery, sensor readings from the instruments that were brought on the traverse, etc.), and gather samples as appropriate. If a return visit to this site, either by an EVA team or a robotic device, is deemed necessary to gather additional data or samples, the position is marked with a small flag or other visible marker or as a "waypoint" for future use within the navigation system that is used for surface traverses. The crew then proceeds to the next site in the plan until all of the sites have been visited or until the crew is required to return to the outpost. At any point in the traverse it may be desirable to stop at unplanned locations due to interesting features that may not have been recognized as such during planning for the traverse. Similar activities will be carried out by the crew at these unplanned sites. Real-time voice and data, along with some amount of video, are sent back to the outpost to those members of the crew who are monitoring the progress of the traverse (along with other duties). On returning to the outpost, the EVA crew will ensure that all curation procedures are carried out and that information that was gathered in the field is transcribed or otherwise stored in the outpost data system. (Sample curation and sample analysis will be described in later sections.)

Intensive field work at a single site may involve one of several activities that are associated with science payloads carried in the DRA manifest or comparable activities that may be part of the unspecified "discretionary Principal Investigator" science. Two specific examples for which there are manifested payloads include the set-up of geophysical/meteorological stations and a small drill.

Expanding on the case of the small drill to illustrate this type of activity, several key scientific and operational questions will require subsurface samples that would be acquired by this tool. Examples include searching for subsurface $H_2O$ or ice, obtaining a stratigraphic record of sediments or layered rocks, or obtaining samples to be used for a search for evidence of past or extant (possibly endolithic) life. A traverse of the type that is discussed above will probably have been carried out to examine candidate sites for the drill, with the acceptable sites being placed in a priority order. Drill equipment will be moved to the site, most likely on a trailer that is pulled by either the unpressurized or robotic rovers, and set up for operations. The set-up process will likely be automated, but with the potential for intervention by the crew. Drilling operations are also likely to be automated, but under close supervision by the crew. At present, drilling is still something of an art, requiring an understanding of both the nature of the material that is being drilled through (or at least a best guess of the nature of that material) and of the equipment that is being used. While drilling is a candidate for a high level of automation, it is likely that human supervision to "fine-tune" the operations and intervene to stop drilling will remain a hallmark of this activity. Core samples will be retrieved by the crew and put through an appropriate curation process before eventual analysis. After concluding drilling at a particular site, the drill equipment will be disassembled and moved to the next site, where this procedure will be repeated.

Because of the nature of the drilling process, there is a high probability that the above-surface equipment will fail or the below-surface equipment will break or seize. Crew intervention is highly likely in either event. In the first case, the crew must decide whether the failure can be fixed in the field or whether the equipment must be returned to the outpost for repair. Either option will involve some amount of equipment disassembly. If subsurface equipment fails, the crew must decide how much of this equipment can be retrieved with the tools that they have available and whether it is worth the effort and resources to make this retrieval. Due to cargo mass constraints, the drill will not have an unlimited supply of drill bits, auger bits, or drill stems. This makes it worthwhile to expend some effort to retrieve as much of the salvageable subsurface equipment as possible and attempt a repair; the alternative is to halt drilling operations until adequate replacements arrive, probably with the cargo flights supporting the next crew.

The two key characteristics that should be noted here are that drilling activities and, by inference, other intensive field work will involve repeated trips to a single location and an extensive interaction with tools and equipment at these sites.

### 6.4.1 Extravehicular activity design and operational guidelines

As a practical matter, the examples that are described above, and other EVA tasks that will be identified as the surface mission matures, would be translated into more specific design assumptions and operational guidelines. These would in turn lead to specific requirements and flight rules. Based on past experience, plans for ISS, and current knowledge of the Mars surface mission, this transformation process has already begun (Griffith, 1998). While these discussions are ongoing and would be subject to change as systems and operations mature, the following list indicates some of the assumptions that are being proposed for Mars EVA activities:

- The buddy system of paired EVA crew members will always be used.
- Standard EVA protocols such as gloved hand access, no sharp edges, touch temperatures within supported limits, and simplified tool interfaces must be applied to every element that is expected to be handled or encountered by suited crews.
- A safe haven must be readily available at all ranges beyond walk-back distance.
- Seasonal effects, such as number of daylight hours, dust storms, and possibly radiation events, will be taken into account during planning, timing, and support of EVAs.
- Planned EVA contingency support will account for sickness, injury, and potential incapacitation of an EVA crew member in addition to suit/equipment problems.
- Time delays between Earth and Mars require that primary support for the EVA crew be provided by the crew that is remaining in the habitat. Earth-based personnel may participate, but as backup. In both cases, real-time voice, video, and data between the EVA crew and the habitat support personnel are required. Loss of these links may, depending on distance, terminate the current EVA.
- Nominally, there would be only one pair of crew members outside the habitat or a pressurized rover at a time. It may be possible to have two pairs of crew members outside in extreme cases, but only for local maintenance/support or in the event that one pair of crew members is rescuing the other pair of crew members.
- EVA during nighttime will be trained and possible, but not nominally planned, and will be constrained to the local area (i.e., in the vicinity of the habitat or a pressurized rover).
- The EVA suits will have minimal prebreathe and require minimal turnaround maintenance between uses.

As is apparent in the previous discussion, conducting geologic investigations on the surface of Mars will require extensive EVA to take advantage of the human element over robotic rovers. The EVA system, therefore, is a critical element in maximizing the science return from a human Mars mission. The EVA system that is currently under development for the lunar surface will operate under environmental conditions that will be inappropriate for use on Mars. Two characteristics of the martian environment dictate this: increased value of the surface gravity from 1/6 g on the lunar surface to 1/3 g on Mars; and the change in the atmosphere from essentially a vacuum to an approximately 10-mbar $CO_2$ and Ar atmosphere.

The present shuttle/station extravehicular mobility unit (EMU) and planned lunar suit system will take advantage of the lunar vacuum to dump waste heat to the external environment by using the phase change that is associated with exposure of $H_2O$ to a vacuum. Such a system works because $H_2O$ that is exposed to vacuum through a finely perforated membrane instantly freezes, creating phase change heat sink that can be used to deposit crew member metabolic heat. The membrane allows the $H_2O$ to be released into vacuum through sublimation. This process, in turn, creates a layer that is continually forming on the "liquid water" side of the ice and is continuously sublimating on the vacuum side of the ice. This system must have a hard vacuum to work; the 10-mbar atmosphere on Mars has too high pressure to allow this kind of system to work. Consequently, an alternate method of crew member heat dissipation must be developed. Relatively little time or effort has been put into such a system. Thus, there are no "leading contenders" for the technological implementation of such a system; it is simply recognized as the one major subsystem that cannot be easily evolved from the current system or the anticipated lunar system.

In addition to dissipation of crew member waste heat, a pressure garment must have an outer layer to prevent heat loss, particularly from hands, feet, and limbs. In the present spacesuit system, this outer layer, which is called a thermal garment, uses vacuum as part of the MLI. Both of these systems make use of the vacuum of the lunar environment, and will not work in the 10-mbar martian atmosphere. Consequently, the suit insulation will require redesign to work in the martian environment. As with the heat-dissipation subsystem, there are no "leading contenders" for the technological implementation of this component of a Mars EVA suit, but the insulating technology is relatively well understood.

The increase in gravity places additional demands on EVA system mass. Tests that were conducted by the JSC Life Sciences Directorate have indicated that while in 1/6 g, the EVA crew member is largely insensitive to the on-the-back weight of the pressure garment and PLSS, and can tolerate masses in excess of 100 kg. In 1/3 g, however, the on-the-back mass of the EVA system becomes critical. The system redesign that would be necessitated by the martian environment would also need to concentrate on reduction of the on-the-back weight to not induce high metabolic rates on Mars surface crew members.

### 6.4.2 Surface transportation: unpressurized and pressurized rovers

Surface transportation for EVA crews will be a requirement from the outset of these Mars missions. Several factors are contributing to this. First, safety considerations for landing may drive landing site selection to a location that is free of terrain features that have the dual distinction of being both "landing hazards" and "interesting geological sites." Second, a crew is likely to exhaust interesting sites within walking distance during an 18-month surface mission, even if there are only a modest number of EVAs allocated for the mission. Third, regardless of how well mission planners can "centrally locate" the landing site, undoubtedly important sites will either be located at a significant distance from the surface base or extended times will be necessary during which to fully explore the area. Even at distances that are considered within walking range, access to surface transportation has been found to enhance crew productivity, both to mitigate crew fatigue and to extend consumable supplies by allowing lower metabolic rates during seated travel. Work by the JSC Life Sciences Directorate has quantified the consumable savings from using rovers as opposed to walking everywhere. This research indicates that for walks that are greater than approximately100 m, it will be more economical to drive than walk. These tests were done in simulated lunar gravity; intuitively, one would expect even better savings in martian gravity. The minimum capability that is needed to achieve this enhanced efficiency is a basic rover that could carry two crew members plus cargo under nominal circumstances, one that is similar to the Apollo LRV.

Thus, the capability to travel easily and quickly away from the landing site will be necessary for the crew to remain fully productive throughout the surface mission.

Two options for crew surface transportation are typically mentioned in Mars mission studies: unpressurized (and, thus, limited-duration) rovers, and pressurized (and, thus, extended-duration) rovers. Each has its advantages, which tend to be complementary, and the availability of both types will provide flexibility for surface operations.

#### *6.4.2.1 Unpressurized rovers*

Unpressurized rovers will obviously require the use of EVA suits by the crew. This implies that the capabilities and interfaces of the unpressurized rover will be intimately tied to those of the EVA suit. This, along with the previously stated reliance on surface transportation for the crew to remain at a high level of effectiveness over a long duration, allows the unpressurized rover to be viewed in many ways as an extension of the EVA suit. From this perspective, many of the heavier or bulky systems that would otherwise be an integral part of the suit can be removed and placed on the rover, or the functionality of certain systems can be split between suit and rover. In the case of offloading capabilities to the rover, navigation, long-range communication, tools, and experiment packages can be integrated with or carried by the rover. In the case of splitting functionality, any of the various life support system consumables (e.g., power, breathing gases, thermal control, etc.) can be located on both the rover and within the EVA suit. This division or reallocation of EVA support functionality may restrict the maximum duration of the EVA suit to something less than that which has been previously demonstrated. However, analysis of Apollo EVA activities using the LRV indicate that approximately 20% of the total EVA time was spent by the crew on the LRV moving from site to site (Trevino, 1998). Mars surface operations can be assumed to be comparable. Thus, the EVA team will have sufficient time for recharge of EVA suit consumables or switching to rover-based support systems to preserve EVA suit consumables. Providing multiple sources of consumables and support systems in the field also enhances crew safety by providing contingency options should EVA suit systems degrade or fail.

Operationally, Mars surface EVAs will be conducted by a minimum of two and by a maximum of four people. (This will always provide for a buddy system while on an EVA, but will also leave at least two people in the SHAB for contingency operations should they be needed.) If unpressurized rovers are used, an additional operational constraint will be imposed on the EVA team. If one rover is used, the EVA team will be constrained to operate within rescue range of the surface base. This could mean that either the team has sufficient time to walk back to the surface base if the rover fails, or that there is sufficient time for a rescue team from the surface base to reach the team. Taking multiple, and identical, rovers into the field allows the EVA team to expand its range of operation because these vehicles are now mutually supporting and, thus, are able to handle a wider range of contingency situations. It is reasonable to assume that, while operating in terrain that is similar to that seen thus far in images of the martian surface, a rover could easily become stuck or otherwise unable to move but still be functional. In a single rover operation, this would be sufficient cause for the EVA team to start walking back to the outpost or to call for assistance from the personnel who are remaining at the outpost. However, under these circumstances, rovers that are not immobilized are available to help extract the temporarily immobile vehicle. In the case of a disabling component failure, the other rover(s) are available to provide power, lighting, etc. as field repairs are attempted or, in a worst case, transport the crew of the failed rover back to the surface base.

This description points out two additional characteristics of the unpressurized rovers. First, it points out that these rovers must be reliable but also easily repairable in the field (or at least have the capability to be partially disassembled in the field so the failed component can be returned to the outpost for repair). Second, it indicates that the rovers must be sized to carry cargo that, if offloaded, is of a sufficient capacity to carry the crew of a disabled rover.

Within these constraints, the unpressurized rovers will be capable of supporting any of the various EVA activities that have been discussed in pervious sections. For purposes of this DRA assessment, relatively little effort was spent on developing details for this system. For other performance calculations, a mass of 250 kg was assumed as a representative value for a vehicle with performance characteristics similar to the LRV (LRV, 210 kg) but able to perform in the martian gravity environment an for approximately 500-sol mission durations.

#### *6.4.2.2 Pressurized rovers*

Pressurized rovers are typically included in Mars mission studies because of their ability to extend the range of the crew in terms of both distance and duration. While exact distances and durations will be dependent on the specific site that is chosen, input received from the HEM-SAG indicated a strong desire to reach locations several hundred kilometers from the outpost for durations measured in days to weeks between resupply. It was also the intent for the crew that would be using the pressurized rover to be capable of performing many of the same functions as at the outpost, but at a reduced scale. Thus, a crew using a pressurized rover can be expected to be capable of commanding and controlling teleoperated rovers, conducting EVAs (comparable to those discussed earlier) within the vicinity of the rover, and otherwise supporting crew members for the duration of their excursion away from the outpost.

For this DRA assessment, a modest, pressurized rover capability was assumed. This rover was scaled to support a crew of two (with the ability to support four people in a contingency) for a period of approximately 1 week without resupply and travel for a total distance of approximately 100 km. These two pressurized rovers are assumed to be nimble enough to place the crew in close proximity to features of interest (i.e., close enough to view form inside the rover or within easy EVA walking distance of the rover).

As with the unpressurized rover, relatively little effort was spent during this DRA assessment on developing details for this system. Previous pressurized rover studies were reviewed and a mass of 4,800 kg was estimated for a vehicle with the capabilities that are assumed for this vehicle.

## *6.5 Science Systems*

Specific science systems that would be used for future human Mars missions will evolve in terms of type, physical characteristics, and quantities as the questions being asked about Mars evolve. In general, these questions can be grouped into the following broad categories: geological, astrobiological, meteorological and climatological, and human physiological. Section 2 has discussed some of the current and anticipated questions that are representative of the specific questions that will drive the ultimate selection of science equipment and characteristics. It is reasonable to assume that this science equipment complement will be made up of common, or "facility," equipment that will be sent on all missions and discretionary, or competitively selected, equipment that may be flown on one or more missions.

For purposes of this study, no attempt was made to anticipate a specific science experiment or instrument complement. Rather, a mass allocation was set to account for these payloads, and the science community was invited to discuss how this mass should be reasonably distributed among the categories that were cited above (no final resolution was made beyond that discussed in section 2). For this reference architecture, an allocation of 1,000 kg was made for equipment that would be used on the surface and of 100 kg was allocated for samples that would be returned to Earth. While these values are somewhat arbitrary, they are representative of the order of magnitude that has resulted from previous bottom-up analyses. An important factor to note is the relative magnitude of these two allocation values that is indicative of the very real fact that it is much more costly to return samples to Earth than it is to deliver instrumentation to the surface of Mars. This fact ,coupled with the extensive amount of time that the crew will spend on the surface, should be used as guidance to develop instrumentation and procedures that are effective at addressing or answering scientific questions while on the martian surface with the goal of minimizing or eliminating the need for returned samples.

# 7 KEY CHALLENGES

One of the principal challenges of future human exploration of Mars is to build a program that is credible in costs and schedules for critical near-term technology development and, at the same time, one that is cognizant of the broader spectrum of technologies that enable longer-term program goals. Success in meeting this challenge depends upon a solid understanding of state-of-the-art engineering systems, as well as a feasible projection of what can confidently be achieved through focused research, technology development, and system-specific advanced development programs.

Human exploration of Mars will build directly on the technologies that are developed for and proven during the robotic phases of the Mars robotic program and lunar outpost missions. Sending humans to Mars will mandate developments in almost all areas of technology as well as understanding system performance and operational concepts to reduce potential risks. In particular, major advances will be required in life support for the in-space and SHABs; space transportation propulsion advancements; EDL of large payloads on Mars, cryogenic fluid management; and utilization of locally produced consumables and power systems, to name a few.

A number of technology requirements apply to more than one mission element. These include substantial increases in ground and surface operations automation; in-space system autonomy; diverse applications of advanced electromechanical manipulator systems, using control approaches ranging from teleoperation through telerobotics to full robotics; and requirements for data and control system components and software that increase the fault-tolerance of system operations, including automated fault detection, isolation, and resolution. Human safety and health during long-duration missions will have high priority, and will pace and direct technology development across all phases.

A major challenge of the human exploration of Mars is the need to dramatically decrease the total mass that must be launched into LEO and transported to the martian surface. Although additional factors, such as crew time, power, and servicing requirements, are very important, reducing launched mass is an overarching need for long-term self-sufficiency and acceptable operations costs. Critical technologies in regenerative life support, aeroassist, and advanced space-based cryogenic engines must be developed to substantially reduce the mass of near-term systems. Mid-term technologies that are critical to decreased mass are surface nuclear power, ISRU, and radiation shielding. In addition, although human expeditions to Mars can be conducted using cryogenic propulsion and aerobraking, nuclear propulsion presents a compelling prospect for tremendously reducing the mass or travel time required.

## *7.1 Human Health and Performance*

As humans extend their reach beyond LEO to the surface of Mars, they will be exposed to the hazardous environment of deep space for lengthy periods; consequently, protective measures must be devised to ensure crew health and maximize mission success. The health and safety of crew members while they travel to and from the Mars and inhabit its surface are key near-term concerns. The explorers must be protected from the space radiation environment and from the physiological effects of reduced gravity. To maintain the fitness and productivity of the crew, medical care must be provided during long stays in very isolated and distant places.

A thorough ground-based research program coupled with flight research on the ISS must be conducted to provide an understanding of the physiological basis for human responses, develop appropriate treatments and countermeasures, and decide how best to support crew members. Simulating the environment that will be inhabited by crews on Mars is an important facet of the research program. Much of the work can be done on the ground, but many studies will require access to space facilities. The ISS is currently serving as a vital test facility for research that demands long exposures to the reduced-gravity loading conditions in spacecraft and on planetary surfaces. That research will establish the baseline for the 6-month transit from Earth to Mars, and is forming the foundation of extrapolations and inferences that is necessary for near-term planning for the 18-month Mars surface habitation and the 6-month return transit to Earth. In preparation for Mars missions, research on the moon will also be essential. Human adaptation to long-term exposure to partial-gravity conditions is a critical component of future long-duration surface operations on Mars. The moon provides an ideal venue for verifying and refining the protocols that are established on ISS.

In many cases, alternative solutions exist to problems that would be faced by humans. For example, countermeasures can be developed for physiological degradation due to reduced-gravity loading on long-duration missions, advanced propulsion systems can be developed to shorten travel time, or vehicles providing an artificial-gravity environment can be developed. As the challenges of sustaining humans in space are resolved, advances in fundamental science, medicine, and technology will follow.

### 7.1.1 Radiation protection

The magnetic field of Earth protects our planet from radiation that is emitted by solar flares and shields it from a large fraction of galactic cosmic rays. When space missions travel beyond that magnetic shield of the Earth, the radiation that is received is different in type and intensity and, thus, the effects of the radiation on living cells are different. The radiation dose beyond Earth orbit could exceed projected exposure limits for astronauts; without adequate protection, crew exposure to high radiation dosages in solar flare events could cause catastrophic effects, including radiation sickness and even death. The long-term chronic (e.g., post-mission) effects of exposure to galactic cosmic rays could include genetic damage, cataract formation, and cancer.

Radiation protection goals fall into three categories: (1) determination of career dose limits and development of countermeasures that can reduce the adverse effects of radiation exposure; (2) provision of sufficient radiation shielding in planetary habitats and in the MTV to protect both crews and sensitive equipment from the normal GCR background, thereby extending the length of time that crews can safely remain in these environments; and (3) establishment of space weather forecasting systems and implementation of sufficient "storm shelters" to warn crews against the transitory. but extreme. Levels of radiation that would be encountered during solar flares.

The NCRP guides NASA on crew-permissible exposure limits However, these limits are based on the characteristics of radiation present in LEO, where the magnetic field of Earth and its atmosphere provide protection from solar flares and GCR. A Mars mission would exceed the current annual dose limit guideline; thus, revised radiation exposure standards for human exploration missions outside the protective magnetic field of Earth will have to be developed. The standards must take into account, in the context of the inherent risk of exploration missions, the risks that would be associated with the specific types of space radiation, such as SPEs and GCR and their biological effects. Even after new exploration guidelines are established, NASA will provide a system of radiation protection that adheres to the ALARA principle, which recognizes that, although an acceptable upper limit of exposure is set, the residual risks should be minimized even further where it is reasonable to do so.

Determination of allowable doses will require better information on the effects of high-energy galactic cosmic rays on living matter. These effects can be studied, in part, using artificial radiation sources at the Brookhaven National Laboratory, Lawrence Berkeley Laboratory, and other radiation research centers. The ground-based research program will generate new information on the biological effects of high-energy heavy ions and secondary particles, and will develop and test dosimetry technology.

Solar flares require different shielding strategies than GCR. Because of the high energy of GCR, extremely thick shields would be required, so it is generally not feasible to use add-on shielding. Protection that would be afforded by substances with a high content of $H_2$, such as propellant and $H_2O$ tanks, will prove useful. The secondary radiation spectrum that is produced by the interaction of the GCR with the shielding material may be more harmful to living tissues than the primary dose itself. On planetary surfaces, the bulk of a planet shields against half the cosmic radiation that is received in space; but again, the generation of secondary radiation from surface materials may prove to be problematic. The martian atmosphere, particularly at lower elevations, provides substantial additional radiation protection. Nevertheless, individual radiation doses will need to be monitored, and the GCR dose will probably be the ultimate limiting factor for human exploration.

The most acute source of space ionizing radiation for Mars explorers is an SPE, which accompanies some solar flares. The amount of radiation can be so large that the dose the explorers, if unprotected, would receive significantly exceeds all limits, and can result in rapid death. However, to protect the explorers for limited periods, "storm shelters" can be constructed in the most heavily shielded areas of the spacecraft and habitats, and can be provisioned with sufficient consumables to maintain humans for the maximum estimated duration of an SPE (from a few hours to several days). The storm shelter and more protected areas of the Mars spacecraft could be occupied by the crew on a intermittent basis (e.g., while sleeping) during transit to provide added protection from GCR. To develop the best radiation shielding strategies for Mars habitats and the transportation vehicles, robotic missions will help to determine the nominal background radiation that is encountered during transits to and from Mars, as well as on the planetary surfaces, and will measure radiation doses that are received during SPEs. These radiation environments will then be modeled and simulated using ground-based radiation research facilities to determine the effectiveness of various shielding materials in protecting living tissues. The results of these studies will influence habitat and vehicle designs.

A system for alerting the crew is essential to planning EVA traverses, which would not be scheduled for periods in which a flare was expected. Warning must be received in sufficient time to allow the crew to return to the habitat or storm shelter before the buildup of radiation from an unanticipated flare puts the crew at risk. Solar flares are currently unpredictable to the extent that warning times at a spacecraft may be as short as 30 minutes. Improved predictions will require long-term observations of the magnetic field of the sun and its relationship to solar flares, and specific warning systems will need to be developed.

#### 7.1.2 Reduced-gravity countermeasures

Microgravity exposure significantly reduces the forces that are normally imposed on the body on Earth. Space flight experience has shown that significant physiological changes occur during exposure to reduced gravity; most notably, bone mineral loss and architectural changes, muscle atrophy, and cardiac de-conditioning, all of which become more severe as stay-time in space increases. Although these effects can be minimized if crews take certain preventive measures while in space, the problem of developing effective countermeasures to reduced gravity is significant.

So far, reduced-gravity countermeasures have relied heavily on exercise regimes, which are usually vigorous and protracted, to provide the desired countermeasures. At the present time, it is not known whether exercise will be capable of maintaining crew health for very long missions. Additionally, astronauts will have difficulty maintaining the required exercise program (2 hours a day or even more) for the protracted periods that are envisioned for most exploration missions. If an astronaut were to suffer an accident or a serious illness and be unable to exercise, more severe de-conditioning would result. A comprehensive reduced-gravity countermeasure program, which is coupled with very fast transits to and from Mars, can reduce the expected loss of physical capacity to within current space flight experience.

The major concern relates to the long transit times to Mars coupled with the demands that would be placed on the crews immediately upon arrival at the martian surface. The baseline transit time to and from Mars is 200 days in zero g. In the case of a flyby abort or an abort to the orbiting crew MTV shortly after landing, the astronauts could remain in reduced gravity for 3 years, which far exceeds human space flight experience. Exercise, nutrition, and pharmaceutical countermeasures show promise in controlling the adverse physiological effects of long-duration exposure to reduced gravity. Three alternative Mars transit options also exist: shorten the outbound and return transit times by using advanced propulsion systems; employ artificial-gravity countermeasures within the spacecraft either by providing an on-board centrifuge or by spinning the spacecraft itself; or accept the higher risk that is involved and proceed with the mission using the best available countermeasures. A zero-g transit has the advantage of more habitable volume (i.e., ceilings and floors) available for use by the crew, whereas artificial gravity has the advantage of providing Earth-like conditions and minimizing deconditioning. In addition, during the execution of the mission, the flight crews will experience gravity field transitions multiple times throughout the mission: 1g at Earth, multiple-g's during launch (minutes), zero-g during transits to Mars (180–210 days), multiple-g's during entry (minutes), 3/8g while on Mars (500 days), multiple-g's during Mars ascent, zero-g during transit back to Earth (180–210 days), multiple-g's during Earth entry, and finally 1g post landing. If artificial-gravity spacecraft are used during transits, multiple additional gravity transitions will be required during major spacecraft maneuvers such as trajectory correction maneuvers.

A zero-g countermeasures program is being conducted on ISS. Appropriate crew stay-time in orbit, combined with the increase in crew size to six, provides an adequate statistical basis for the vital countermeasure information. During the countermeasures program, crew members serve as test subjects while performing normal operations in support of ISS. In addition, countermeasures that are developed to mitigate the most severe de-conditioning effects of microgravity will be used at the lunar outpost and on Mars to maintain crew health and performance in these reduced-gravity environments. Zero-g countermeasures may not be sufficient to maintain crew health and performance for a Mars mission, however. Adverse physiological changes due to reduced gravity may be prevented by exposure to some level of artificial gravity, but the specific level of gravity and the duration of the exposure that is necessary to prevent deconditioning are not yet known. Some level of artificial gravity should amplify the effectiveness of exercise countermeasures. Although artificial gravity should reduce or eliminate the worst deconditioning effects of living in zero g, rotating environments frequently cause undesirable side effects, including disorientation, nausea, fatigue, and disturbances in mood and sleep patterns. If artificial gravity were to be employed, significant research must be done to determine appropriate rotation rates and durations for any artificial-gravity countermeasures. The decision on whether artificial gravity must be employed to adequately support crews on their transits to and from Mars, as well as the decision on the necessary gravity level and rotation rate, has significant implications for vehicle design and operations. Life sciences requirements for artificial gravity must be developed using an integrated approach that combines physiological studies and engineering feasibility. Preliminary physiological requirements can be developed using ground facilities to simulate weightlessness, fractional gravity, and rotational effects. For example, slowly rotating rooms can be used to examine the chronic physiological effects of continuous rotation. However, these requirements can only be verified by experience in artificial-gravity environments in space flight that are free of the many confounding factors of ground-based simulations.

The results of these studies will determine which strategies are necessary to maintain crew health during long-duration exposure to different gravity environments. Research that will be conducted on the lunar surface will provide important information about the effects of the one-sixth lunar gravity on humans. This will be an early data point in determining whether fractional gravity levels can assist in maintaining human health and performance.

### 7.1.3 Medical care

Maintaining crew health is required for sustaining a high level of performance and productivity both in transit and on the surface of a planet. Health systems will be required to provide appropriate medical care, environmental monitoring and regulation, and optimization of human performance. The approach to health and performance systems is to evolve toward increasingly higher levels of self-sufficiency.

Human Mars missions will have small, highly autonomous crews that would be situated in remote locations and unable to return immediately to Earth in the event of a medical emergency. Therefore, on-site medical care would be needed to accommodate major and minor illnesses and injuries and to provide critical surgical capability. The design of space medical care systems will adhere to acceptable U.S. standards. Providing medical care in remote locations is a challenge that has traditionally been met with immediate provisions for stabilizing illness or injury followed by transporting the patient to a medical facility that provides an optimum level of care. Experience in remote Earth and space flight settings has shown that medical care is essential in returning crew members to normal activities, averting serious illness, and preventing unnecessary medical transport operations.

Medical care systems for Mars outposts and for the MTV will build on and expand the capabilities of the ISS to include in-patient, out-patient, and dental care. These systems will also provide enhanced Earth-based support systems, in-space support, medical computer-aided artificial intelligence systems, and Earth-to-remote locations telemedicine capabilities using state-of-the-art telecommunication systems for consultation in diagnosis and treatment. As time spent on planetary surfaces increases, medical care capability will also expand to provide diagnosis, laboratory analytical capabilities, anesthesia, surgery, and pharmaceutical support. The area of medical care will provide perhaps the most stringent demonstration of crew autonomy, given the 3- to 20-minute time lags in communications with Earth coupled with the potential for literally life-or-death decisions.

One unique issue that is related to medical support on Mars is that the martian surface material itself may present a health hazard to the crew. Analysis for toxic and irritating substances and for any potential biological hazard must be done prior to human exploration; the MSR mission is planned to directly address this issue.

Research and advanced development to extend the shelf-life of certain pharmaceuticals and blood products are required, as are the development and test of operational procedures in reduced-gravity and zero-g environments. The capabilities will be demonstrated using moon and Mars advanced development health maintenance facility testbeds.

### 7.1.4 Supporting human life

Maintaining a safe environment for human habitation goes beyond the minimum required to sustain life by providing adequate air, food, $H_2O$, and waste-handling systems. The habitable environment must also be conducive to maximizing crew productivity by minimizing physiological and emotional stresses. The environment must be monitored and controlled for the presence of toxins, either microbial or physiochemical, and it must maintain appropriate temperature, humidity, and atmospheric composition. The buildup of toxic substances in a tightly closed environment (the “tight building” syndrome) is a design challenge that may be made more difficult, relative to all previous human space missions, by Mars planetary protection requirements that might prohibit the venting of waste gases and liquids. Analytical methods need to be refined to predict the toxic buildup rates as well as the projected levels and sources for extended lunar and Mars missions.

It is a major challenge to provide a reliable, life-sustaining environment in locations that are naturally devoid of food, air, $H_2O$, and nutrients, and to do so without incurring a mission-limiting dependence on provisioning of equipment and supplies from Earth. Without these commodities, there can be no exploration, no reaching out, and no discovery. Life support systems for exploration will provide these commodities in all phases of the exploration missions: traveling in space transfer vehicles, living in SHABs, and working on planetary surfaces: These systems must have capabilities for air revitalization, $H_2O$ purification, food supply, waste processing, environmental monitoring and contamination control, thermal and humidity control, and fire suppression. Optimizing systems for operation in microgravity during the transit to and from Mars and in the 1/3g field on the martian surface will need to be traded with the benefits that would potentially be gained through commonality in reduced development costs and increased system redundancy.

Perhaps the biggest challenge will be the development of regenerative life support systems, which could eventually provide food as well as recycle wastes, with a subsequent reduction in the quantity of supplies that must be transported from Earth. However, little is known about the operational characteristics and risks that are associated with long-term operation of regenerative life support systems incorporating integral food production. System performance must meet a variety of standards ranging from nutritional requirements to environmental quality. Challenges include minimizing the electromechanical infrastructure that is needed to reliably sustain food systems in both nominal and off-nominal situations and the need to maintain food system viability during the prolonged periods of vehicle dormancy that will be necessitated by mission design constraints.

Current spacecraft life support systems provide a habitable environment in space, but they are logistically costly to operate. Food is stored, air is recycled, and most $H_2O$ must be resupplied from Earth. By mass, $H_2O$ is the most burdensome commodity. A notional mass balance depicting the basic metabolic needs for a four-person crew is shown in figure 7-1.

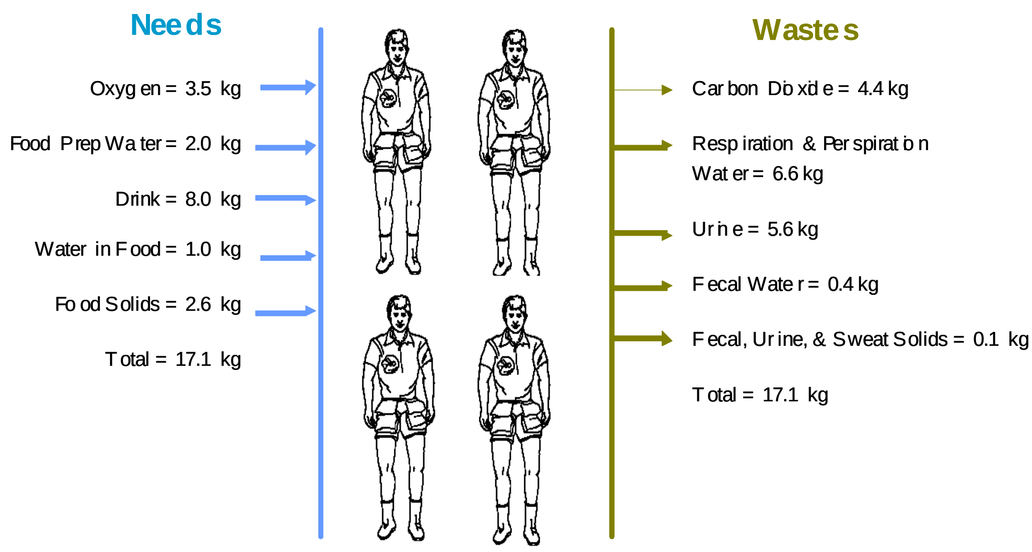



**Figure 7-1.** Metabolic mass balance for a four-person crew.

Additional $H_2O$ beyond that depicted in figure 7-1 might also be required to for such needs as bathing, washing dishes, flushing toilets, and washing clothes. The ISS Program is developing systems that will use recycling technology to supply a portion of the potable $H_2O$ and part of the hygiene and wash $H_2O$ for the crew. For the MTV, development of a more highly regenerative system, which is a natural evolution from the ISS systems but with a higher degree of $H_2O$ recovery and less dependence on expendable usage, might enhance mission design while offering the potential for high reliability founded in years of ISS operational experience. For planetary surface facilities, most of the development work can be ground-based.

Considerations for the design of life support systems for human exploration missions typically lead to the use of closed-loop systems, wherever practicable, to reduce logistics requirements as well as to open-loop systems wherever logistics penalties are tolerable within the mission architecture.

The in-space transfer vehicles as well as SHABs that would be used during human Mars missions will require advanced closure of the life support system. Research and technology development for both physical-chemical and bio-regenerative life support systems is in progress. Because of the time that will be required to achieve technology maturity, early lunar outpost elements will likely rely on physical-chemical life support systems derived to some extent from ISS systems.

In addition, the atmosphere of Mars can be used to generate consumables for the crew to supplement potential losses from the habitats due to outgassing and EVAs as well as $O_2$ for the crew for enhanced exploration for EVA systems. Oxygen can be extracted from the atmosphere as well as buffer gases including $N_2$ and Ar as well as $H_2O$. Although this is a very promising capability, further assessments, including the human health impacts of such an approach, must be fully developed.

#### 7.1.5 Behavior, performance, and human factors

Space environments, like other isolated and confined environments, induce stress; additionally, there are physiologically induced stresses. If these stresses are not appropriately managed, combined environmental and physiological stress is likely to result in behavior and performance deterioration during long-duration space missions. Humans have never embarked on space flight missions approaching the scale of the exploration that is now envisioned; the best analogs so far may be Antarctic expeditions and undersea experiences. Although no Earth-based analogs are perfect, such analogs provide insight into some of the unique attributes that would be present in space exploration – namely, alteration of day-night cycles, telecommunications to outpost operations, absence of other living creatures, self-sufficiency, and profound isolation – that affect crew dynamics and performance on space exploration missions. Using the analog environments and specialized ground simulation facilities as testbeds, and building on data that were obtained from ISS and the lunar outpost, strategies will be developed to support the increasingly complex and demanding Mars missions.

The exploration missions must be carefully examined from a space human factors perspective. The key issue is the effect that prolonged exposure to the space flight environment has on individual psychological and behavioral functioning and on crew effectiveness and performance. Psychological, social, perceptual, and behavioral conditions affect crew performance, productivity, and safety. Spacecraft architecture and outfitting are particularly relevant because they can either enhance or reduce effective performance. Decisions that are made in all of these areas drive habitat design; and it is clear that for long-term, extended missions, crew accommodation volumes will need to be larger than those on ISS.

Crew composition will be based on those personal and interpersonal characteristics that promote smooth-functioning and productive groups, as well as on the skill mix that would be needed to sustain complex operations. Studies addressing these areas and the influence of task and authority structures and introduction of new members and unfamiliar crews need to be conducted to determine effects on crew performance and productivity. Positive interactions and communications between ground and the crew during all segments of the exploration missions are essential. In addition, task assignments for lengthy missions must be perceived by the crew members as productive and significant. Performance timelines must be developed that are realistic in their use of crew time and skills. Based on the studies, crew training, task assignments, and support strategies will be developed, implemented, and monitored.

Crew composition, training, and skill retention should be studied in ground-based laboratories, simulations, and analog testbeds. A ground-based lunar simulator should be developed to evaluate concepts, procedures, and equipment to gain an understanding of the human factors and psychosocial issues that are related to crew performance and lunar habitat design and operation. One of the major issues for long-duration missions is cross-training, which is necessary because of the limited or nonexistent capability for return to Earth in the event a crew member becomes incapacitated. Moreover, the small number of crew members compared to the large range of tasks to be performed requires a high degree of proficiency in multiple specialty areas and the retention of that skill when opportunities to practice it are limited. In addition, since crew members may dedicate entire careers to one mission, career development and training take on greater importance.

The studies that are discussed above will result in the definition of effective design and functional standards, environmental and operational requirements, individual and group stability parameters, and authority and command structures. The results of these studies will ensure that crew members are physically and mentally able to perform required tasks and that systems, equipment, spacecraft, habitat, rovers, vehicles, tools, and operations are designed to promote safe and effective performance.

### 7.1.6 Human health and performance critical challenges and technology needs summary

Table 7-1 summarizes the human health and performance challenges and technology needs, while table 7-2 details the various human health and performance venues.

**Table 7-1.** Human Health and Performance Challenges

| **Current Knowledge or Capability Gaps** |
|---|
| • Determine the effect of long-term stowage in space on food.<br>• Obtain a better understanding of the effects of radiation beyond LEO on the immune system and microbial characteristics, and on overall chemical toxicity (determine whether radiation produces any synergistic, additive, chemical potentiation or chemical antagonism effects).<br>• Obtain a better understanding of the long-duration effects of space flight beyond LEO on the human immune system.<br>• Obtain a better understanding of changes that occur in microorganisms during extended missions beyond LEO.<br>• Obtain a better understanding of the environment of Mars.<br>• Develop a martian Dust Health Standard. |
| **Technology Needs** |
| Radiation Protection<br>• Develop advanced shielding technologies.<br>• Develop radioprotectants and pharmaceutical countermeasures against radiation.<br>• Develop strategies for individual-based risk assessment for crew selection.<br>Reduced-g Countermeasures<br>• Develop the capability to assess the time course of skeletal changes for periods of 6 months and longer.<br>• Develop pharmacotherapeutic monitoring and treatment technologies.<br>Habitability and Environmental Health<br>• Develop emerging technologies in food processing.<br>• Develop an automated acoustic monitoring system.<br>• Develop a technology that promotes autonomy, such as $H_2O$ remediation systems, contamination-resistant materials, and "smart" medical diagnostic systems.<br>• Develop a technology for monitoring the martian rover for dust (contingent on open or closed rover design).<br>Life Support Systems<br>• Closed-loop life support systems.<br>• Reliable, robust systems that require minimal crew support to operate and maintain.<br>• Systems that are capable of using locally generated consumables including $O_2$, $H_2O$, and buffer gases.<br>• Systems that operate in both zero-g (transits) and partial-g (surface) environments.<br>Human Factors<br>• Develop risk assessment and monitoring tools that passively detect individual stress and crew cohesion issues.<br>• Construct a Rest and Recreation Center to provide psychosocial support for living and working in space, and tailor it to transit time and surface time. |

**Table 7-2.** Human Health and Performance Testing Venues

| Venue | Radiation Protection | Reduced-g Countermeasures | Medical Care | Life Support | Human Factors |
|---|---|---|---|---|---|
| **A. Earth Surface** | • Accelerate research to determine the biological effects of radiation | • Bed rest and other analog environment studies of zero-g and reduced-g exposures | • Remote and isolated sites (Antarctica, Devon Island, NEEMO) for medical care training and verification | • Ground-based long-term system closure | • Remote and isolated sites (Antarctica, Devon Island, NEEMO) for psychological and psychosocial training and countermeasure verification |
| **B. Earth Atmosphere** | • N/A | • N/A | • N/A | • N/A | • N/A |
| **C. Earth Orbit** | • Shielding effectiveness<br>• Dosimetry and monitoring | • Long-duration exposure and countermeasure effectiveness | • Medical care techniques in remote environments | • System closure and performance<br>• System maintenance | • System design for long-duration remote operations<br>• Psychological and psychosocial evaluation and countermeasure verification |
| **D. Lunar Transit and Orbit** | • Space weather flight data from the Science Mission Directorate (SMD) Inner Heliosphere Sentinels<br>• Active dosimetry and monitoring<br>• Shielding effectiveness | • | • | • | • |
| **E. Lunar Surface** | • Shielding effectiveness<br>• Long-duration exposure<br>• Active dosimetry and monitoring | • Long-duration exposure and countermeasure effectiveness<br>• Hypo-gravity data and countermeasures | • Medical care techniques in remote environments | • System closure and performance<br>• System operation and maintenance | • System design for long-duration remote operations<br>• Psychological and psychosocial evaluation and countermeasure verification |
| **F. Deep Space Transit** | • | • | • | • | • |
| **G. Mars Orbit** | • | • | • | • | • |
| **H. Mars Atmosphere** | • | • | • | • | • |
| **I. Mars Surface** | • Robotic characterization of the surface radiation environment<br>• Data to validate models and characterize landing sites | • Operational long-duration exposure and countermeasure effectiveness assessment<br>• Operational hypo-g data collection and countermeasures assessments | • Robotic characterization of surface hazards | • Ability to use locally produced consumables | • Operational psychological and psychosocial countermeasure effectiveness assessment |

## *7.2 Space Transportation*

### 7.2.1 Earth-to-orbit transportation

The heavy-lift launch vehicle that is known as Ares V is currently under development within the NASA CxP. Ares V is the lynch-pin to placing payloads of large mass and volume into LEO. Of the obstacles that the program will face, technological challenges are seen as one of the most unpredictable. Understanding this, the Ares V lunar campaign reference vehicle is specifically designed to have a minimum of low-TRL systems while meeting the requirements of the architecture.

Some challenges will inevitably arise out of the sheer physical scale of the Ares V, which in many respects surpasses the Saturn V. The payloads to be integrated will be factors of two to four larger than any previously attempted. Ground handling and operations of these payloads will need to be carefully studied and understood. Support operations will likewise be challenged in most aspects, as the Ares V will require more fuel (cryogenic), support gasses (He), transportation needs, and larger boosters than have previously been supplied, which will likely lead to a more complex support integration process.

The challenges of scale will increase in complexity if the vehicle and its supporting infrastructure are upgraded for the demanding launch requirements of the Mars campaign. Orbital assemblies requiring multiple launches will need either a much higher rate of launch than has been attempted with vehicles of this magnitude or very-long-duration loiter capabilities.

Keeping such scale affordable, even in relative terms, may require rethinking some of the traditional assumptions, and some of the reference ground rules for the Lunar Campaign Trade studies will need to be conducted at several levels to determine the best way to reconfigure and equip the Ares V for use in the Mars campaign.

#### *7.2.1.1 Potential upgrades to lunar campaign*

The 2007 Ares V analysis study identified a number of potential upgrades to the Ares V to allow it to better meet or, preferably, exceed the requirements for the lunar campaign. These upgrades are also applicable to the Mars campaign, although the payload definitions are quite different and the corresponding deltas are also different. Upgrades that were identified include using composites on the core, RSRBs, an EDS, addition of a sixth RS-68, regenerative RS-68s, and hydroxy-terminated polybutadiene (HTPB) RSRBs with optimized nozzles. These were shown to each provide 1 to 5 t of payload increase to TLI and to the LEO assembly orbit (220 nmi/772-km circular).

Composite structures figure prominently in identified possible changes to the structure. Although more complex and not as fully understood as metals, properly constructed composites can offer significant decreases in weight for the same strength. Composites can be used to replace metal in many of the designed structural components, with the primary candidates being the unpressurized structures followed by the fuel tanks. While the core stands to lose the most per-stage weight through use of composites, the EDS has the most increase on payload.

Composites introduce some challenges to the program. The properties of composites are not as well known or uniform as metals, so much more extensive testing will be required to work with and qualify the pieces. Composite manufacture, which is from metal manufacturing, may require retooling for facilities that were previously used to work metal components. With the notable benefits composites bring to bear, however, further investigation of the tradeoffs are certainly warranted.

Regenerative RS-68s would decrease the weight of the engine by reducing the ablative layers on the nozzle. Reclaiming part of the waste heat to preheat the fuel would increase the efficiency by some amount, and aid combustion. Regenerative nozzles do, however, have the potential to increase risks with additional mechanical complexity, and will probably require additional testing.

In addition to composite cases, the fuel in the RSRBs could potentially be changed to the HTPB fuel, which is more energetic and can create a higher internal pressure. This pressure, in turn, creates higher thrust. Alliant Techsystems Inc. (ATK), which is the supplier of the RSRBs, also believes that it can optimize the RSRB nozzles to obtain better performance. Together, these improvements represent a increase in payload. Increasing the maximum operating pressure does put higher loads on the casing, and would increase both the cost of the booster and the amount of initial testing required to qualify it.

Other, more minor improvements can be realized over time with improvements in materials and components, and more solid data on the actual vehicle loads.

#### *7.2.1.2 Potential alternate configurations*

Of the configurations that were studied during 2007, two families of alternate configurations are of particular interest for potential application to the Mars campaign. One line of configurations seeks to add extra capacity by additional boosters.; the other seeks to optimize staging points by adding a third stage, thereby creating a lower thrust dedicated EDS (figure 7-2). The first potential alternate configuration (study configuration 45.0.05) uses additional “Delta IV –Type” liquid boosters strapped onto the sides of the Ares V core to gain additional thrust at lift-off. This configuration has the benefit of using a known stage that does not have as much gross mass penalty as an additional solid booster. The configuration may introduce complications in transport and launch pad requirements due to the perpendicular orientations of the booster pairs about the core.

The second configuration that was identified is study configuration 45.0.13, which adds a second pair of RSRBs, which would be placed in a "dogbone" set of pairs on either side of the vehicle to minimize the difference between this configuration and the lunar campaign Ares V. The third possible configuration, which is taken from study configuration 47.0.100, omits the third stage of this vehicle. The Mars application of this vehicle is equivalent to a Mars-redesigned second stage that is optimized not for TLI injection but for LEO assembly orbit injection. A significant drawback of configuration 47.0.100 is its small payload envelope. By shrinking the size of the second stage rather than offloading EDS propellant and using a shorter shroud nosecone, it should be possible to increase the size of the cylindrical payload envelope that is required to reduce the number of Ares V launches needed for delivery and assembly of cargo and crewed MTV components.

All of the alternate configurations would require, through additional loads and masses, the redesign of some primary structures on the baseline Ares V. However, with proper planning during the design of the baseline, it should be possible to reduce the impacts of these changes by having them introduced as planned block upgrades.

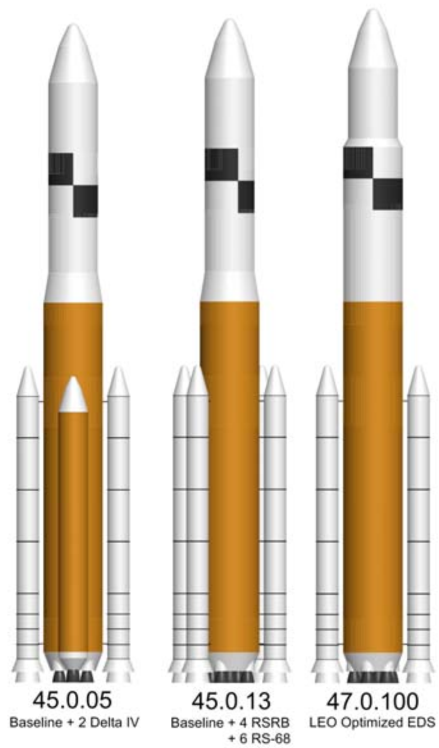



**Figure 7-2.** Potential alternate Ares V configurations.

### *7.2.2 Advanced chemical propulsion*

Chemical transportation systems have been used for decades. They are well understood and have substantial heritage as well as existing test facilities. For human exploration missions beyond LEO, however, this method of propulsion has drawbacks even considering the advances in fuel types and efficiencies that could come with more research funding. The biggest drawback is mass. Carrying a large volume of fuel to distant destinations, such as Mars, and back propagates all the way back to the initial mass to LEO requirements. Advanced chemical propulsion technologies are best suited for more near-Earth mission concepts.

#### *7.2.2.1 Liquid oxygen/liquid hydrogen propulsion*

The propulsion options that were considered for the chemically propelled human Mars mission include the standard RL10-B2 engine on the in-space stages. This engine uses a fuel and oxidizer combination of $LH_2$ and $LO_2$, respectively, at a mixture ratio (MR) of 5.88. At this MR, the assumed thrust is 24,750 lbf with an $I_{sp}$ of 462.2 seconds (value used for all performance computations).

While the RL10-B2 does not represent an “advanced chemical” technology, many challenges must be resolved before the engine can be applied to a human Mars mission. A major unknown is whether the engine turbopumps will start after long periods of inactivity. For example, the TEI engines are idle for approximately 2 years in the deep-space environment. Therefore, extensive testing is required to ensure that the engines will start under these conditions. Thermal cyclic testing is also required to guarantee that the engine nozzle material expansion and contraction are within acceptable parameters.

One engine modification, which was not used in this study, that may be applicable to the mission is operating the RL10-B2 at a mixture ratio other than 5.88. A recent study performed at MSFC (Shana, 2005[7]) analyzed the engine performance over a wide range of MR. The results were obtained using Propulsion Sizing, Thermal, Accountability, and Weight Relationship First Order Modeling Tool (P-STAR). Operating at an MR of 5.0 may result in about a 5-second increase in $I_{sp}$ compared to the standard operation at 5.88. This increase in $I_{sp}$ would lead to substantial propellant savings. However, actual hardware testing is required to recertify the engine at this off-standard operating condition.

#### *7.2.2.2 Liquid oxygen/methane propulsion*

The technology that was proposed for the Mars descent and/or ascent propulsion systems involves an $LO_2/CH_4$ propellant combination. Methane was chosen as the fuel so that ISRU can be used to produce the required ascent propellant at Mars rather than carrying the fuel from Earth. This is not a new concept, having been thoroughly analyzed in previous Mars design reference missions. These missions used pump-fed engines with an MR of 3.5 and thrust values ranging from 15,000 to 22,000 lbf with $I_{sp}$’s of 377 to 379 seconds. Currently, no pump-fed $LO_2/CH_4$ engines are in production and only pressure-fed engines are in development. Much research and testing are required to produce a highly reliable pump-fed engine that can meet the human Mars mission requirements.

For this study, P-STAR, which was mentioned above, was used to generate performance values for this propellant combination in an expander cycle engine (MSFC, 2007[8]). The chamber pressure was set to 900 psia and the nozzle area ratio was assumed to be 200. These performance values and engine sizes are consistent with those in a recent $LO_2/CH_4$ expander cycle study (Crocker, 1998[9]), which involved a 20,000-lbf engine with a mass of 335 lbm. This study quoted an MR of 3.5 and $I_{sp}$ of 369.1 seconds for this engine. For the current mission, the MR of 3.5 is also assumed with the P-STAR-predicted $I_{sp}$ of 369.5 seconds.

Like the $LO_2/LH_2$ case that was mentioned above, the $LO_2/CH_4$ engines face the challenge of having to start after sitting idle for an extended period of time (e.g., on the martian surface). Pressure-fed engines have been considered to alleviate this concern. Without the rotating turbo-machinery, pressure-fed engines are much simpler and more reliable than their pump-fed counterparts. However, while the engine dry mass for the pressure-fed engine is lower than that of the pump-fed engine, the overall feed system mass is much higher due to the higher pressure that must be maintained in the propellant tanks (250 psia vs. 50 psia). The required He pressurant (and tanks) is also greater. This problem is made even worse due to the lower (in general) $I_{sp}$ and corresponding higher propellant requirement of the pressure-fed technology. These factors result in much lower payloads that can be delivered. Therefore, pump-fed engines are chosen for $CH_4$ engines in the current mission. The work and testing that are required to verify that the $LO_2/LH_2$ engine starts after long idle times will, it is hoped, also solve any issues with the $LO_2/CH_4$ start capability.

### 7.2.3 Nuclear thermal propulsion

NTP technology was demonstrated to high-TRL levels during the Rover/NERVA programs (Koeing, 1986[10]). A variety of fuel forms were developed, and a broad range of different thrust-class engines were ground-tested at the Nevada Test Site (NTS). Near the end of the program, “open-air” testing of the engines was replaced by “contained” testing using an effluent treatment system (ETS) to process the $H_2$ exhaust. While the continued development of chemical propulsion systems has led to performance advances in the non-nuclear engine and stage component areas (e.g., $LH_2$ turbopumps, regenerative-cooled nozzles, and lightweight cryogenic tanks) that would be required for an NTP MTV, further work and funding is required in

[7]Diez, Shana, "Expander Engine Mixture Ratio Trades," Internal Analysis Report, MSFC/ER22, January 2005.
[8]MSFC, LOX/LCH4 P-STAR analysis performed by Joe Leahy (MSFC/ER21), September, 2007.
[9]Crocker, Andrew M. and Peery, Steven D., "System Sensitivity Studies of a LOX/Methane Expander Cycle Rocket Engine," AIAA-1998-3674.
[10]D.R. Koeing, “Experience Gained from the Space Nuclear Rocket Programs (Rover/NERVA),” LA-10062-H, Los Alamos National Laboratory (May 1986).

the nuclear area to: (1) recapture, improve, and mature the candidate NTP fuel types; (2) develop state-of-the-art (SOTA) engine designs; and (3) determine the most viable, cost-effective approach for engine ground testing.

This section of the Mars DRA 5.0 report reviews the accomplishments of the Rover/NERVA programs, the technology SOTA of nuclear and non-nuclear engine/stage components, and the two primary facility options for “contained” ground testing that are being considered by the DOE. Discussed also are the key elements of a representative NTP development schedule, which would be produced jointly by NASA and DOE, and its compatibility with an initial human Mars landing mission in the 2031–2033 timeframe.

#### *7.2.3.1 Nuclear thermal propulsion accomplishments and technology state-of-the-art*

The viability of NTP was demonstrated and its technology validated to approximately TRL 5–6 during the Rover/NERVA nuclear rocket programs (~1955–1972). Twenty rocket reactors were designed, built, and ground-tested in integrated reactor/ engine tests that demonstrated: (1) a wide range of thrust levels (~25, 50, 75, and 250 $klb_f$); (2) high-temperature nuclear fuels that provided $H_2$ exhaust temperatures of approximately 2,550 K; (3) sustained engine operation (over 62 minutes for a single burn on the NRX-A6 engine); (4) accumulated lifetime; and (5) restart capability (>2 hours during 28 startup and shutdown cycles) during the NRX-XE experimental engine tests that were conducted in 1969. The total program cost for the 20 engines, Test Cells A and C, the Engine Test Stand, and the engine maintenance assembly/disassembly (EMAD) facility that were used for engine buildup, remote teardown, and post-irradiation examination was about $1.4B (an investment of ~$7.9B in 2008 dollars).
The baseline fuel form that was used in Rover/NERVA was CVD-coated $UC_2$ particle fuel that was immersed in graphite and extruded into hexagonal FEs that were approximately 0.75 in. across the flats and approximately 52 in. long. Zirconium carbide was used to coat both the particles and the FE surfaces to prevent erosion of the graphite by the $H_2$ propellant. Higher-performance ($UC_2$-ZrC) in graphite “composite” fuel was also produced in the same FE geometry. Electrically heated composite fuel was tested by Westinghouse in hot $H_2$ at 2700 K for about 600 minutes (ten 1-hour cycles). Subsequent nuclear tests using composite fuel were conducted in the nuclear furnace in 1972. During these tests, the composite FEs underwent significant irradiation testing at peak power densities of approximately 4,500 to 5,000 $MW/m^3$ and $H_2$ exhaust temperatures of approximately 2450 K for about 2 hours. Composite fuel is expected to perform satisfactorily for approximately 2 to 4 hours at exhaust temperatures up to approximately 2,800 K (Burkes, et al., 2007[11]). Composite fuel also has a higher coefficient of thermal expansion that more closely matches that of its ZrC coating, thus helping to reduce coating cracks and $H_2$ erosion of the graphite.

An attractive alternative NTP fuel is a ceramic-metallic “cermet” fuel consisting of uranium dioxide ($UO_2$) in a tungsten (W) metal matrix material ($UO_2$-W) with W-alloy cladding. As the primary backup to the carbide-based Rover/NERVA fuels, cermet fuel underwent extensive nuclear/non-nuclear testing in the 1960s under the GE-710 and Argonne National Laboratory (ANL) nuclear rocket programs, but it was not developed into an operational engine. Non-nuclear, hot $H_2$ exposure tests at temperatures up to 3,000 K (with cycling) established the viability of cermet fuel for NTP use. Irradiation tests, which were conducted under both transient and steady-state operating conditions, further indicated a robust fuel with potential for high burn-up and improved fission product retention using its W-alloy cladding. Based on a substantive experimental database, extensive supporting documentation, and anticipated performance potential, both composite and cermet fuels have the potential to meet the operational requirements for NTP systems, and have been recommended by both DOE and NASA as the two logical choices for consideration in a future NTP development effort.

In contrast to NTP fuel development, which has seen only minimal investment since the Rover/NERVA programs were terminated in January 1973, non-nuclear engine components have undergone significant improvements. High-pressure $LH_2$ turbopumps, high heat flux, regenerative-cooled nozzles, and lightweight, radiation-cooled nozzle skirt extensions have been developed and are used on today’s SOTA $LO_2/LH_2$ chemical engines, all of which have flown in space. Existing chemical rocket hardware can potentially be adapted and/or operated at de-rated conditions for use on the 15–25-$klb_f$-thrust class NTRs that are being considered today. Examples of these include the RL 60 $LH_2$ turbopump design that can be adapted to supply the required approximately 7.6–12.6 kg/s $LH_2$ flow rates; also the hot plenum chamber and regenerative-cooled nozzle components can be operated at lower pressure, temperature and heat flux levels than their chemical engine counterpart. The size of a 300:1 radiation-cooled nozzle skirt extension for a 15–25 $klb_f$ NTR is either smaller or comparable in size to that used on the current RL 10B-2 engine (figure 7-3).

[11]D. Burkes, D. Wachs, J. Werner (DOE/INL), G. Bell, J. Miller, P. Papano (DOE/ ORNL), and S. Borowski (NASA/GRC), “The Rationale and Justification for Selection of Carbide “Composite” and Ceramic Metallic “Cermet” NTP Fuel Options,” a Joint DOE / NASA White Paper for NASA HQ (May 2007).

Lastly, future NTP Mars transfer stages should be able to use the same approximately 10.0-m diameter Al/Li $LH_2$ tank technology that is being considered for the Ares-V "core stage" but at half the length. Typical NTP $LH_2$ tank lengths vary from approximately 12.5 to 20 m depending on their application. Leveraging existing manufacturing infrastructure at the Michoud facility should help reduce both stage development and recurring costs.

#### *7.2.3.2 Ground test facility options and development challenges*

Ground test demonstrations of NTP components, subsystems, and full-scale systems are a necessary part of flight system qualification. Current environmental protection (National Environmental Policy Act of 1969 (NEPA)) standards prohibit any significant release of radioactive particulates into the air from a nuclear test facility. The two primary options (Hill, et al., 2007[12]) for testing NTP engines are: (1) an above-ground contained test facility (CTF) that is outfitted with an ETS to scrub and filter the $H_2$ exhaust of lo- level fission product gases; and (2) the underground geological soil and porous rock (alluvium) at the NTS, which can be used for in-situ capture, holdup, and subsequent filtration of the exhaust – the so-called SAFE [subsurface active filtration of exhaust] concept (Howe, et al., 2001[13]). In the SAFE concept, which was proposed by Los Alamos National Laboratory (LANL) and NTS scientists, the engine exhaust would be injected into currently existing vertical boreholes at the NTS.

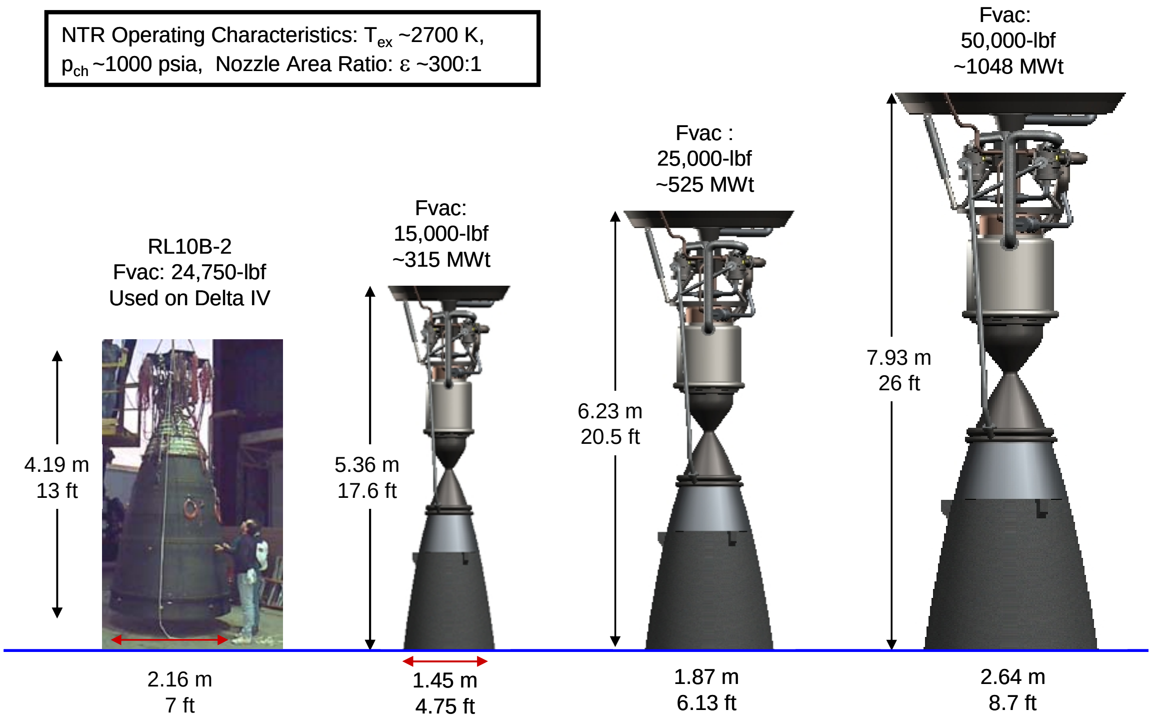



**Figure 7-3.** Size comparison of RL 10B-2 chemical engine and different thrust NTR engines.

The design and technical feasibility of an above-ground ETS was successfully demonstrated during the Rover program with the nuclear furnace fuel element test reactor. After exiting the nuclear furnace reactor core, the $H_2$ exhaust was sprayed with $H_2O$ to cool the gas and remove any particulates from the exhaust stream. A heat exchanger was then used to reduce the temperature further before the effluent was passed through a silica gel bed to remove the $H_2O$ and dissolved fission products. The exhaust gas was then passed through $N_2$-cooled, activated charcoal filter beds to remove the noble gases (Krypton and Xenon). The exiting $H_2$ stream contained no detectable fission products and was subsequently released to the atmosphere. At an operating power level of approximately 44 $MW_t$, the nuclear furnace represents an approximately one-tenth scale

[12] T.J. Hill and J.E. Werner, "Ground Testing of Space Nuclear Thermal Propulsion Systems: A Review of Potential Ground Test Options and a Recommendation," INL/EXT-07-13339, Idaho National Laboratory (September 2007).
[13] S.D. Howe, B. Travis and D.K. Zerkle, "SAFE Testing of Nuclear Rockets," *Journal of Propulsion and Power*, Vol. 17, No. 3, pgs. 534-539 (2001).

demonstration of the ETS that would be needed for use with the 25-$klb_f$ -class NTR engine that was baselined for the Mars DRA 5.0 study.

With its potential for reduced complexity and cost, the SAFE concept appears to be an attractive alternative to above-ground testing using an engineered ETS. The viability of SAFE, however, has only been demonstrated analytically in studies that were conducted by LANL and, recently, in a DOE-funded study by the Desert Research Institute (DRI) (Decker, et al., 2007[14]). LANL and DRI have both recommended follow-on non-nuclear, subscale testing to further validate the feasibility of the SAFE concept. DRI proposes using Ar to simulate the $H_2$ effluent from the NTP system. Both elevated and ambient temperature Ar gas tests would be conducted to simulate engine test and post-test conditions. The Ar would also be spiked with a radioactive Krypton-85 tracer to permit monitoring of gas permeation through the alluvial soil. If the test results are positive, the next step would be testing at the NTS using an RL-10 engine to obtain more prototypic test data on $H_2O$ injection for exhaust gas cool-down, borehole pressurization and equalization in the surrounding alluvial soil, and gas migration through the soil again using a tracer gas. Estimated costs for initial tests are approximately 1–3M$ with subsequent RL-10 engine tests in the 5–10M$ range.

The Idaho National Laboratory (INL) and NTS are two likely and “complementary” sites for future NTP ground testing. Positive factors for NTS include the suitability of the alluvial soil and the availability of boreholes for SAFE testing. The high-security device assembly facility (DAF), which is also located at NTS, can be used for engine assembly (attachment of $LH_2$ turbopump and nozzle assemblies to the reactor core/pressure vessel) and “zero-power” critical tests prior to full-power ground testing. Following completion of an engine test series, the radioactive NTP reactor would have to be shipped in a DOE transfer container to the INL materials and fuel complex (MFC) for disassembly, post-irradiation examination (PIE), and disposal preparation at the hot fuel examination facility (HFEF) because this capability no longer exists at NTS. Similarly, soil characteristics at INL do not support the SAFE testing approach, so a new ground test facility (GTF) and ETS would need to be constructed. Further study of both GTF options and “proof-of-concept” testing of the SAFE concept by DOE are required before final selection of the preferred GTF approach and site location. Ground testing at either site will be limited to single-engine tests with a truncated nozzle, not complete vehicle system verification tests in a simulated vacuum space environment. Flight qualification of an entire stage, using clustered engines, will be done on precursor cargo missions to the moon initially, then on Mars cargo flights before flying humans to Mars.

#### *7.2.3.3 Key elements of nuclear thermal propulsion development schedule*

Figure 7-4 shows the key elements of a joint NASA/DOE NTP development schedule that is compatible with initial human flights to Mars in the 2031–2033 timeframe. They include (1) focused nuclear and (2) non-nuclear technology development; (3) GTF concept selection and construction, and (4) NTP engine and stage design, development, and testing. Each of these four primary areas includes an initial approximately 5-year “preparation phase”. In the nuclear technology area, Rover/NERVA “composite” fuel samples and element segments, using improved CVD coatings, will be fabricated and tested in a non-nuclear, hot $H_2$ environment initially, then undergo irradiation testing in the advanced test reactor (ATR) at INL to validate and recertify fuel performance. Similar fabrication and testing will be conducted on $UO_2$-W “cermet” fuel and cladding materials to mature them to the point where an informed “down-select” decision to a primary fuel type and element geometry can be made circa 2015.

[14]D. Decker, C.A. Cooper, R. Jacobson, P. Oberlander and D. Shafer, “Preliminary Numerical Modeling and Sub-scale Experimental Design of a Nuclear Rocket Test Facility with Exhaust Sequestration at the Nevada Test Site,” *DHS Publication No. 41238*, Desert Research Institute (September 2007).

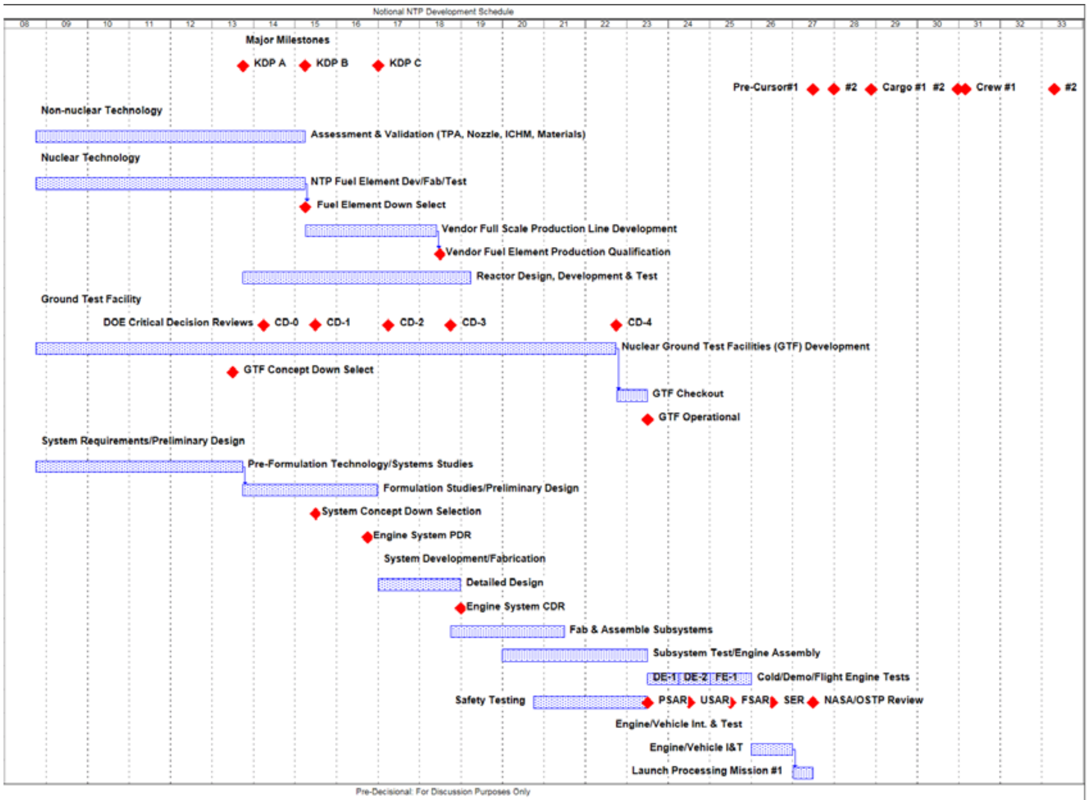



**Figure 7-4.** Representative NASA/DOE NTP development schedule.

Development and validation of the non-nuclear engine and stage components are not expected to be pacing elements in the NTP development schedule. Many of these components and subsystems already exist and have flown in space, as discussed previously. The main technical issue is their tolerance to a radiation environment for prolonged periods and whether selective use of particular types of materials may be required. Representative fabrication materials for NTP TPAs, nozzles, controls, valves, and instrumentation have already been identified and demonstrated during duration testing of the NRX-XE, which was conducted at the NTS in 1969.

Continued refinement and updating of the Mars DRA and mission payloads will be needed to better define the NTR engine and stage size, and operating characteristics. Requirements definition and preliminary "in-house" design work are performed prior to KDP A [Key Decision Point/Phase A], which starts in late 2013. Engine design work uses SOTA numerical models to determine reactor neutronics and energy deposition within the reactor subsystem (Schnitzler, et al., 2007[15]), provide thermal, fluid and stress analysis of fuel element geometries (Stewart, et al., 2007[16]), and predict engine operating characteristics and overall mass (Fittje, 2007[17]). Phase B begins in 2015, following an approximately 18-month Phase A period. Preliminary engine and stage design activities with industry are initiated and a PDR occurs in late 2016. "Authority to proceed" with Phase C/D occurs in 2017, followed by the CDR in circa 2019.

The GTF is the long lead item in the NTP development schedule. Assuming that initial engine tests begin in 2023, and that there is a typical 5-year design, construction, and checkout period, the DOE needs to have its GTF budget request submitted by 2014 (Critical Decision-0 (CD-0)) following concept down-select in 2013. CD-0 allows the start of conceptual design and

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[15]B.G. Schnitzler and S.K. Borowski, "Neutronics Models and Analysis of Small Nuclear Rocket Engine (SNRE)," AIAA-2007-5618, 43rd Joint Propulsion Conference, Cincinnati, OH (July 8-11, 2007).
[16]M.E. Stewart and B.G. Schnitzler, "Thermal Hydraulic Simulations of NTP Reactor Core," AIAA-2007-5619, 43rd Joint Propulsion Conference, Cincinnati, OH (July 8-11, 2007).
[17]J.E. Fittje, "Upgrades to the NESS (Nuclear Engine System Simulation) Code," AIAA-2007-5620, 43rd Joint Propulsion Conference, Cincinnati, OH (July 8-11, 2007).

the preparation of appropriate NEPA documentation, along with the appropriate permits that are obtained from the state with regulatory authority. This information provides the basis for continuing with detailed design on the GTF. Approval to begin construction on the GTF occurs in 2018 (Critical Decision-3 (CD-3)).

Once the GTF becomes operational, ground test operations begin with the first demonstration engine (DE-1) in mid-2023. A second engine (DE-2) is tested in 2024, followed by ground testing of the flight-type engine (FE-1) in mid-2025. If all then goes to plan, the first Mars NTP precursor mission to the moon would be launched in mid-2027 with a second lunar cargo or crewed flight occurring in early 2028. Components for the two Mars cargo vehicles would then be launched, assembled via autonomous rendezvous and docking, and depart LEO in late 2028. The crewed mission would depart LEO and land on Mars in 2031. Additional development and/or test time is available if the first human Mars landing is delayed until 2033, which is the “minimum-energy” opportunity over the 15-year cycle. The preceding cargo flights would then depart LEO in 2031.

#### *7.2.3.4 Summary and recommendations*

NTP is a proven technology that has the potential to enable future human Mars missions with reasonable IMLEO requirements and credible numbers of Ares V launches. However, to recapture, mature, and flight-qualify NTP systems in time to support future cargo and crewed Mars missions in the post-2030 timeframe, meaningful, sustained investments need to start in the next several years. These investments will be aimed at: (1) establishing firm NTP engine system requirements using updated Mars mission analysis and payload estimates; (2) recapturing “composite” Rover/NERVA fuel element technology, and maturing $UO_2$ in W metal “cermet” fuel technology; (3) performing high-fidelity modeling, design, and engineering of candidate engine systems; (4) preparing the necessary test facilities; and (5) conducting the required nuclear/non-nuclear demonstration tests of NTP fuels, components, and subsystems in preparation for “contained” full-scale ground testing of demonstration followed by flight-type engines. Assuming approximately 5 years of technology preparation prior to authority to proceed (ATP) and an approximately 10-year development phase, NTP flight testing can begin in the late 2020s in time to support initial human Mars flights in the 2031–2033 timeframe.

### 7.2.4 Electric propulsion

Electric propulsion is another class of in-space transportation that has benefits for human exploration. Electric propulsion concepts use solar or nuclear power to accelerate propellant to higher exit velocities than those that could be achieved from a chemical reaction. Such systems have the advantage of a high $I_{sp}$ system that maximizes engine efficiency, usually at the expense of thrust. Solar and nuclear electric propulsion offers very high $I_{sp}$ (2,000 to 10,000 seconds), which results in a reduction in propellant mass requirements. By nature, low-thrust electric propulsion systems offer increased mission flexibility compared to high-thrust chemical propulsion systems (i.e., unconstrained by launch windows). At power levels of 5 mW, electric propulsion has the potential to reduce initial mass to LEO for Mars cargo missions. For the cargo missions, trip time to Mars will be relatively long compared to chemical propulsion systems. At considerably higher power levels (40 to 200 mW), nuclear electric propulsion can potentially reduce trip time for piloted mission applications. Megawatt-level nuclear power systems with low specific mass will be required. For the higher-power nuclear electric propulsion concepts (40 to 200 mW), a new reactor development program will be required, as well as development of energy conversion, heat rejection, and power management and distribution techniques to achieve low-power system-specific masses.

One common requirement for all electric propulsion systems is a substantial supply of either solar- or nuclear-generated power. Solar power has the advantages of safety and technology readiness, as nuclear systems require extensive shielding measures and most nuclear technology development programs have been significantly scaled back. Large PVAs must be built and deployed to power the systems for a trip to Mars or near-Earth locations, while none of the shielding requirements that all nuclear options carry would be of concern. Research into how to deploy and orient these structures and how to get maximum efficiency are of prime importance. However, solar arrays must be made larger for missions away from Earth’s neighborhood as the sun’s energy density decreases with distance. For the category of in-space transportation, however, the FOMs clearly show the importance of developing solar electric propulsion. The technologies that are required for large, megawatt-class nuclear and solar electric transfer vehicles are at a lower stage of technical and operational maturity and, thus, were not addressed as part of the 2007 DRA 5.0 activity.

### 7.2.5 Entry, descent, and landing

#### *7.2.5.1 Mars entry descent and landing system technology (Engelund & Manning)*

The current NASA ability to land robotic payloads on the surface of Mars is largely reliant on the EDL technology set that was developed during the Mars Viking Program in the late 1960s and early 1970s. The NASA flagship 2009 Mars mission, the MSL, has reached the landed payload mass limit capability, approximately 1 t, using the Viking-based technology set, which includes the blunt-body 70-degree sphere cone aeroshell, the SLA 561-V TPS material, and the supersonic disk-gap-band parachute system. In fact, as a result of the high mass, larger ballistic coefficient, and correspondingly higher heating environment entry of the MSL system, the MSL project was forced to make a very late design change (post CDR) to the forebody TPS material, replacing the Viking-heritage SLA-561V material with a PICA material that was currently being developed for the Orion Block I LEO and Block II lunar return TPS. The 1-t landed mass capability of the MSL EDL system is a factor of 40 below what will likely be required to achieve a human-scale Mars mission. As NASA strives to land larger mass robotic missions – e.g., MSR – and looks forward to human missions to Mars, additional EDL technologies must be identified and developed to the point that they are viable candidates for robotic and/or human mission sets. In addition, technologies that would be adequate for the lower range of masses may not be applicable or scalable to the very large landed mass EDL systems that would be associated with human-scale missions. Because of the limited NASA technology budget, it is imperative that the minimum-cost technologies that would be required for the entire range of desired landed masses that provides the required reliability be identified.

The very low atmospheric density at Mars prevents the use of traditional terrestrial aerodynamic decelerators as a means to attain subsonic velocities for landing, as is done on Earth. The challenges that are associated with the development of a human-rated, high-mass (100+ t) Mars entry system remain large. While there is considerable uncertainty in the ultimate outcome of human-scale landing system designs, several technology options provide candidate pathways. Certain combinations of aerocapture and entry, descent, and landing (AEDL) technologies that may be achievable and that would result in robust performance and acceptable risk architectures have been identified and deserve further study. Several of these are discussed below. While what appears below is not an exhaustive list, these components could include, but are certainly not limited to, the following technology and architecture options:

**Slender body entry and decelerator options**: Slender body hypersonic lifting entry systems include the general class of bi-conics, tri-conics, ellipsleds, and winged systems (e.g., shuttle) (figure 7-5). Like the shuttle, these systems achieve substantial lift in the hypersonic flight regime by flying at high angles of attack. However, even with shuttle-like wings, due to the very low atmosphere density on Mars, adding lift alone does not allow high-ballistic-number vehicles to attain subsonic terminal speeds prior to impact with the ground. Fortunately, the amount of L/D that is needed for precision targeting, which has been demonstrated in previous studies, is on the order of 0.18; this enables these entry systems to use the “excess” lift to provide additional altitude so that some other form of high-Mach supersonic decelerator may be safely used to bring the vehicle to subsonic speeds at reasonable altitudes.

Provided the length-to-width ratio of slender body vehicle is above about 2.5, these systems present greater drag area than do blunt bodies of similar diameter. Another advantage is the very high volume and low packing density that these systems provide. For landed systems that have complex form factors or long tendril-like components, slender body designs provide excellent alternatives. It is possible that the slender body structure and the TPS would also double as a launch shroud when launched from Earth, thus saving considerable mass to LEO.

These slender body aeroshell systems have several potential drawbacks, however. In particular, the payload load-paths for launch vs. entry are essentially orthogonal. This requires additional structural mass additions to the design. Likewise, unless the supersonic decelerator system is designed to align accordingly, the supersonic deceleration loads may be in yet a third direction. The general problem of how to perform the supersonic deceleration and then configure and expediently extract the propulsive lander from within the slender aeroshell is, so far, an unsolved problem. The aeroshell mass that would be required to support the TPS, aero, and launch loads also make these systems quite heavy. However, there are multiple pathway options that are worthy of pursuit.

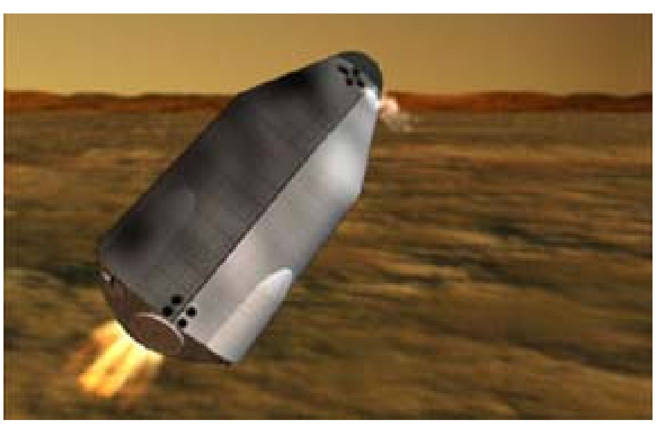

**Figure 7-5.** Candidate slender body aerocapture and entry, descent, and landing aeroshell design.

For example, a very large longitudinal, supersonically deployed, inflatable aerodynamic decelerator could be configured so that the inflation and deceleration loads are in parallel with the entry loads (figure 7-6). This inflatable decelerator system would be sized to bring the entire system from Mach 4 to below Mach 1 at approximately 5 km above the landing site surface. Mach 4 or Mach 5 is considered slow enough that aero-heating concerns are few. If the inflatable decelerator system provides a sufficiently low ballistic number for the system (e.g., < 50 kg/m$^2$), the "bottom half" of the slender body aeroshell could then be safely released above Mach 1 (as presumably it, and its TPS, would have a sufficiently high ballistic number to achieve a clean separation with no re-contact). Likewise, at an appropriate altitude (e.g., 1.5 km), the lander would also be released, perform propulsive avoidance maneuver and terminal descent deceleration, and proceed to the target landing location.

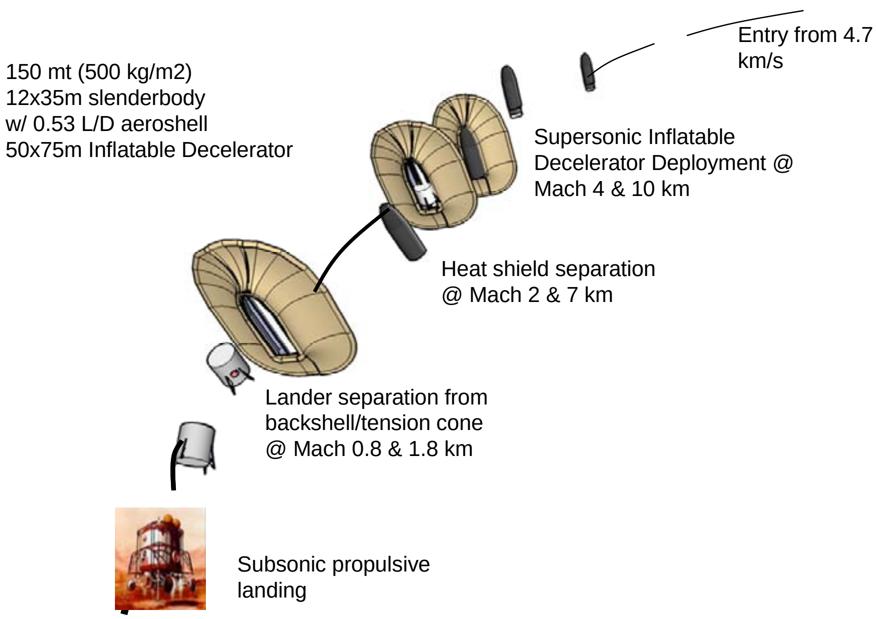


**Figure 7-6.** Slender body and inflatable decelerator concept.

These types of inflatable decelerator systems would be quite large; therefore, to avoid complex dynamic interactions, they would need to be inflated and structurally stiffened in seconds. Significant forward work and research and development into the fundamental deployment strategies and details must be made before the supersonic inflation issues can be considered mature enough for consideration in an operational robotic or human-scale mission architecture.

In contrast, a supersonic retro-propulsion system (as baselined in this DRA 5.0 study) could be used in lieu of the deployable aerodynamic decelerator options and directed counter to the velocity vector through outlets that would be blown out of the "bottom" of the slender body aeroshell. In this case, it remains unclear how the lander would be extracted from the aeroshell prior to landing. Subsonic separation aids such as clustered ring sail parachutes could be employed to pull the lander out the back of the aeroshell, however, it may be that the aeroshell would be retained all the way to the surface and the supersonic retro-propulsion nozzles would then double as throttled terminal descent engines. Detailed models and fundamental knowledge for both classes of aerodynamic and propulsive-assisted systems for a slender body entry aeroshell system would be required such that appropriate cost, mass, performance, and reliability trades may be considered.

**Blunt body entry and descent options**: Blunt body hypersonic entry systems include: lifting fixed blunt bodies (such as the MSL), pre-deployed blunt body entry heat shields, and hypersonic inflatable blunt aerodynamic decelerators. While these systems have low lift (typically 0.18 to 0.2 L/D ratios), they have sufficient L/D to perform precision guidance for landing site targeting.

These systems have some advantages over slender body designs; most notably, the launch, entry, and even supersonic decelerator load paths are all essentially in the same direction, allowing very efficient primary structure mass fractions. The lower drag area also reduces the surface area of the required TPS. Another advantage is that these entry designs are “in family” with other Earth and Mars entry system designs (i.e., Apollo, Orion, Viking, Pathfinder, MER, Phoenix, and MSL) and are, thus, natural extensions of these systems. Finally, like the slender body aeroshell (as well as the Apollo and Orion design), the aft body of the Mars entry system can serve as a forward launch shroud, thereby saving mass to LEO. These benefits may result in superior entry mass-to-useful landed payload mass fractions for blunt body entry systems to those for slender body entry systems.

A very significant downside for rigid, non-deployed blunt body heat shields is that the diameter limitation that would be imposed by the launch vehicle constrains the drag area of the heat shield (for slender bodies, it is the combination of diameter and length). An Ares V launch vehicle is expected to allow a maximum OML diameter of about 10 m. The entry ballistic number (and, hence, the entry mass) would then limited so that reasonable altitude constraints for the deployment of any supersonic decelerator can be maintained. This limit, and its implications on maximum useful landed payload mass, remains to be explored in detail but is expected to be worse than slender body entry systems. It may be that, even with superior mass fractions, rigid non-deployed blunt body systems may be unable to deliver a six-crew Mars habitat or a MAV in a single landing. Further work is required to determine the magnitude and implications of these constraints.

The ballistic number (and, thus, the entry mass) constraint may be mitigated by using either mechanically pre-deployed or inflatable heat shield designs. These systems have been entertained for many years; however, significant development work remains to prove the concepts to the degree that would be required for human systems. Recent work has illuminated some of the challenges in packing and inflating large inflatable heat shields. One advantage that these systems have is that they can be smaller in diameter than inflatable supersonic decelerators, such as the tension cone class of decelerators, and yet be large enough that subsonic terminal velocities can be achieved with the use of supersonic devices. The challenges of inflation under high-dynamic-pressure conditions, as would be required for supersonic inflatables, would thus be avoided. Clean coaxial separation of the central rigid heat shield and lander could then proceed along the lines of current Mars robotic missions.

A potentially significant challenge that is associated with the inflatable class or even some pre-deployed mechanical heat shield systems is that they require new, flexible TPSs as well as new mechanisms for hypersonic guidance. While these challenges do not appear insurmountable, a significant amount of work is required.

**Supersonic aerodynamic inflatable decelerators**: These decelerators beg some key questions. For example: How massive are these systems? Can they be safely deployed under high-dynamic-pressure, supersonic conditions? How quickly can they be inflated? What is the inflation mechanism? What are the aero-thermal environment concerns? How are they attached to the aeroshell? Can they be formed for slender bodies? How are they packed? How do they remain rigidized? How flexible can they be? Does the vehicle need to be guided when in use? If so, how? How are these systems tested at full scale? Many of these questions will require detailed simulation and materials testing.

**Supersonic retro-propulsion**: Supersonic retro-propulsion may provide a more robust alternative to other decelerator options, but this options also begs for answers to some key questions. For example: How much is the deceleration benefit from retro-propulsion due to changes in the vehicle drag vs. impulsive deceleration? How are the plumes formed so that the drag is maximized? How are the plume and flow managed so that aft-body aerodynamic force instabilities are manageable? How is the propulsion initiated, and how does it penetrate or bypass the aeroshell? How does the hot flow affect aero-thermal conditions? How is the propulsion used to separate the aeroshell? How are these systems tested at full scale?

Surveys of the state of the art from work done in the 1960s as well as modeling and wind tunnel experiments could expand our limited understanding of supersonic retro-propulsion.

**Hypersonic aerodynamic inflatable decelerators (inflatable heat shields)**: These systems, an example of which is shown in figure 7-7, also have their share of questions that need to be addressed. How are these systems rigidized for the relatively long duration leading up to and including entry? What are the candidate TPS materials that are compatible with storage, inflation, and flexible flight dynamics? What about aerocapture, are these systems compatible? What is the range of required heat rates and loads that these systems must endure? How is guidance achieved? Are there complex, flexible-body-control interactions? How are these systems tested at full scale?

**Figure 7-7.** Example of a hypersonic inflatable aerodynamic decelerator.

**Rigid hypersonic aerodynamic deployed decelerators**: While some work has been undertaken, an enumeration of the mechanical design possibilities needs to occur. What are the possible low-mass design options? Umbrella-like articulated heat shields? Segmented robotic on-orbit assembly? What are the mass fractions for these systems? How are the verified? Can they be used for aerocapture as well as entry? How are they stowed for launch?

**Thermal protection systems development**: The Mars AEDL TPS technologies will also require significant investment and be largely dependent on the AEDL architecture selected. These TPS technology development efforts are integral to the technology and system-level development and down-select process and need to be conducted as parallel development paths. The candidate Mars DRM AEDL architecture will define TPS challenges and new questions that need to be addressed in the short term. Some of these are: What TPS materials would need to be developed for inflatable heat shields? Does an inflatable heat shield need to be manually inspected before entry? Is this a show-stopper for inflatable heat shields? What is needed to ensure TPS reusability between aerocapture and entry (dual-use TPSs)? Does there need to be a repair plan prior to entry? How is thermal soak-back after aerocapture managed? If the TPS is exposed on the outside of the launch shroud, will it be adversely affected by the launch environment?

While progress in the development of ISS and lunar return TPSs for the Orion/CEV are well in hand for Block I and Block II, the Block III Orion/CEV TPS development requirements for the higher Mars-to-Earth return velocities place greater heating demands on TPS performance. These issues are discussed in greater detail in subsequent section.

#### *7.2.5.2 Forward work and requirements for Orion Block III TPS*

The CEV Aerosciences Project (CAP) team has been working for several years to improve the understanding of fundamental shock-layer radiative heating processes and have reduced the uncertainty (factor or two) that previously existed for lunar return conditions. A major uncertainty in shock-layer radiation is the level of heating in the very short wavelength region of the spectrum vacuum ultraviolet (VUV), which is difficult to study experimentally and to model in terms of radiative transport reaching the surface of the TPS. It is anticipated that the work of CAP team will continue to improve these issues for lunar return conditions. As discussed in section 5.5, when the Earth entry speed increases from lunar return speeds to the higher values for Mars return (from 11km/s to 14 km/s), radiative heating increases by a factor of four, and becomes the driver for heat load and TPS mass. Clearly, for Orion Block 3, efforts to understand shock-layer radiation are critical, and current efforts by the CAP team will be a valuable starting point for this future research on shock-layer radiation for Mars return conditions. Although the DRA 5.0 mission design limits Earth entry speeds to 12 km/s, it is suggested that TPS concepts for higher entry speeds be pursued.

It is believed that there is a large gap in the aerothermodynamics technology for Orion Block 3 for entry speeds much greater than lunar return speeds and thus a. It is felt that a 5- to 7-year development program, including efforts on shock-layer radiation, is required to address the needs of Orion Block 3. Elements of the needed program include the following:

**a) Facilities Needed for Validating Radiation Models**: Facilities that are capable of duplicating hot, contaminant-free air at conditions such as those that will exist in the bow shock layers that will form about Orion Block 3 at entry speeds up to 14 km/s are needed. One way to achieve this capability is by upgrading the NASA Ames Research Center (ARC) Electric Arc Shock Tube (EAST), that is currently providing such data for the CAP team at lunar return speeds.

**b) Code Development Needed:** Currently, Orion Block 2 TPS mass predictions are made with thermal response models such as FIAT (Chen, et al, 1999[18]) or STAB (Williams, et al, 1995[19]), both of which are tuned to a specific ablator based on arc jet data and measured material properties such as thermal conductivity of the virgin material. The thermal response codes use results from real-gas CFD codes such as DPLR (Wright, et al, 1996[20]) or LAURA (Cheatwood, et al, 1996[21]) and NEQAIR (NASA RP-1389[22]), radiation heating environments as their boundary condition.

For lunar return conditions, the shock-layer gases are weakly ionized. As speed increases, ionization will increase significantly. While the FIAT, STAB, NEQAIR, DPLR and LAURA codes are state-of-the art and took years of development to produce, they are inadequate to treat Orion Block 3. New codes should be developed that fully couple fluid flow, shock-layer radiative transport, and in-depth ablator response. In this fashion, the over-prediction of combined radiative and convective heating in the current models can be accounted for, as can the blowing of pyrolysis gases into the boundary layer. Coupling of radiative transfer with the fluid flow can account for probable reductions in the heating (and TPS mass) by the coupled effects of radiative cooling in the shock layer and its absorption by the boundary layer species. As done in the past, these advances will require ground-testing and theoretical analysis to ensure that the “physics” that is embodied in the new codes is “right” and that the required gas and materials properties are available. In addition to extensive ground testing, flight experiments to validate the new codes will be required as discussed below.

**c) Orion Block III heat shield shaping**: A companion approach that could be used to address TPS mass increases that are driven by radiative heating at the higher Earth return speeds is that of shaping the forebody heat shield (Arnold, et al, 1992[23]). Shaping of the forebody of Orion is practical and relatively easy, since current designs employ single-use heat shields, thus making upgrades to the Block 3 heat shield and using Block 2 after bodies. Given the computational modeling advances that have been made in the areas of aerodynamic predictions, it is not too difficult to accomplish a shape that provides nearly an equivalent aerodynamic coefficients and response for GN&C and yet produce a more desirable heating environment. Figure 7-8 shows an asymmetric shape whose purpose is to modify the shock-layer shape to be more inclined to the normal flow, thereby weakening the shock-jump for each streamline and reducing the local levels of radiation over a majority of the shock layer. Near the stagnation region, the flow direction will be normal and the full radiation levels will be reached; but overall, the TPS mass that is required will be reduced. In addition to addressing the radiation issue, a shape such as is shown in the right side of figure 7-8 also will increase the L/D ratio of Orion Block 3 with additional aerodynamic benefits.

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18 FIAT: Y.-K. Chen and F.S. Milos, "Ablation and Thermal Response Program for Spacecraft Heatshield Analysis," Journal of Spacecraft and Rockets, Vol. 36, No. 3, 1999, pp. 475-483.

19 Williams, S. D., Curry, D. M., Bouslog, S. A., and Rochelle, W. C.,“Thermal Protection System Design Studies for Lunar Crew Module”, Journal of Spacecraft and Rockets, Vol 32, No3., 1995, May-June, 1995.

20 Wright, M.J., Candler, G.V., and Prampolini, M., “Data-Parallel Lower-Upper Relaxation Method for the Navier-Stokes Equations,” AIAA Journal, Vol. 34, No. 7, 1996, pp. 1371-1377.

21 Cheatwood, F. M. and Gnoffo, P. A., “User’s Manual for the Langley Aerothermodynamic Upwind Algorithm (LAURA),” NASA TM-4674, Apr. 1996

22 Nonequilibrium and Equilibrium Radiative Transport and Spectra Program: User’s Manual,” NASA RP-1389, NASA, December 1996.

23 Arnold, J. O., Tauber, M. E. and Goldstein, H. E. Aerobraking Technology for Manned Sapce Transportation Systems, IAF-92-0764, 43 rd Congress of the International Astronautical Federation August 29-September5, 1992 Washington, DC .

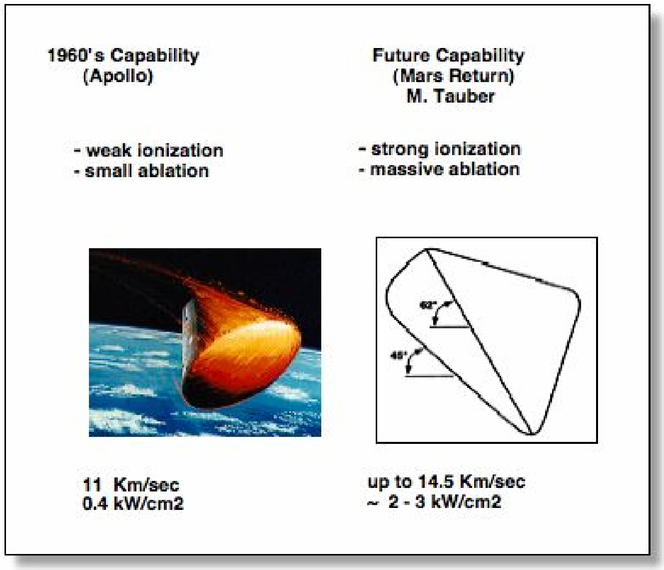



**Figure 7-8.** Apollo/Orion Block II heat shield shaping.

**d) Additional aerothermodynamics needs**: Other outstanding issues in aerothermodynamics besides that of radiative heating must be considered in Orion 3 design because there are processes at play for which current models are inadequate. These have arisen in analysis (Wright, et al, 2006[24]) of the performance of the baseline SLA heat shield material for the MSL that led to the adoption of a tiled PICA heat shield design, post CDR. These issues are listed below as they have direct correlation to the aerothermodynamics of Orion Block 3 and represent a technology gap for the its development.

- Catalytic heating on ablators.
- Turbulent roughness heating augmentation.
- Blowing reduction to heating levels.
- Ablation/computational fluid dynamics coupling.

In summary, large technology gaps exist in aerothermodynamics relating to the technology needsas entry speeds beyond lunar speeds for Orion Block 3. It is estimated that if entry speeds were increased to 14 km/s, the current technology that would be needed for Orion Block 3 is at TRL 2–3 and that it will take 5 to 7 years to bring aerothermodynamics technology to TRL 7, that is, to be ready to embark on hardware Orion Block 3 development. Elements of such a technology program in aerothermodynamics must include ground experimentation and facilities.

This work should include facilities research for upgrades, e.g., those that were mentioned above for the EAST facility, and possibly new aerothermodynamics ones, should start early to ensure the test capabilities are in place to provide databases and guide in development of new predicative codes. Codes accounting for the shortcomings of the state of the art today should be a major focus of new technology development. Ongoing work by the CAP team provides a good technical foundation for forward work in terms of technology and, more importantly, training of the next-generation aerothermodynamicists that is critical to agency plans for human Mars exploration.

[24] Wright, M.J., Edquist, K.T., Hollis, B.R., Olejniczak, J., and Venkatapathy, E., “Status of Aerothermal Modeling for Current and Future Mars Exploration Missions,” Paper No. 2006-1428, IEEE Aerospace Conference, Big Sky, MT, Mar. 2006.

***Thermal protection system technology for Orion Block III:*** This section begins with important lessons learned regarding programmatic aspects of TPS research and development, and concludes by recommending a future course of action for this program that has applicability to other future NASA missions.

As pointed out by NASA Administrator Griffin (Griffin, 2007[25]), the agency developed a broad base of outstanding technical capabilities in the 1960s, but later national and agency priorities led to their loss. Specific to TPS, the focus on reusable systems by NASA resulted in atrophy of personnel and facilities. The Apollo ablator AVCOAT went out of production. The SLA that was developed in the 1970s for the modest Viking entry had to be qualified for Mars Pathfinder for entry into the Martian atmosphere at 7.5 km/sec and was subsequently used for the MER missions and on Phoenix. SLA was baselined for MSL and very recently thermal testing proved (Beck, et al, 2009[26]) its limitation. In the mid 1990s, PICA was invented; subsequently, this material enabled the successful Stardust mission whose Comet Wild II sample return in January 2006 stands as the fastest-yet Earth entry at 12.9 km/s. Other ablator research, at a very low level, was ongoing in the 1990s mainly at ARC and also at the small company Ablative Research Associates (ARA) on families of materials that would be applicable for Earth and Mars entries. In 2001, ARC began to rebuild agency ablative capabilities around a single veteran of the Apollo era under the aegis of the NASA In-Space Propulsion Program that also supported ablator research by ARA. With the advent of the ESAS study, NASA came to the realization that the nation did not maintain the capability and had to re-develop ablators for Orion Block 2; this led to the formation of the agency-wide CEV TPS ADP, which did much to rebuild the national capability. As will be discussed below, current facilities for Orion Block 2 TPS testing are inadequate.

Heritage arguments often fail due to lack of clear understanding of the material limit capability and years of effort had to be expended to “bring back” industrial capability. This lesson learned – i.e., that technology that met the TPS of the needs of the last project may not work for the next one – strongly points to the need for a sustained program within the agency for TPS and related entry technologies, e.g., aerothermodynamics and high-enthalpy facilities, that stand ready to meet the increasingly more severe requirements of the agency. An excellent example discussed above is the future need for a mid-density ablator capable of performing in the 12- 14 km/s speed range for Orion Block 3.

PICA is not a viable option for Block 3. One might wonder why PICA TPS worked for Stardust at return speed of 12.9 km/s, but is not applicable to Orion Block 3 at similar speeds. Owing to its smaller size, radiative heating on Stardust was small compared to that expected for Orion Block 2, and this accounts for its peak heating being about 1,000 W/cm2 and close to that for the lunar return conditions of the CEV. Importantly, yet another difference in the Stardust vehicle and application of PICA for Orion follows from their difference in size. For Stardust, the PICA heat shield was a uni-piece construction. Manufacturing limitations require a tiled approach for Orion Block 2. Testing by the ADP indicates that the tile gaps can successfully work for the lunar return conditions, but it cannot be concluded that that a PICA tiled solution could work for Orion Block 3, since data to for gap performance at these conditions do not exist, and test facilities to study this also do not exist. Moreover, Orion Block 2 testing does not simulate radiative heating. Sizing results on Orion Block 3 for fully dense carbon phenolic show that this solution is too heavy to be practical.

Since it is apparent that no off-the-shelf ablator for Orion Block 3 exists, even at 12 km/sec, or will be forthcoming from the NASA’s efforts for Orion Block 2, a large technology gap in TPS exists.

It then seems very appropriate for a TPS research program to undertake development of a Block 3 ablative TPS and also to work ablators for delivery of heavy-mass payloads to the surface of Mars, as discussed elsewhere in this document. It is suggested that work be done to develop new families of mid-density carbonaceous ablators and possibly new aft shell TPS systems at approximately the levels of the current ADP. This research/development should focus on extensive ground test and code development that “marries” real-gas aerothermal, radiative transport, in-depth ablator response into a single

25 Michal D. Griffin, NASA Administrator “Human Space Exploration”: The Next 50 Years, National Aeronautics and Space Administration March 2007.

26 Beck, R.A.S., Driver, D.M., Wright, M.J., Laub, B., Hwang, H.H., Slimko, E.M., Edquist, K.T., Sepka, S.A., Willcockson, W.H., and Thames, T.D., "Development of the Mars Science Laboratory Heatshield Thermal Protection System", AIAA-2009-4229, 41st AIAA Thermophysics Conference, San Antonio, TX, June 22-25, 2009.

analysis tool, as mentioned above in the section on aerothermodynamics. This high-fidelity TPS tool should not only also address the issues that were discussed in the aerothermodynamics section above, but a high fidelity

TPS code should be part of the over-arching engineering analysis tools that are highly automated versions of the approach that was also described herein for Orion Block 2. Work on Orion Block 3 should include a demonstration of heat shield manufacturing capabilities and close integration with carrier structure developments.

***Facilities needed for thermal protection system research and development for Orion Block 3*:** The Orion project understands that current TPS testing capability is insufficient, even for lunar return conditions. Work is underway to upgrade existing arcjet facilities with radiative heating to better simulate lunar return for Orion Block 2. Orion Block 3 testing will require new capabilities that are far more capable and include both convective and radiative Components. It is very important that new TPS test capabilities be a part of the Agency’s portfolio. If the agency is to execute missions beyond lunar exploration, dedicated flight-testing both at sub-and full scale will be mandatory for human missions as was done for the Apollo program.

#### *7.2.5.3 Mars human-scale aerocapture and entry, descent, and landing technology roadmap*

During the course of this study, the team developed a proposed AEDL technology roadmap (figures 7-9 and 7-10). The roadmap was derived directly from the human planetary landing systems capabilities roadmap schedule that was developed in previous studies in early 2005. It has been updated to reflect the latest mission and development schedules that are associated with both the robotic Mars Exploration Program (SMD) and the human Constellation Systems (ESMD) timelines. It provides a rough estimate and an outline of the tasks and timelines that would be needed to develop a credible Mars AEDL system as well as subsystem designs for the first human Mars mission, which is currently planned for Mars launch opportunities in the early 2030 decade. While the NASA goal of a first launch in the early 2030s provides a reasonable target, this roadmap was not designed around any “required” launch date. Instead, the team generated a timeline of critical path activities and estimated how long each activity would reasonably last, beginning in 2007. Remarkably, there appears just enough time to complete the advance development work leading to the initiation of the flight system design qualification efforts that would be required in the early 2020s to achieve an operational human-rated system for launch in the early 2030 decade. In other words, it appears that the selection, design, and full-scale development of the final Mars EDL system architecture is on the critical path between now and an early 2030 decade human Mars mission initial operational capability.

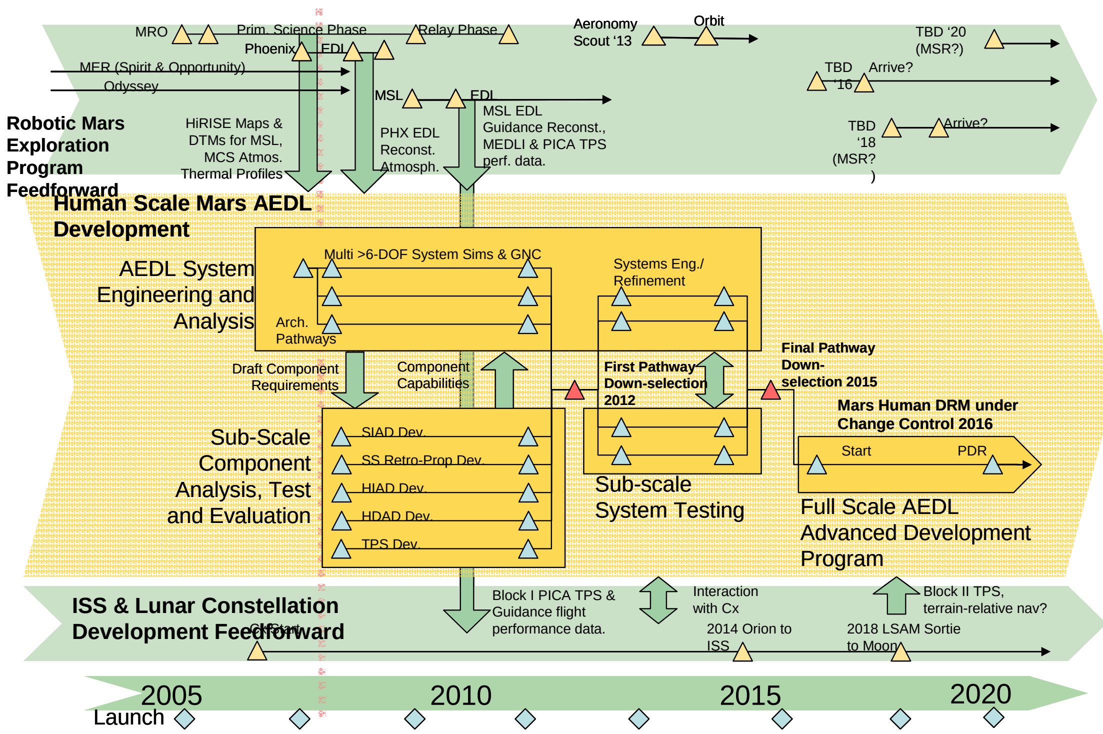



**Figure 7-9.** Human Mars architecture aerocapture through landing technology roadmap (2005-2020).

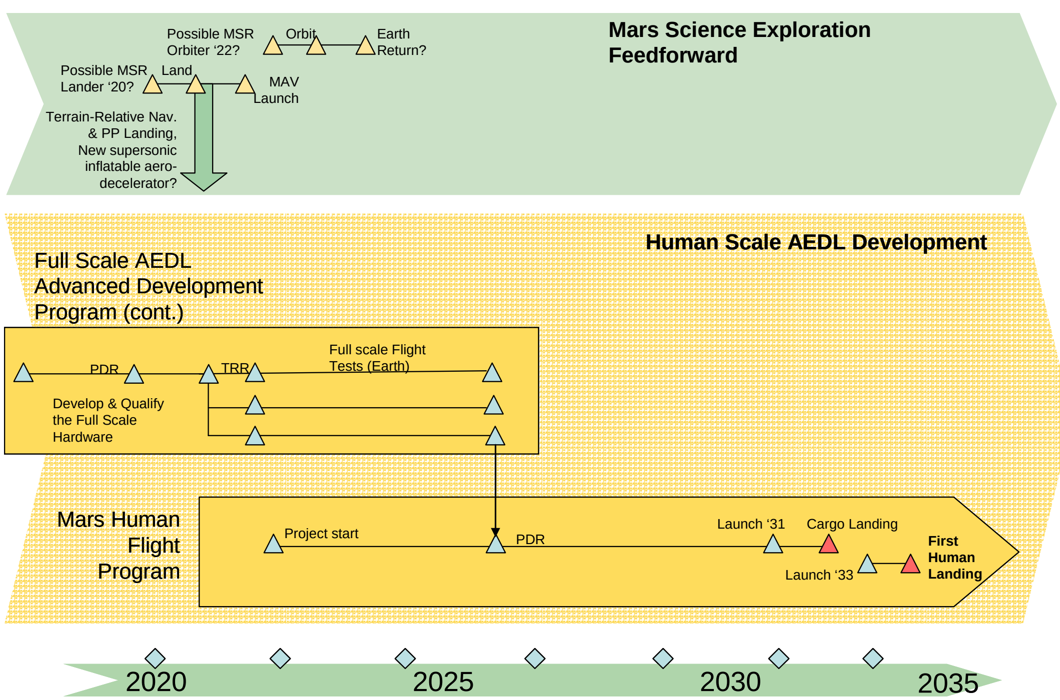



**Figure 7-10.** Human Mars architecture aerocapture through landing technology roadmap (2020+).

The proposed roadmap is really a long-duration development schedule with some key milestones. It is organized into a series of high-level components, each with its own objectives. These are: AEDL Systems Engineering and Analysis; Subscale

Component Analysis, Test and Evaluation, Subscale System Testing; Full-scale AEDL Advanced Development Program; and the Mars Human Flight Program.

The roadmap components are described in detail in the sections below. Also noted on the roadmap figure(s) are the current planned mission and development schedules for both the robotic MEP (upper) and the human ISS and lunar Constellation systems (lower). Where possible in the outyears, these development paths should be aligned and merged to take advantage of synergistic requirements and technology development opportunities to enable the eventual human-scale AEDL systems.

**Aerodynamic and entry, descent, and landing systems engineering and analysis**: The primary objective of this roadmap component is to first determine which AEDL system designs have the highest probabilities of being developed into a safe landing system, prior to down-selecting to one or more viable alternatives. System engineers, using detailed multi-degrees-of-freedom (DOFs) simulations of the proposed EDL system would work with AEDL component specialists to define the draft component requirements for each set of components in each of the proposed AEDL mission architectures. These specialists would propose and execute tests that would support or refute the draft requirements. The AEDL systems analysis and engineering team would then refine, modify, or abandon certain pathways based on this information. To allow for sufficient time to perform the needed component analysis and tests to gain a greater understanding of the component capabilities and limitations, the roadmap team estimated that it requires approximately 3 to 4 years before sufficient information would be available to make a down-selection decision to an appropriate, robust AEDL architecture (complete by ~2012).

Once one (or more) candidate architecture(s) has been thoroughly analyzed and modeled, including use of high-fidelity simulations, subscale system interaction testing would be planned. Depending on the DTOs, these are likely to require system test design and model updates as these tests proceed and are completed. Finally, by the year 2015 sufficient system and component test and design data would have been accumulated so that the reference Mars AEDL architecture (as well as the Mars design reference mission) would be settled and placed under full configuration change control.

At this point, the systems analysis and engineering team would be folded into an advanced development program whose objective would be to develop, verify, and validate the full-scale AEDL system components.

**Subscale component analysis, test, and evaluation**: The objective of this component is to acquire sufficient EDL component capability knowledge so that system architecture down-selection can proceed on or about the year 2012. The team estimated that all of these key systems could, with proper funding, be explored over a relatively few number of years once the draft component requirements were defined by the AEDL systems team.

**Subscale system testing**: It may well be that before any final pathway can be selected, one or more subscale terrestrial flight tests will be required. These tests may involve flights in both hypersonic and/or supersonic conditions under Mars-like dynamic pressure and heating conditions. Like the NASA balloon-launched decelerator tests (BLDTs) that supported the Mars Viking supersonic disk-gap-band parachute technology development of the early 1970s, these tests could be used to understand the larger-scale dynamics and systems interactions that are associated with full-scale systems. However, future work would be required to properly define the explicit test objectives of these test programs.

Whether needed or not, the objective is to complete an ensemble of subsystem and element technology flight tests so that the final AEDL architecture can be selected with high confidence by no later than the year 2015.

**Full-scale aerodynamic and entry, descent, and landing advanced development program**: This program would be a preamble to the actual Mars Human Flight Program (i.e., Constellation Mars). It would be a highly centralized and focused effort to design, analyze, verify, and validate the actual flight designs. The objective would be to execute the work that would be necessary to arrive at the program-level PDR of the Mars Human Flight Program with a validated full-scale AEDL design. Central to this would be a series of full-scale flight tests (likely at Earth) that would validate the designs of all key AEDL components. Noting that it may not be possible to design a single test that covers the entire EDL flight domain, these tests may be broken up into focused tests (also at or near full scale) that address some specific flight domain.

As the AEDL components need to be designed in flight detail, this advanced development program must follow a project-like life cycle to ensure that the components are delivered to the full-scale flight tests on time. The roadmap team felt that this program would require “traditional” durations from project inception through component and test plan PDR on or about the year 2020. The flight tests would be planned for the early 2020s and would cover durations similar to other full-scale terrestrial flight test durations (e.g., Apollo 1). When complete, each of the AEDL systems would have been designed, tested, and validated at full scale by the time of the Mars Human Flight Program PDR in the year 2024.

The road-mapping team debated at length the need for a subscale flight test at Mars of a human-Mars AEDL design. A particular concern was expressed by many that the first landing at full scale at Mars should not be crewed. There are too many untested attributes of a Mars EDL that are too difficult to reproduce at Earth to trust that the first full-scale flight can be safely flown with astronauts on board. The first flight should at least be a pre-deployed surface asset with an EDL system that is identical to that used by the follow-on crewed landing.

However, the team was split on the necessity for a subscale test flight at Mars. While it was agreed that much could be learned, the time and cost for a single subscale test flight would detract from a higher quantity and quality of focused full-scale testing at altitude on Earth. In particular, the team noted that such a flight would add about 6 years to the development timeline and would not be able to meet the year 2030 first launch target. In the roadmap that is presented here, the subscale at-Mars test program was eliminated.

**Mars Human Flight Program**: This is the actual program that develops the entire human Mars mission (at least the first mission). In keeping with other large NASA projects, its PDR would be about 4 years prior to launch of the first landed (uncrewed) asset. This system would be constrained to use the EDL components that would be validated by the ADP; however, it would focus on mission design and payload design.

**Feed-forward opportunities from a robotic Mars science program**

*Orbital reconnaissance using Mars reconnaissance orbiter mapping for human site selection*: MRO imagery will most likely be the primary source of site selection and scientific mission design for the Mars Human Flight Program. Since MRO science is current scheduled to terminate in the year 2009, candidate MSL landing sites that will be assessed by MRO prior to the year 2009 will provide the bulk of landing site candidates for the Mars Human Flight Program. NASA should consider a human Mars landing site assessment activity prior to closure of MRO science operations.

*Mars atmosphere measurements*: Successful AEDL at Mars requires knowledge of the nominal atmospheric density as well as mesoscale and local weather conditions. EDL adds the additional complication of the need for wind knowledge. Whether aerobraking at altitudes of the order 100 km, aerocapturing at 30 to 50 km, or performing an EDL sequence where maximum aerodynamic loads occur near 40 km followed by terminal descent to the surface, latitudinal, seasonal, diurnal atmosphere variations, and winds (EDL) must all be considered to determine nominal flight environments. Currently, uncertainties in the nominal atmospheric density across these altitude bands range from 25% to 200%. Furthermore, unmodeled, short-temporal and spatial-scale variations can significantly affect the success of the mission.

The three Mars aerobraking orbiters (MGS, Odyssey, and MRO) have made in total approximately 2,000 atmospheric passes and have provided significant data to refine the high-altitude, upper atmospheric models that would be used for aerobraking. The success of these missions and the atmospheric knowledge gained from them have made aerobraking at Mars a mature operational mode. The knowledge that was gained, which is now included in future aerobraking mission designs, includes the observation that at 100 km, density profiles have frequently doubled in 20 km of purely down-track (constant altitude) motion. Although smaller in relative amplitude, similar waves have been identified in the middle atmosphere during the five successful NASA EDL missions, Vikings I and II, Mars Pathfinder (MPF), and MER Spirit and Opportunity. Several EDL missions encountered unexpectedly high winds in the lower 10 km and, perhaps, even clear air turbulence. Precision landing will be particularly difficult with high, unexpected winds. To design successful missions, the current approach is to absorb these large uncertainties and variations using safety margins between 25% and 100%, which typically translates to large increases in system mass. In addition, this uncertainty may preclude the consideration of many scientifically significant landing sites.

Large uncertainties in atmosphere models directly increase mission risk, reduce payload margins, and limit potential landing sites. Adequate knowledge of atmosphere properties is fundamental to mission success, as much as knowledge of the strength of Al alloys or the space radiation environment. The Mars atmosphere has two characteristics that significantly increase aerocapture and EDL risk. The first characteristic is that the martian atmosphere responds rapidly and dramatically to regional and global dust storms that can cause large density and wind variations throughout the entire atmosphere to 100 km and higher within a few days. Improving the physical models in general circulation models (GCMs) is required to address this issue. The second characteristic is that topographically forced winds can be greater than 10 m/s and can produce landing errors of over 1 km during the EDL parachute phase. There are essentially no validation data for the mesoscale models (MMs) that are used to generate these predictions. Atmospheric measurements to date have suffered either from a lack of vertical resolution, spatial coverage (latitude and longitude), or temporal coverage (seasonal and diurnal). A global measurement campaign will be needed to improve the physics of both the GCMs and the MMs.

NASA has obtained limited atmospheric data to confirm the atmospheric models that would be used in the design and evaluation of EDL and aerocapture at Mars. The five successful landers each provided accelerometer data during their respective EDL phases, which has been used to infer localized density and wind characteristics. The Viking landers and MPF have provided some localized ground level data on pressure, temperature, and winds. The thermal enclosure system (TES), thermal emission imaging system (THEMIS), and maintenance control system (MCS) instruments have provided low-resolution temperature and dust characteristics, but the data that were gathered are insufficient to reduce the uncertainty in the density models, and provide no data to improve the wind models. This uncertainty in density and wind profiles has had significant impact on the mission design, spacecraft design and capability, mission risk, and landing site selection for the current Phoenix and MSL development efforts. As the landed missions evolve from MSL to human missions, and aerocapture is added to the mission toolbox, it is imperative that the modeling of the atmospheric density and wind profiles be improved upon. There has been some forward planning within the MEP for a future Mars science orbiter that would provide more detailed atmosphere profile measurements in the lower atmosphere; however, these plans are preliminary. The Mars Scout aeronomy missions will provide valuable Mars atmosphere chemistry and history data; however, it is not clear that this could aid AEDL development.

**Future Mars robotic lander entry, descent, and landing design and performance data**

*Phoenix entry, descent, and landing performance*: Phoenix EDL reconstruction data in 2007 will provide valuable high-resolution atmosphere density reconstruction as well as overall performance model data that may be extrapolated to other AEDL systems.

*Mars Science Laboratory entry, descent, and landing performance*: MSL EDL reconstruction data in the year 2010 will again Provide valuable high-resolution atmosphere density reconstruction as well as the first guidance performance data for precision and, ultimately, pinpoint landing.

*Future landers*: After MSL in 2009, there are currently no U.S. Mars landers planned prior to the first MSR lander in the late 2010s. Provided that the MSR landed payload mass can be properly managed (e.g., a small MAV), this lander is likely to be a direct derivation of the MSL AEDL design. However, if the landed payload turns out to exceed the landed mass delivery capability of the MSL of about 900 kg, augmentations to the MSL design may be required. This would also require that a larger launch vehicle be used than is currently envisioned as well as a considerable effort to qualify this system. However, design options could provide valuable feed-forward capability that could be used by the human mission AEDL design.

The only other Mars lander that is being considered in the 2010 decade is the European ExoMars lander. This lander is based on MER and MSL-like technology and does not appear to offer any direct feed-forward technology or data opportunities to human-scale AEDL systems.

### 7.2.6 Space transportation critical challenges and technology needs summary

Space transportation challenges and testing venues are detailed in tables 7-3 and 7-4, respectively.

**Table 7-3.** Space Transportation Challenges

| **Current Knowledge or Capability Gaps** |
|---|
| • Ability to use the Ares V payload shroud as the Mars EDL system<br>• Synergism of lunar and Mars heavy-lift needs<br>• Oxygen-based propulsion for Mars ascent<br>• NTP testing and performance<br>• Ability to land large (40 times current capability) payloads on the surface of Mars |
| **Technology Needs** |
| Earth-to-orbit Transportation<br>• Large composite structure manufacturing<br>• HTPB propellants<br>• Dual-use shroud<br>• High reliability and high availability<br>• No more than 30 days between launches<br>Advanced Chemical Propulsion<br>• Oxygen/$CH_4$ propulsion for Mars ascent<br>Nuclear Thermal Propulsion<br>• Development of NERVA-based “composite” and “cermet” fuel options<br>• Bi-modal NTR for thrust and power generation (enhanced technology option)<br>• NTR $LO_2$ “after-burner” nozzle for variable-thrust/$I_{sp}$ operation (enhanced technology option)<br>Electric Propulsion<br>• Multi-megawatt electric propulsion (~500 kWe to multi-MWe-class electric thrusters)<br>• Multi-megawatt power generation systems (solar or nuclear)<br>Entry, Descent, and Landing<br>• Hypersonic decelerators<br>• Supersonic decelerators (inflatables, retro-propulsion, etc.)<br>• Advanced lightweight TPSs<br>• Terminal landing hazard avoidance |

**Table 7-4.** Space Transportation Testing Venues

| Venue | ETO Transportation | Chemical Propulsion | Nuclear Thermal Propulsion | Electric Propulsion | Entry, Descent, and Landing |
|---|---|---|---|---|---|
| **A. Earth Surface** | • Large-scale composite manufacturing<br>• System performance during lunar mission campaign | • Engine testing | • NTR engine testing | • Thruster development testing<br>• High-power generation testing | • TPS material testing and certification |
| **B. Earth Atmosphere** | • N/A | • N/A | • N/A | • N/A | • N/A |
| **C. Earth Orbit** | • Automated rendezvous and docking in LEO demonstrated | • Flight test demonstration<br>• Active cooling (cryo-coolers) | • N/A | • Large-scale system deployment tests<br>• System operation and control tests | • N/A |
| **D. Lunar Transit and Orbit** | • Ares V EDS for major maneuvers | • $LO_2/CH_4$ engine demonstration | • Potential demonstration on lunar missions | • Potential demonstrations on lunar missions | • N/A |
| **E. Lunar Surface** | • N/A | • $LO_2/CH_4$ engine demonstration<br>• Long-duration cryogenic engine performance | • N/A | • N/A | • N/A |
| **F. Deep Space Transit** | • N/A | • Deep-space engine tests and demonstration<br>• Potential use on robotic missions to outer planets | • Deep-space engine tests and demonstration<br>• Potential use on robotic missions to outer planets | • Deep-space engine tests and demonstration<br>• Potential use on robotic missions to outer planets (nuclear) | • N/A |
| **G. Mars Orbit** | • N/A | • N/A | • N/A | • N/A | • N/A |
| **H. Mars Atmosphere** | • N/A | • N/A | • N/A | • N/A | • Scalable precursor flight tests at Mars<br>• Full-scale cargo mission aerocapture<br>• EDL Mars flight certification<br>• Ares-V shroud as aeroshell |
| **I. Mars Surface** | • N/A | • Demonstration of $LO_2/CH_4$ engine on robotic/cargo missions | • N/A | • N/A | • Precision landing and hazard avoidance |

## *7.3 Surface Systems*

### 7.3.1 Advanced habitation systems

Structural materials advancements to provide large, livable volumes, both in transit to and from Mars as well as during surface exploration, while minimizing mass are desired for human exploration missions. Limited volumes and the complexity of packaging the Mars lander and surface systems within the aerodynamic shell of the entry system will most likely require advanced inflatable structures. Key technology thrusts include habitat concepts and emplacement methods (including remote and autonomous operations), advanced lightweight structures (inflatable vs. traditional "hard shell"), and the development of integrated radiation protection for crew health and safety.

In addition, developing technologies that can significantly reduce the consumables that would be required to support the crew during a long-duration mission are also critical for human exploration of Mars. Technologies include air and $H_2O$ loop closure, environmental monitoring, solid waste processing, thermal control, and food processing. Advanced sensor technologies to monitor and intelligent systems to control the environmental "health" of the advanced life support system, including air and $H_2O$, are also needed.

- Devise inflatable concepts that provide packaging and mass savings as compared to conventional designs

- Provide radiation protection without significantly increasing habitation system mass
- Be capable of autonomous operations of the integrated systems
- Be capable of deployment, assembly, and checkout autonomously
- Require minimal crew time for operations and maintenance
- Provide nearly closed (~99%) provision of air and $H_2O$
- Evolve to providing increased (50%) food production for planetary crews
- Be capable of using local planetary resources ($O_2$, $H_2O$, soil)
- Be capable of performing waste processing and recovery of useful resources

### 7.3.2 Extravehicular activity and surface mobility

The success of exploration missions depends on the ability of humans to work on and explore planetary surfaces. This success will depend on productive eEVA conducted at great distances from the surface landing site or outpost. During these missions, astronauts will be exposed to a range of gravity conditions and a diversity of environments. With the normally intense activity that would be expected on the exploration missions, issues of productivity, usability, durability, and maintainability of EVA systems become acute. Operational and medical considerations will include pre-breathing procedures, life-sustaining system capability, environmental health, radiation protection, and emergency-mode operations. Allowing humans to make the transition simply and effectively between activities that are conducted inside and outside vehicles will both enhance productivity and increase overall mission safety. EVA systems must be provided for the moon, Mars, and space operations in orbit and in transit.

Repeated, productive surface activity for 4 or more hours each day for up to 6 days a week is a requirement. Planetary surface systems, including suits, will have to be maintained by the crew, and must be resistant to contamination by surface materials such as dust. EVA systems must provide a safe, nontoxic environment, with food and $H_2O$ supplies that are nutritious, esthetically pleasing, and free of contamination. In addition, several operational considerations are important to the effective use of suits or other individual mobility devices. The first consideration is to minimize the time that would be needed to go from inside the spacecraft out to the planetary environment; of particular importance is the difference between habitat pressure and the pressure of the EVA system. The greater the difference, the more time must be spent adjusting to the generally lower outside pressure to avoid the bends. The second consideration is to maximize the time that would be available for productive activity outside the spacecraft; suit mass and the ability to supply food and remove waste are particularly important. It is important to minimize restrictions on human capability by providing adequate thermal control, greater suit mobility – in particular in the gloves, torso, and boots – and enhanced communications capability for explorers and home base interactions. A third overall requirement is the maintainability of the system, allowing reuse without extensive overhaul. Current capabilities in planetary surface EVA systems are derived from Apollo systems, which provide an excellent starting point for future missions. For long stays on planetary surfaces, however, these systems do not meet the stated mission requirements in most areas. Apollo technology equipment is not maintainable, and it is too massive to use on Mars, where the gravitational level is 0.38 that of the Earth. The space shuttle and ISS systems are even more massive and were designed for only limited use in microgravity environments. New systems will be required and will evolve from enhanced spacesuits that are similar to those now in use; in addition, development of teleoperated tools that will allow an astronaut within a closed mobile chamber to manipulate devices outside the chamber may be incorporated and could blur the line between suits and habitats.

Suit development is an area that will require focused research and technology efforts emphasizing lightweight and durable materials, glove design, dust contamination protective measures and techniques, lower torso mobility systems for walking, ancillary mobility systems for surface transportation, and long-term reusability and lightweight, compact, PLSS technologies.

Mars exploration will be preceded by extensive experience conducting exploration on the lunar surface. This experience will prove out many of the concepts that are associated with Mars exploration, including:

- Routine EVA operations
- Suit design to minimize crew wear and tear and fatigue levels over an 8-hour EVA
- Gloves that enable tactility and dexterity with reasonable crew fatigue over an 8-hour EVA for lunar surface operations
- Geosciences exploration techniques, including geophysical and geochemical prospecting, and remote-sensing tools to assist crew field geology operations

- Simultaneous, synergistic operations between EVA crewmembers and robotic assistants
- PLSS recharge during EVA
- Management of in-situ silicate and metallic oxide particulates

These concepts will be critical to conducting routine EVAs on the martian surface over a 500-day period, particularly in situations where purely robotic exploration has proven less effective than exploration by human crew members.

Although the lunar surface has the potential to gain the EVA experience that will be needed for crewed Mars exploration, a number of differences between the martian environment and the lunar environment will dictate substantive changes in the EVA architecture. These environmental differences include the increase in martian gravity to 0.38g relative to 0.17g on the moon, and the difference between the lunar atmospheric "pressure" of $1.3 \times 10^{-13}$ mbar and the 10-mbar atmospheric pressure on the martian surface. In addition, the issues that are associated with the possible existence of fossil or extant life will require measures to be take to meet planetary protection protocols that are not necessary for operations on the lunar surface.

Present and planned PLSS/thermal garment designs for orbital and lunar applications make extensive use of space vacuum to induce a phase change, thereby creating a heat sink, and to inhibit conductive cooling from appendages through the suit to the external environment using MLI. The martian atmosphere, at 5–10 mbar, has sufficient mass properties to inhibit vacuum-dependent thermal management schemes (L. Trevino, personal communication, 2007). This means that we will need to develop (1) new methods of dumping crew member metabolic heat to the environment, thus preventing heat-storage-related injuries (hyperthermia, heat exhaustion, heat stroke), and (2) new methods of preventing excessive heat leak through the pressure garment to the martian environment, thereby preventing cold exposure injuries (frostbite, hypothermia).

Unlike the lunar surface, exploration of Mars must be conducted with issues that are relative to planetary protection, particularly management of organic contaminants that would be released by EVA suits and PLSSs. The nature of pressure garments is that leakage of some quantity of internal atmosphere is unavoidable. For suits that have a significant usage history, such as the Mark III suit, the leak rate can be as much as 1,500 cc/min. Operational suits, such as the shuttle EMU, experience leak rates about an order of magnitude less, on the order of 100 cc/min (J. Kosmo, personal communication, 2004). Leakage includes the gas that is used to pressurize the garment as well as any airborne particulates that can escape past garment seals, which can include microbes and latent virus particles that are shed from crew members. A critical engineering and operational challenge will be to manage this leak rate and character, potentially through the use of improved seals, sterile over-garments, or covers around mechanical connection areas such as glove and neck rings, and through use of operational practices that minimize human crew member entry into areas that are suspected of having extent or fossil martian life.

For Mars surface exploration, scientific diversity is obtained by extending the range of human explorers via both unpressurized and pressurized rovers. Long-range pressurized rovers may be large, complex machinery upon which much depends. A thorough understanding of operational issues and failure modes will be essential. Both Earth analogs as well as lunar missions will play a vital role in determining the performance and operational scenarios, including maintenance and repair, of surface mobility systems.

Since surface mobility systems will use many of the same types of mechanical equipment, structural elements, and materials as other mechanical surface systems, long-term reliability of large rover systems in extreme planetary environments needs to be established. Common systems need to be qualified for multiple uses. Programs such as those that are employed by the U.S. Army Cold Regions Laboratory and data from MER and other future Mars rovers are highly relevant. We need to learn how to make mechanical elements perform reliably in environments for which we have little direct experience.

Once lunar operations commence, close monitoring of the performance and health of the mobility system should be performed, not just to predict system availability but to validate expected performance extended to Mars. In addition, the effectiveness of the mobility systems as true enablers of human activities should be assessed. Overall work efficiency indices (WEIs), which measure the overhead time that it takes to prepare for an EVA compared to the productive EVA time that is provided, should be determined; for example, how much time on remote surfaces is actually provided by pressurized rovers. The true WEI should be carefully documented and projected to the Mars case to allow refinement or alteration of the concept. In short, lunar surface mobility operations will play a major role in defining similar Mars activities at the mission system level, especially if the system configurations and construction are similar. Predicting the ability to move over and access a planetary surface using the lunar experience is key to the success of Mars exploration, and is embodied in the plan for the Vision for Space Exploration.

### 7.3.3 Subsurface access

Drilling and collection of samples from the subsurface of Mars will be an important science activity to be performed by human mission crews. Three possible mission scenarios have been described (see table 7-5) that differ primarily by the range of mobility that would be offered to the crew and the depth of drilling that is associated with each scenario.

**Table 7-5.** Example Subsurface Access Scenarios

| Scenario | Traverse Duration | Drill Depth |
|---|---|---|
| 1. Small Pressurized Rover | Several days | 10 m |
| 2. Large Pressurized Rover | Several weeks | 100 m |
| 3. Habitat Based | None assumed | 1,000 m |

Scenario 1, which can be accomplished with drill technology that has development heritage from the Mars robotics technology program, has a total drill system mass of less than 100 kg. Scenarios 2 and 3, which require more elaborate and massive systems, are at a much lower TRL from a flight system point of view. For this reason, precursor programs for the 10-m class system differ from those for the 100-m class system and are deeper, although there are common elements. Design/development steps include the following:

- Select drill emplacement approach (drill trailer vs. mounted on human-occupied rover)
- Study to select/determine the most promising cutting and cuttings transport approaches; options include surface-driven rotary cutting, down-hole rotary cutting, laser ablative cutting, and heat fracture cutting
- Develop drill string changeout, core capture, core handling, and storage systems
- Develop drill head changeout system (rotary cutting bits must be changed often)
- Design cuttings transport system (fluids or gas-jet cleaning)
- Drill fault-diagnostic and autonomy systems to operate the drill
- Develop hole casing system

#### *7.3.3.1 Drilling precursor developments*

It is important to select the most feasible and reliable drilling approach that would be likely to work on Mars. Based on vast terrestrial experience, this drilling approach is most likely to be rotary cutting.

Research on cuttings transport method is probably the most important element of design that must be addressed first theoretically and then in the terrestrial laboratory before selecting the best approach for Mars. Cuttings transport simply will not behave the same way as on Earth due to the low gravity and low atmospheric pressure and temperature on Mars. Efficient cuttings transport will be important, especially since rapid rate of penetration is desired. For the shallow-drilling case, dry rotary cutting may suffice but is likely to be slower than desired. Alternatively, a pressurized stream of Mars atmosphere could be used to aid in transporting cuttings that will be produced on a drill site using an air compressor. This method may also work in a deep hole to assist with transporting cuttings into a cuttings cup that is periodically shuttled to the surface and emptied when full. However, for deep drilling, the use of low-temperature drilling mud that is based on $H_2O$/brine combinations could plausibly work better, and this approach should be seriously evaluated.

Research on bit design for Mars is also needed. Cutters need to be chosen that are optimized for either a wide range of materials that might be encountered, or a selection of bits needs to be specialized to the most likely materials that would be encountered with bit changeout to occur according to the material type that is actually encountered. Provisions should be made for bit changeout due to bit wear every 10–20 m of drilling. This means that the bits must be considered an expendable resource, and bit changeout, either by autonomous means or crew-assisted, should be planned.

For shallow drilling, it is feasible to retrieve the drill string and core barrel to the surface for each core that is retained. For deep drilling, this would be time prohibitive, so a core barrel that can be shuttled to the surface without retrieving the drill string would be necessary.

Deep drilling will almost certainly require hole casing. Casing will be used to stabilize the hole, preventing collapse that results in getting the down-hole elements stuck and resulting in a loss of the drill head and/or string. The most feasible approach is to plan to case the upper 100 m of the hole with a casing that has a graduated diameter; i.e., a diameter decreasing with depth. This requires a drilling plan in which the larger diameter bits are used in the upper drilled sections, then casing is inserted, and smaller diameter bits are subsequently used to go deeper. Casing can be made of relatively thin-walled, lightweight tubing. Studies should identify the most mass- and volume-efficient casing material. Quite possibly an inflatable flexible tubing can be identified that becomes rigid once emplaced. Cements for sealing the casing into the borehole must also be identified that will work at martian temperatures.

A system must be developed for retrieving cores and transferring them into sterile liners for storage. These cores might be captured in liners during the drilling process, and the liners may be sterilized on the outside, thereby serving to keep the core material separated from contamination by the crew.

#### *7.3.3.2 Precursor mission requirements for drilling*

In addition to development, the drill systems need to be terrestrial field tested to verify their performance. These tests should be conducted in the most Mars-like environments that are available, such as low-temperature winter arctic environments. Since lunar missions may also require drilling with similar depth requirements, shared heritage with lunar drilling systems is possible and desirable, but the cuttings transport approach that works best on Mars and the moon may differ due to different environmental parameters of gravity, atmospheric pressure, and temperature and the fluids that are likely to be available in each case.

Precursor missions for Mars drilling should characterize the environment that is to be drilled as thoroughly as possible. Needed information includes the type of lithology, and how it varies with depth and the porosity or degree of consolidation expected. Subsurface geophysical sounding should be performed either by previous human missions or by robotic missions to determine that the desired target of the drilling is present. For example, radar or electromagnetic sounding that indicate the presence of a subsurface aquifer should be used as a guide as to where to drill. Ground-based data with high spatial resolution should be used since the investment in a deep well is large and aquifers may be substantially heterogeneous. Finally, a subscale robotic drilling mission should be performed to verify that the assumed cuttings transport approach will succeed.

### 7.3.4 Advanced space power

Highly reliable sources of power will be required for human Mars missions because life support is a critical function. Surface power systems were assessed for three architecture options:, nuclear and solar options for the main base and fuel cells, and the batteries and isotope systems option for long-range pressurized and local rovers. The following sections detail current planning for these technologies.

#### *7.3.4.1 Advanced power generation*

**Solar power**: A PV solar power system uses solar cells that are configured into an array and, typically, coupled to an energy storage device. Energy storage is required to provide power when the array does not see the sun or when the power output is attenuated below load requirements. Energy storage also provides peak power demand. The current solar cells that are available – e.g., advanced triple junction gallium arsenide/Germanium (GaAs/Ge), which are used on the MERs – achieve approximately 27%. Even with high-efficiency solar cells, array areas that would be needed to produce the required powers for a human mission become very large. Thus, a key technology area is the deployment and retraction mechanisms as well as methods to effectively clean large arrays.

Significant design challenges for solar power are listed below:

- Power loss due to dust accumulation (abatement techniques mandatory)
- Array deployment methods for large areas
- Diminished output power during dust storms
- Variability of magnitude and duration of storms
- High-efficiency, lightweight rollout array for dust storms

Dust accumulation must be abated. A power loss of 10% was factored into the array area to accommodate anticipated martian dust accumulation. MER array data suggest that typically it takes 40 to 50 days to degrade to that level (i.e., the level at which a power loss of 10% is experienced). A robust method must be identified that can operate robotically since the arrays will be operational during the pre-deploy phase. The full power system will be deployed to make the ascent $O_2$. Several methods of dust abatement have been discussed, including the use of compressed gas “blow off,” mechanical wiping, vibration to fluff of the dust, and electrostatic repulsion. MER has shown that wind, most likely at a favorable “angle of attack” to the array surface, does restore lost power. It is impractical to make an array that would be big enough to provide full power needs at the approaching 60% losses that were experienced by the MERs.

Array deployment is also an area of concern. The array size that was selected was 2.5 m high × 58 m long and there are 10 array wings. The array and companion RFCs are housed in an enclosure with the array box on the ends. Each PV/RFC must be offloaded from the lander and emplaced in a particular orientation with the solar array facing south. This would be accomplished tele-robotically from Earth.

The power team did not assess various methods for array deployment, but a workable scenario whereby a rover would be outfitted with a special attachment would assist in “pulling out” the array that might be supported by a telescoping header beam that was extended from both ends of the enclosure. Drop-down legs would swing down at intervals to support the entire array wing (item 3 below). Resources did not allow a detailed assessment, but the team did develop issues to be addressed in follow-up work. RFC technology is discussed below.

1. Suitable topographic site for full, unobscured array extension
2. Reliable self or assisted array mast and array panel deployment
3. Automatic anchoring for array support and wind loading capability
4. Validated methodology for dust removal

**Nuclear power**: A roadmap for fission surface power (FSP) technology development leading to possible flight applications is shown in figure 7-11. The technology development could include three elements: (1) conceptual design studies, (2) advanced component technology, and (3) non-nuclear system testing. The conceptual design studies would expand on recent analyses that were conducted by the Jupiter Icy Moons Orbiter (JIMO) Project, naval reactors (NRs), the NASA ESAS Team, Glenn Research Center (GRC), and LANL. The intent would be to increase the depth of study in areas of greatest uncertainty such as reactor integration and human-rated shielding. The advanced component technology element would address major technology risks through development and testing of reactor fuels, structural materials, primary loop components, shielding, power conversion, heat rejection, and PMAD. Non-nuclear system testing would be conducted via a full-scale, electrically heated technology demonstration unit (TDU). The TDU would provide a modular, technology testbed in which to investigate and resolve system integration issues. The three elements are highly coupled with component technology selections that were predominantly determined through the conceptual design studies and TDU hardware elements that were provided from the advanced component technology element. As new components are developed, they would be inserted and demonstrated within the TDU system context.

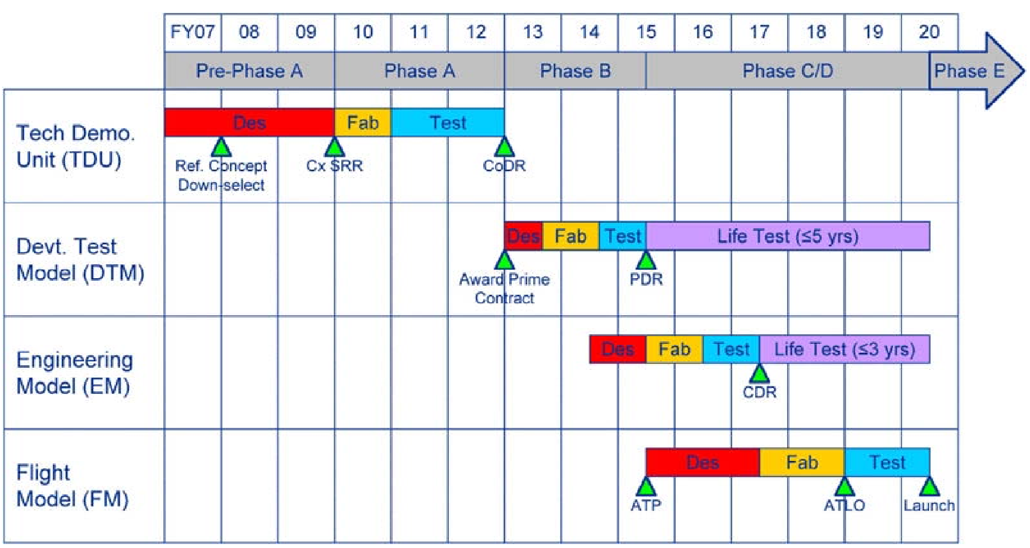



**Figure 7-11.** Technology roadmap and relationship to flight system development.

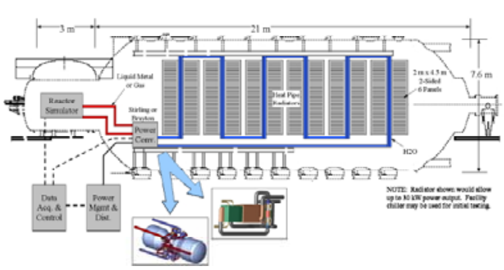
Significant progress can be made in FSP technology over the next few years with moderate funding. The proposed technology development would address both the reactor and the balance-of-plant with the primary objective of reducing development risk and cost. A nominal budget of $10M to $20M per year over the next 5 years would be sufficient to sustain proposed technology development activities. Conceptual design studies would be a key element providing direction on system requirements, mission integration, and technology selection. The trade studies would also help to identify and prioritize component technology investments. The component tasks would focus on hardware development and risk reduction.

Completion of the power conversion and heat rejection technology tasks that were started under the JIMO Project provides an opportunity to accelerate the maturation of several key technologies. Many of the JIMO tasks are relevant to the surface power application, and represent meaningful hardware-based tasks. Among the test hardware deliverables is a 100-kWe-class Brayton alternator test unit, an experimental 30-kWe twin turbine closed-loop Brayton power system, three full-scale multi-heat pipe radiator panels, and nine high-temperature $H_2O$ heat pipes with various wick designs.

The FSP advanced component technology element would build on these activities while expanding the breadth to include reactor- and shield-related development. Additional component technologies that could be pursued that would be specific to the FSP application include reactor fuels, structural materials, primary loop components, shield materials, high-power Stirling conversion, and high-voltage PMAD. On the nuclear side, initial irradiation tests could be performed on candidate fuel forms. In parallel, materials testing could evaluate radiation effects and fill gaps in thermal-mechanical property databases. Additional reactor-related items for development include primary pumps, heat exchangers, accumulators, control drive actuators, and instrumentation. Since shielding has a major influence on design and mass, several early experiments could be conducted to evaluate material and packaging options. On the plant side, component development activities could expand on JIMO efforts while focusing on lunar and Mars environment issues. Of particular interest would be radiators and transmission cabling that are suitable for planetary surfaces and amenable to the various power conversion options. The component technology element would also include the further development of multi-kilowatt, 900 K Stirling converters.

A crucial element of the near-term technology plan is the design and test of a full-scale, end-to-end, electrically heated TDU. A notional test layout for a 30-kWe TDU is presented in figure 7-12. Most of the current fission design concepts and trade studies are based on technology development that was conducted in the 1960s through 1980s. This test could provide a much-needed experimental validation of the overall power system that is based on modern design and fabrication methods to anchor flight reactor performance projections. The major test goals could include to: (1) demonstrate system performance, (2) verify manufacturing capabilities, (3) obtain comprehensive temperature, pressure, and flow data under steady-state and transient conditions, (4) expose component interactions and interdependencies, (5) develop safe and reliable control methods, and (6) validate analytical models. The TDU activity would help to stimulate industrial infrastructure for component design and fabrication, and would provide critical as-built mass and cost data. In addition, the TDU would provide NASA and DOE personnel with valuable hands-on operating experience that would support a successful transition to flight development.

1. Reduce Development Risk
2. Verify System-Level Performance in Simulated Environment
3. Characterize Component Performance in a System Context
4. Obtain Comprehensive Temperature, Pressure, and Flow Data under Steady-State and Transient Operations
5. Develop Safe and Reliable Control Methods
6. Validate Analytical Codes
7. Gain System Operations Experience
8. Extend NASA/DOE Core Competencies in Nuclear Systems
9. Stimulate Industrial Infrastructure for Component Design and Fabrication
10. Demonstrate Manufacturing Methods
11. Obtain As-Built Mass and Cost Data
12. Provide Tangible and Measurable Technology Milestone




Notional TDU Test Layout in GRC Vacuum Facility #6

**Figure 7-12.** Technology demonstration unit test layout.

The TDU could include a high-fidelity reactor thermal simulator, which would be developed jointly by MSFC and DOE. Specific issues to be addressed would include non-fuel core materials (possibly a mix of stainless steel and superalloys), core support, core thermal hydraulics, performance, and safety. The thermal simulators would be designed to closely mimic heat from fission. Testing would measure reactor flow distribution and temperatures, and be used to benchmark design tools. Testing could also validate steady state and transient reactor module behavior, including geometric effects that could affect operations and safety. High fidelity, non-nuclear reactor module testing could increase confidence in cost, mass, and performance estimates of future flight reactors.

The proposed TDU implementation approach could include a multi-design, multi-vendor competitive development process, as shown in figure 7-13. Initially, two parallel design concepts would be pursued: liquid metal-cooled Stirling and gas-cooled Brayton. Each concept could have two vendors conducting competing power conversion conceptual design studies. At the conclusion of the conceptual designs, one vendor could be down-selected for each design concept to complete a detailed design. In parallel with the detailed design studies, NASA could conduct experimental Pathfinder tests for each concept, using existing, subscale hardware. At the conclusion of the design studies and Pathfinder tests, a single design concept, either LM-Stirling or GC-Brayton, could be selected for fabrication and test. Other component technologies, such as radiators and PMADs, could be developed separately under the advanced component technology element and incorporated into the TDU as they are completed.

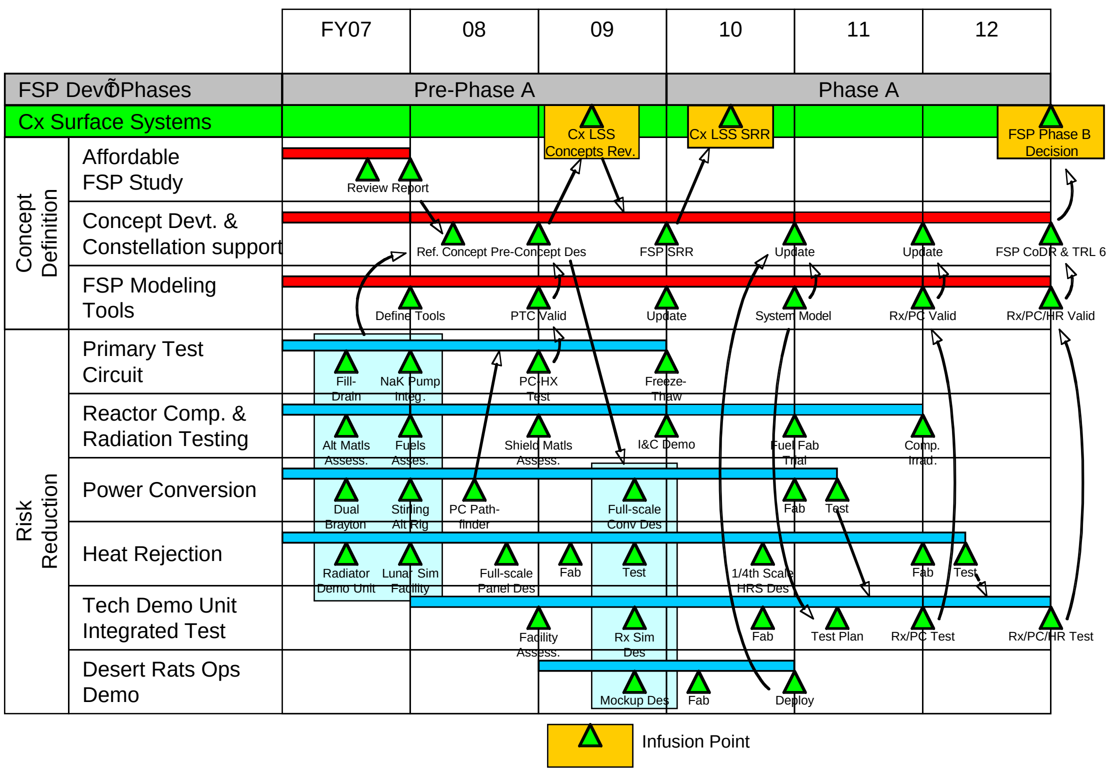



**Figure 7-13.** Technology development program.

If current funding remains steady, a development program could be started in time for use at a lunar base. Validation of autonomous operation and system lifetime on the moon would be a significant step in reducing the overall system and programmatic risks prior to use for the Mars mission.

**Isotope power**: Isotope power systems were assessed for the pressurized rover application, in particular, a DIPS. A constant power source helped to reduce nighttime battery load and array area. In the case of the smaller pressurized rover in the telecommuter option, the hybrid design of solar, battery, and isotope seemed to be a reasonable power option. Taking the example at 0.5 km/hour speed, power system mass without the DIPS was approximately 1,300 kg, and with a 2.5-kWe DIPS power system mass was about 600 kg. The isotope system also allowed sufficient reduction in array area so that the 160-$m^2$ deployable array could become a fixed, top-mounted 20-$m^2$ array. In fact, a slightly larger DIPS could eliminate the array altogether. Optimization of isotope power to battery size (for peak power demand) depends largely on the ratio of drive power to base rover load. It is recommended that once a detailed sortie is finalized, a trade could be done to minimize the isotope power level (reduced Pu) and optimized battery mass system mass.

Isotopic power systems offer continuous power much as the nuclear fission system. The practical range of the nuclear fission system is on the order of several kilowatts due to the availability of $^{238}Pu$, which is produced by neutron exposure to Neptunium-237 ($^{237}Np$). Plutonium-238 has many attractive features compared to other isotopes: lower radiation (minimal, low-mass shadow shield), high-power density, and an 87.7-year half-life. Plutonium-238 fueled the radioisotope thermoelectric generators (RTGs) that were used in all past NASA missions.

The use of radioisotopes by NASA is well established since Apollo (Apollo lunar surface experiments package (ALSEP)) and has enabled more than 30 outer-planet missions as well as the Viking landers. The latest mission, New Horizons/Pluto, used an RTG with approximately 200 We of power. The largest RTG that was produced for Ulysses and Galileo was about 300 We. Both of these missions used the GPHS-RTG. These systems have a proven, long-lifetime capability with more than 20 years of deep-space operations. While RTGs have been a workhorse for our space missions, they only have about a 4% to 6% conversion efficiency.

These systems work on converting the natural radioactive decay (of largely alpha particles) heat into an electric current. The thermoelectric devices are limited in conversion efficiency and, thus, high power systems would require large amounts of radioisotope fuel. The Savannah River facility that produced the $^{238}Pu$ has been shut down with plans to restart production at a combination of alternate facilities at future date. Therefore, there is currently a limited supply of $^{238}Pu$, and a strong competition for it to support future NASA missions.

The DOE and NASA are jointly working on an advanced Stirling radioisotope generator (ASRG) program to develop a system that would have an output of approximately 150 We. Figure 7-14 shows a rendering of the ASRG system.

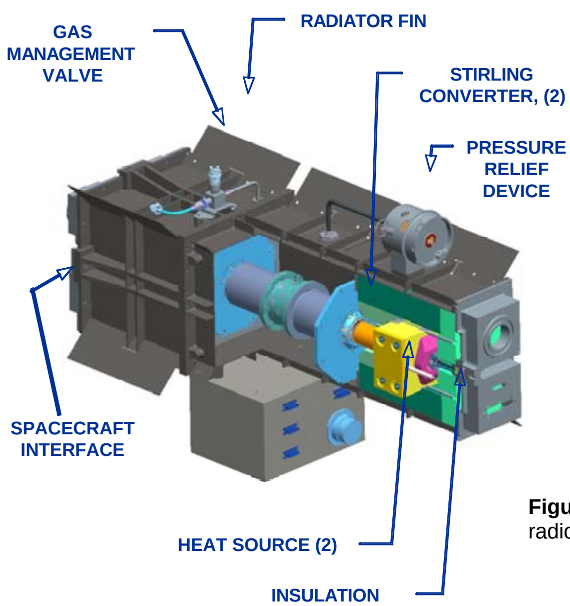



**Figure 7-14.** Advanced Stirling radioisotope generator engineering unit.

The advanced conversion technologies that are proposed here can provide a four- to five-fold increase in isotope utilization, thus drastically reducing mission cost while making prudent use of our scarce resource of isotope fuel for future missions. For example, the ASRG produces equivalent power with two GPHS modules (~1 kg plutonium dioxide ($PuO_2$)) compared to that of an RTG that would need eight GPHS modules. The GPHS module has flight heritage and the Stirling technology also has flight heritage, but as a cryocooler system and not as a power production system. Thus, a key component of the technology work is development of the linear alternator/controller that would be needed for power production.

While current Stirling converter efforts have focused on the use of a small number of GPHS modules (one to two) to provide thermal energy, there is no fundamental reason that larger numbers of GPHS modules could not be used to provide power levels in the kilowatt (electrical) range. This concept has a number of advantages compared to other power generation schemes: (1) the GPHS module is fully defined and space launch qualified, (2) emitted radiation is very low and would allow easy access and placement of the system close to the end user, (3) heat source development costs will be low, and (4) the GPHS heat source could be easily simulated with electrical heaters, thereby allowing extensive life testing in existing facilities. The negative aspects of GPHS modules are the cost and the limited supply.

Figure 7-15 shows the Stirling converter technology plan leading up to full system integration with the ASRG flight unit. As with all past RTGs, the DOE has responsibility for all isotope flight hardware.

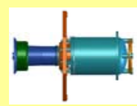
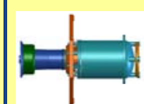
| | Technology Dev. (03-06) | | | Flight Transition (07-09) | | | Flight (10-13) |
|---|---|---|---|---|---|---|---|
| | ASC-1 | ASC-0 | ASC-1HS | ASC-E | ASC-E2 | ASC-E3 | ASRG |
| |  |  |  |  |  |  |  |
| Material | MarM-247 | | MarM-247 | | MarM-247 | MarM-247 | MarM-247 |
| | | IN718 | | IN718 | | | |
| Tasks | | | | | | | |
| | Demo high efficiency and Low Mass<br>High temp components and joints demo.<br>- Heater head<br>-Displacer | Develop Hermetic Processes<br>Identify and resolve development al issues<br>Initiated QA/process documents<br>Extended Operation, in air | Demo Hermetic Processes on High temp units<br>Improve processing (brazes, gas bearings, etc.)<br>Extended Operation, in thermal-vacuum | ASRG Integration Interfaces<br>Improve processing (i.e. closure weld, flow bench, etc.)<br>Configuration control (ERB) | Develop and Implement Quality Project Plan<br>Based on DOE Nuclear standard<br>Enhance interfaces<br>Refine high temp. processes & joints<br>Enhance reliability and manufacturability<br>Infrastructure<br>Dedicated Facility<br>Design Software<br>Laser Welder<br>Inspection (CMM)<br>Epoxy Mixer | Manufacture refinement as needed<br>Refinement to Include any new mission or generator derived requirements | NASA completes Sunpower ASC technology development and hands off to DOE for flight implementation |

**Figure 7-15** Stirling converter technology plan.

The technical approach for a high-power system is to use it as much as possible with the technologies that would be developed for ASRG and other separate high-power Stirling engine programs. While a multi-kilowatt effort does not currently exist, these systems scale well to higher power without major design differences. Figure 7-16 shows the key design parameters that are defined for the Mars multi-kilowatt system.

| Parameter | Nominal Value or Range |
|---|---|
| Net electric power | 5.0 kWe |
| Controller/PMAD efficiency<br>Stirling converter electrical output<br>Effective hot end temperature<br>Effective cold end temperature | .93<br>5.5 kWe<br>925 to 1,275 K (650 to 1,000°C)<br>>350 K |
| GPHS maximum operating temperature | 1,275 to 1,375 K (1,000 to 1,100°C) |

**Figure 7-16.** Parameter values for a high-power dynamic isotope power system.

#### *7.3.4.2 Advanced energy storage*

**Development areas for fuel cell, electrolyzer, and regenerative fuel cell**: The NASA FC development effort is focused on both primary fuel cell (PFC) power systems and RFC energy storage systems. An RFC system is a combination of a PFC and an electrolysis system, along with associated integration hardware. The FC and RFC work is categorized into six major areas: flow-through primary PEMFC development, non-flow-through primary proton exchange membrane fuel cell (PEMFC) development, high-pressure electrolysis development, RFC technology development, passive thermal development, and advanced membrane-electrode-assembly (MEA) development.

**Flow-through development**: Flow-through PEMFC technology is characterized by recirculating $H_2$ and $O_2$ reactant streams that remove product $H_2O$ that is generated at the electrode surface within each individual cell of the fuel cell stack. Recirculating reactant streams dictate the need for some type of device to initiate and sustain the recirculating flow, and for another device to separate the product $H_2O$ from the two-phase stream that is exiting the stack. In the case of existing SOTA flow-through PEMFC systems, these devices are typically active mechanical components, such as the pump and active $H_2O$ separator. Any fuel cell system using these active components bears their weight, volume, parasitic power, reliability, life, and cost penalties. This is the impetus to replace active mechanical components with passive devices in flow-through PEMFC systems, thereby minimizing the penalties that would be associated with ancillary components.

**Non-flow-through development**: A promising alternative to flow-through technology is non-flow-through PEMFC technology, in which product $H_2O$ that is generated at the electrode surface wicks across the adjacent gas cavity, through a hydrophilic membrane, into an $H_2O$ coolant cavity within each cell of the stack. There are no recirculating reactants, and, hence, no requirement for providing either recirculation or external product $H_2O$ separation from two-phase reactant streams. Therefore, there is no need for components that provide these functions, whether active or passive, and no resulting weight, volume, parasitic power, reliability, life, or cost penalties. Based on the results of testing both the flow-through and the non-flow-through systems, NASA will be in a position at the end of 2009 to down-select between these two competing PEMFC technology approaches.

**High-pressure electrolysis development**: Development of high-pressure electrolysis technology will be initiated in 2008 through solicitation of Innovative Partnership Program (IPP) and Small Business Innovative Research (SBIR) proposals. Electrolyzers are a key component of an RFC energy storage system. In this system, FCs consume reactant $H_2$ and $O_2$ gases to produce electrical power, with liquid water as a by-product. An electrolyzer reverses this reaction, consuming electrical power to break down liquid water into gaseous hydrogen ($GH_2$) and oxygen ($GO_2$). Preliminary work will be directed towards determining the maximum allowable pressures for electrolysis operation, as higher pressures result in smaller and less-massive reactant storage tanks. In determining the maximum pressure limits, special attention will be paid to the method of supplying reactant $H_2O$ to the individual electrolysis cells within a stack. Candidate options include both liquid and vapor feed, on either the $H_2$ or $O_2$ sides of the cell. Operational considerations, such as managing humidity levels in the reactant product gases, will also be taken into account.

**Regenerative fuel cell technology development**: Initial RFC development efforts in 2008 will focus on evaluating reactant management and storage issues, as any further development that is related to a specific FC or electrolysis technology would be premature until the most promising of these candidate technologies is better defined. Preliminary work will be directed towards studies of how to manage the humidity in the gas streams, and dissolved gases in the liquid streams, as the RFC cycles between FC and electrolysis modes of operation. These reactant management functions are independent of any particular FC or electrolysis stack technology. Ancillary components that perform these generic functions, as well as appropriate liquid and gas storage tanks, will be identified. After completing these initial studies, and should funding allow, NASA plans, for the 2009–2011 timeframe, to include development of appropriate reactant management ancillary components and reactant storage tanks. As fuel cell and electrolysis technology options are narrowed and the fidelity of the respective hardware increases, these technologies will become much better defined, as will the evolving RFC integration concepts that tie them together. This will allow the most appropriate reactant management and storage concepts to be pursued at increasing levels of hardware fidelity up through TRL 5, in concert with concurrent fuel cell and electrolysis development

**Passive thermal development**: Fuel cell and electrolysis stacks both require some type of thermal management. While operating, the stacks generate waste heat; while in standby or shutdown modes, the stacks must be kept warm to prevent freezing of liquid water. Typically, fuel cell and electrolysis stacks rely on a coolant cavity within each cell of the stack for thermal management, with a pumped liquid coolant loop of de-ionized $H_2O$ flowing through individual cell coolant cavities to an external heat exchanger. Because this pumped liquid coolant loop relies on an active mechanical component, it shares the same weight, volume, parasitic power, reliability, life, and cost penalties as those flow-through PEMFC active components that provide reactant recirculation and product $H_2O$ separation. ETDP work to date has been focused on developing passive thermal management technologies to replace the active pumped liquid coolant loops that are typical of most fuel cells and electrolyzers. In particular, NASA GRC has been developing both pyrolytic graphite plates and flat-plate heat pipes for direct insertion into fuel cell and electrolysis stacks, thereby replacing individual cell coolant cavities and eliminating the need for active pumped liquid coolant loops.

**Advanced membrane-electrode-assembly development**: MEAs are a key electrochemical component within all PEMFC and electrolysis stacks. The physical characteristics of the MEA, its chemical composition, and its catalyst formulations play a role in determining its electrical performance and durability. The better the electrical performance of any given MEA, the less reactants required to produce that electrical performance and the lower the mass and volume requirements for the reactants and their respective storage tanks. Following continued performance increases, which are expected to match and then exceed alkaline fuel cell (AFC) electrical performance; the best MEA technology that is available at the time will be integrated into the FC, electrolysis, and RFC development path from TRL 5 through TRL 6.

Figure 7-17 summarizes the technology tasks that would be required prior to a full RFC system development program. The component technology down-selection process would result in an optimized deign and reduce programmatic risks for the future development program. It is anticipated that a PV/RFC system performance and lifetime would be established on a lunar mission.

| Development Area | 2008 | 2009 | 2010 | 2011 | 2012 | 2013 | 2014 |
|---|---|---|---|---|---|---|---|
| Flow-Through Primary PEMFC | | | | | | | |
| Passive Component Down-Select | ✓ | | | | | | |
| Active TRL 5 | | ✓ | | | | | |
| | | | | | | | |
| Non-Flow-Through Primary PEMFC | | | | | | | |
| Active TRL 5 | | ✓ | | | | | |
| | | | | | | | |
| Primary PEMFC | | | | | | | |
| Flow-Through vs. Non-Flow-Through | | | | | | | |
| Down-Select | | ✓ | | | | | |
| Achieve TRL 6 | | | | | ✓ | | |
| | | | | | | | |
| High-Pressure Electrolysis | | | | | | | |
| Achieve TRL 4 | | ✓ | | | | | |
| Achieve TRL 5 | | | | ✓ | | | |
| | | | | | | | |
| RFC Technology | | | | | | | |
| Achieve TRL 6 | | | | | | | ✓ |
| | | | | | | | |
| Passive Thermal | | | | | | | |
| Fuel Cell Insertion | | ✓ | | | ✓ | | |
| Electrolysis Insertion | | | ✓ | | | | |
| RFC Insertion | | | | | | | ✓ |
| | | | | | | | |
| Advanced MEAs | | | | | | | |
| Fuel Cell Insertion | | ✓ | | | ✓ | | |
| Electrolysis Insertion | | | ✓ | | | | |
| RFC Insertion | | | | | | | ✓ |

**Figure 7-17.** Cell and regenerative fuel cell technology progression.

### 7.3.5 Planetary protection

Official NASA policies on planetary protection comply with the guidelines that were set out by COSPAR, which is a consultative to the United Nations Committee on the Peaceful Uses of Outer Space. COSPAR maintains guidelines on planetary protection in compliance with Article IX of the 1967 Outer Space Treaty, which requires signatory parties to conduct planetary missions so as not to jeopardize the potential for future science, or to cause harmful contamination of the Earth by returned material. NASA planetary protection requirements specify protocols to minimize the probability of transporting terrestrial organisms to locations on Mars where they could jeopardize future missions to explore for life or its chemical precursors (forward contamination), and to prevent the release of putative martian organisms from returned materials into the Earth biosphere (backward contamination). NASA policy also requires a designated planetary protection officer who would certify that NASA missions launched to Mars comply with NASA planetary protection requirements as well as COSPAR planetary protection guidelines.

NASA policy, which is based on COSPAR guidelines, specifies a scale of five different categories of planetary missions, ranging from fly-bys to sample return to Earth. Increasing levels of stringency on measures that would be taken to minimize forward- and back-contamination are required as a mission is assigned to a higher category. The highest level rating of any Mars mission thus far flown is Category IV (direct contact lander); in this category, active sterilization measures must be taken to limit the bioload that a lander could potentially transport forward to contaminate Mars. The magnitude of the allowable probable bioload that is carried on the lander takes into account the possibility of survival and growth of a terrestrial organism in the martian environment; locations on Mars where there is the possibility of coming into contact with liquid water have the most restrictive requirements on bioload. Missions that return samples of Mars material to Earth are, by definition, Category V missions. In addition to the cleanliness requirements of Category IV missions, Category V missions have additional requirements that are intended to limit the possibility of inadvertent back-contamination of Earth. There is currently a relatively well-understood set of practices and procedures in the robotic exploration community for assuring compliance with NASA requirements and COSPAR guidelines on planetary protection, including the use of approved materials, components, and sterilization technologies.

NASA currently has official requirements documents that dictate planetary protection requirements for robotic missions; however, the official NASA planetary protection policy also applies to human crewed missions, and an official requirements document is in preparation for when human missions become feasible. It is likely that the consensus requirements for planetary protection of Mars will continue to evolve over time, as more is understood about the planet itself as a result of continuing scientific exploration. For example, in 2002 COSPAR divided Category IV for Mars into three subcategories, in which “Category IVc” applies to missions that are entering a “special region,” which is defined as “a region in which microorganisms from Earth are likely to propagate, or a region which is likely to have a high potential for the existence of extant martian organisms.” A further refinement of this definition is currently under consideration by COSPAR, by which special regions are specified according to the parameters of temperature (>–25$^{o}$C) and water activity (>0.5). Any hardware contacting a special region is subject to the same high standards of cleanliness and sterilization that were met by the Viking landers. Although no locations are currently known to meet the parametric definition of a “special region,” the subsurface of Mars, as well as the surface features suggesting a reasonable probability that $H_2O$ may be present, such as the erosional “gullies” and their associated “pasted-on terrain,” will be protected as special regions until data indicate otherwise.

Some of these regions are potentially of highest interest for locating landing zones and habitats in human exploration scenarios because large, persistent bodies of $H_2O$ or ice could be important resources for in-situ utilization as well as targets of high value for scientific exploration. At the same time, it is unlikely that humans could inhabit a Mars base for hundreds of days at a time without inadvertently leaking terrestrial microorganisms (from spacesuits, habitat air locks, and the like) or becoming contaminated by martian materials (due to inhalation of martian dust, etc.).

Previous considerations on planetary protection for human missions (see section 3.8) concluded that most of the potential conflicts between human exploration and planetary protection have technological or operational solutions. Potential targets for human exploration could be certified as “zones of minimum biological risk” (terminology proposed by the Space Studies Board) based on data that would be obtained by precursor missions prior to acceptance as potential landing sites. Human habitation sites should be located at a sufficient distance from special regions to prevent contamination, as determined based on improvements in our understanding of transport phenomena on Mars. Subsurface exploration would be restricted to specially sterilized, perhaps teleoperated, robotics until an improved understanding of those environments might support (or perhaps not) altering access restrictions. Any of these options would likely require one or more precursor missions to validate procedures and, ideally, the return of surface samples for detailed analyses in laboratories on Earth. Reconciling the practical implications of human exploration with policies and requirements for planetary protection (forward and backward) is of ongoing concern to planetary protection practitioners as well as to advocates of human exploration, and the various approaches that have been proposed will continue to be considered and refined.

Specific technologies that will be essential to permit human activities on Mars include: closed-loop life support capabilities that minimize the amount of material that would be released into the martian environment and remove or kill microbial contamination that might be present; capabilities for cleaning and/or sterilizing hardware that would be used to access the martian surface and subsurface, e.g., similar to those that are used in polar and glacial drilling projects on Earth today; capabilities for isolating humans from uncharacterized martian materials; and capabilities for monitoring crew health and microbial populations throughout the mission so that medical records are available for comparison with post-exposure data.

### 7.3.6 Surface systems critical challenges and technology needs summary

Surface systems challenges and testing venues are addressed in tables 7-6 and 7-7, respectively.

**Table 7-6.** Surface Systems Challenges

| **Current Knowledge or Capability Gaps** |
|---|
| • Obtain better understanding of the environment of Mars including dust, dust storms, and dust accumulation as well as the radiation environment |
| **Technology Needs** |
| Advanced Habitation |
| • Lightweight structural approaches that would package within aeroshell constraints |
| • Systems that would provide radiation protection without significantly increasing habitation system mass |
| • Capable of performing deployment, assembly, and checkout autonomously and/or robotically |
| • Systems that are highly reliable and maintainable |
| • Designs to minimize release of consumables and contaminants to the environment |
| EVA and Surface Mobility |
| • Systems with lower weight than lunar counterparts |
| • Capable of using locally produced $O_2$ and $H_2O$ |
| • Robust and capable of protecting the crew from the dangers of sharp rocks and objects as well as from operating in a dusty environment |
| • Thermal control for the martian environment |
| • Highly dexterous systems that would maximize mobility |
| Subsurface Access |
| • Drill cutting transport methods (both mud and mud-less techniques) |
| • Mars-applicable bit designs |
| • Lightweight hole casing for deep drilling |
| • Core handling and planetary protection |
| Surface Power |
| • Robust, reliable nuclear power generation of a minimum of 30 kWe continuous |
| • Safe, reliable backup power for contingencies |

**Table 7-7.** Surface Systems Testing Venues

| Venue | Advanced Habitation | EVA & Surface Mobility | Subsurface Access | Surface Power | |
|---|---|---|---|---|---|
| **A. Earth Surface** | • Technology development<br>• Demonstration of system performance and operational concepts in Earth analogs | • Technology development<br>• Demonstration of system performance and operational concepts in Earth analogs | • Drill performance in Mars analog environment<br>• Operational scenario testing including human/robotic partnerships | • Technology development<br>• Laboratory demonstration of system performance | • |
| **B. Earth Atmosphere** | • N/A | • N/A | • N/A | • N/A | • |
| **C. Earth Orbit** | • Inflation techniques and system performance<br>• Zero-g performance | • EVA system zero-g performance | • N/A | • N/A | • |
| **D. Lunar Transit and Orbit** | • N/A | • N/A | • N/A | • N/A | • |
| **E. Lunar Surface** | • Mars prototype system performance, reliability, and maintainability<br>• Deployment, assembly, and checkout – autonomously and/or robotically<br>• Integrated radiation protection | • Mars prototype system performance, reliability, and maintainability | • Mars prototype system performance, reliability, and maintainability<br>• Human-deployed and -operated deep drilling | • Mars prototype reactor performance, reliability, and maintainability<br>• Autonomous operations | • |
| **F. Deep Space Transit** | • Demonstration of system performance for long periods in deep space | • EVA system zero-g performance | • N/A | • N/A | • |
| **G. Mars Orbit** | • N/A | • N/A | • N/A | • N/A | • |
| **H. Mars Atmosphere** | • N/A | • N/A | • N/A | • N/A | • |
| **I. Mars Surface** | • N/A | • N/A | • Robotic demonstration of shallow (10-m) drilling<br>• Environmental knowledge at drill site is critical. This includes rock size and lithology, distribution, geophysical methods to suggest that desired subsurface target (e.g., ground ice, subsurface liquid water) exists | • Possible testing of subscale reactor deployment, thermal control, and all operations | • |

## 7.4 Cross-cutting Systems and Miscellaneous Needs

### 7.4.1 In-situ resource utilization

The use of non-terrestrial resources can provide substantial benefits to a variety of future space activities by dramatically reducing the amount of material that must be transported from Earth to a planetary surface. ISRU is a critical component of long-term, largely self-sufficient outpost operations. By extracting and processing local resources to obtain or make $O_2$, $H_2O$, $CH_4$, and buffer gas consumables for life support, EVAs, and ascent propulsion, significant mass reductions or increased payload to the Mars surface is possible. There are two primary resources of interest on Mars: the atmosphere, which is mostly made up of $CO_2$ (95.5%), $N_2$ (2.7%), and Ar (1.6%), and $H_2O$, which exists in the top meter of Mars soil. Since NASA and international robotic missions have shown that $H_2O$ can be found globally across the Mars surface, and Mars

Odyssey mission data suggest that regions with up to 8% to 10% $H_2O$ by mass can be found in the top 1 m of the martian surface, the extraction and use of Mars can be found suggests both benefits and challenges to future missions.

Numerous studies have been performed evaluating Mars atmospheric resource collection and processing techniques, and significant development work on technologies and systems that are associated with atmospheric resource collection and processing was performed from 1995 to 2001 in support of Mars ISRU robotic precursor missions and a possible MSR mission using ISRU as a precursor to human missions and validation of ISRU in a mission-critical role. This work included both ground-development activities as well as development of robotic precursor payload hardware. However, these efforts were cancelled after the Mars 98 Surveyor lander mission failure, so development challenges remain. Since past Mars ISRU development efforts only considered Mars atmospheric resources, new work is required for Mars surface $H_2O$ excavation and extraction.

#### *7.4.1.1 Mars atmospheric collection and separation*

The significant benefit of Mars atmosphere $CO_2$, $N_2$, and Ar as a resource is that it is available globally at known concentrations. Nitrogen and Ar are very good buffer gases for crew breathing as well as purge gases for science experiments. Carbon dioxide is a good source of both $O_2$ and C for the production of $O_2$, $CH_4$, and other hydrocarbons that may be of interest. Pressurized gases (whether bulk atmosphere or separated $CO_2$, $N_2$, or Ar) are also beneficial for inflating habitats and structures as well as for use in cleaning dust from surfaces and sensitive areas. However, the Mars atmosphere is at low pressure (~0.1 psia) at the surface.

Before Mars atmospheric $CO_2$ can be used or processed, it must be collected, separated, and pressurized; typically at or above Earth ambient pressure (>14.7 psia), to increase the efficiency of $CO_2$ processing concepts. The challenges that must be addressed for Mars ISRU include the following:

- Operation for >300 days without crew support
- Separation of atmospheric dust from the Mars atmosphere (filter clogging as well as downstream process impacts)
- Mass and power-efficient process to acquire the low-pressure gases and large pressure increase that would be required (>100:1 pressure ratio)
- Mass and power-efficient separation of atmospheric gas constituents ($CO_2$, $N_2$, and Ar)

Three primary methods for $CO_2$ collection and pressurization have been evaluated: mechanical pumps, microchannel adsorption, and cryogenic separation ($CO_2$ freezing).

**Mechanical pumps**: To deliver $CO_2$ to a processing unit, a >100:1 compression ratio is required from Mars atmospheric pressure to $CO_2$ processing unit pressure. Since most mechanical pumps are efficient up to around 10:1 compression ratios, a two-stage compressor is required. All studies on Mars ISRU that have evaluated two-stage compressors that are based on terrestrial pumps have identified the following drawbacks to this method:

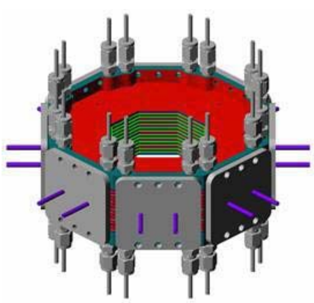
- Two-stage (100:1) mechanical compressors are extremely massive for ISRU systems for human missions (several hundred kilograms)
- Two-stage (100:1) mechanical compressors require a great deal of power for ISRU systems for human missions (several kilowatts)
- Two-stage (100:1) mechanical compressors require inter-stage cooling of the gases, thereby increasing unit complexity and mass
- Mechanical compressors that interface with the Mars atmosphere pose a risk of failure due to dust getting into moving parts and affecting lubricants

Therefore, no further development efforts on mechanical pumps have been pursued, and work has focused on the two other methods of $CO_2$ collection that were mentioned.

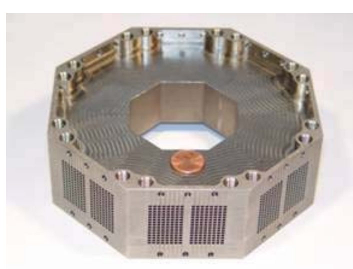
**Micro-channel adsorption (gas/solid or gas/liquid interaction)**: Work on $CO_2$ acquisition that was based on adsorption was originally based on similar work for removal of $CO_2$ from air for life support systems. Large beds were used with long periods of time for adsorption. ARC, JPL, JSC, and Lockheed Martin Aeronautics (LMA) all developed different-sized beds for Mars atmospheric $CO_2$ collection based on these concepts and using the Mars day/night temperature swing to help promote the collection of $CO_2$ and minimize the power that would be associated with temperature swing adsorption beds. Since solar powered missions could only process $CO_2$ during daylight hours when large amounts of power were available, this approach seemed perfect. While the process was shown to be feasible, there were several major drawbacks:

- Adsorption beds were large and massive to collect the required $CO_2$ at night for processing during the day.
- Large beds had high pressure drops to flow and also had limited diffusion of $CO_2$ to adsorption material. This made the beds inefficient, as compared to theoretical storage capability, and also required the pumps to continuously flow Mars atmosphere through the beds with a very low starting inlet pressure
- Large beds required significant thermal energy to desorb $CO_2$ during daytime processing operations. While some thermal energy could be recuperated from $CO_2$ and $H_2O$ processing, significant electrical energy was still required

Based on both the success and drawbacks that would be associated with large $CO_2$ adsorption beds, a different approach was investigated. Instead of large beds with long adsorption/desorption cycles, smaller beds with rapid adsorption/desorption cycles that would minimize pressure drop and diffusion limited capacity loss were considered. Using the new concept of microchannel chemical and thermal systems (MCATS), the DOE/PNNL was contracted to examine, develop, and test rapid-cycle adsorption (gas/solid) and adsorption (gas/liquid) methods. The concept involved eight separate beds in different parts of the adsorption and desorption cycle sequence. MCATS technology minimized the mass and diffusion distance of each bed while maximizing thermal recuperation by transferring heat from adsorbing to desorbing beds in the cycle sequence. PNNL first evaluated single-sorption pump cells to examine different solid and liquid sorbent materials as well as to understand the impact of different adsorption/desorption cycle times on mass, power, and product separation/delivery efficiency. Mars atmosphere capture requirements that were based on a system to support an MSR mission using ISRU were selected because previous studies showed that an ISRU system for an MSR mission using solar power was approximately one-fifth scale compared to an ISRU system for a human mission using nuclear power. Solid material adsorption was found to be the best approach, and further work was then performed on developing and testing an eight-cell sorption pump to provide a 10:1 pressure ratio, with the goal of using two pumps to obtain the 100:1 desired pressure ratio from the Mars inlet pressure to $CO_2$ processing pressure (see figure 7-18). Based on single- and eight-cell sorption pump tests, significant mass savings over large-sorption bed pumps would be achievable. Calculations from experimental data show that the mass of zeolite (adsorption material) that would be needed for a 2-minute cycle process is approximately1 kg, whereas it is >500 kg for a cycle that would be based on the martian day (~24.6-hour diurnal cycle). While significant work was accomplished by PNNL, the program never was completed due to budget cuts. Therefore, optimization of flow rate and cycle time for the eight-cell sorption pump, due to issues with the fluid heat exchanger pump, could not be completed. PNNL also evaluated the eight-cell sorption pump for possible use in life support system $CO_2$ removal from air by examining 10%/90% $CO_2$/$N_2$ mixtures. While results were very encouraging, issues with some co-adsorption of $N_2$ could not be completely eliminated with the available funding. ISRU atmosphere collection subsystem mass, power, and volume in section 6.2 are based on analytical and experimental results from this effort. Further work in this area is highly recommended.



(a)



(b)

(c)

**(a)** Model of the Completed Sorption Pump Structure with Tube Stubs for External Gas and HX Fluid Connections; **(b)** Photograph of a Diffusion-Bonded Titanium Eight-Cell Sorption Pump; **(c)** Photograph of the Eight-Cell Sorption Pump Experimental System

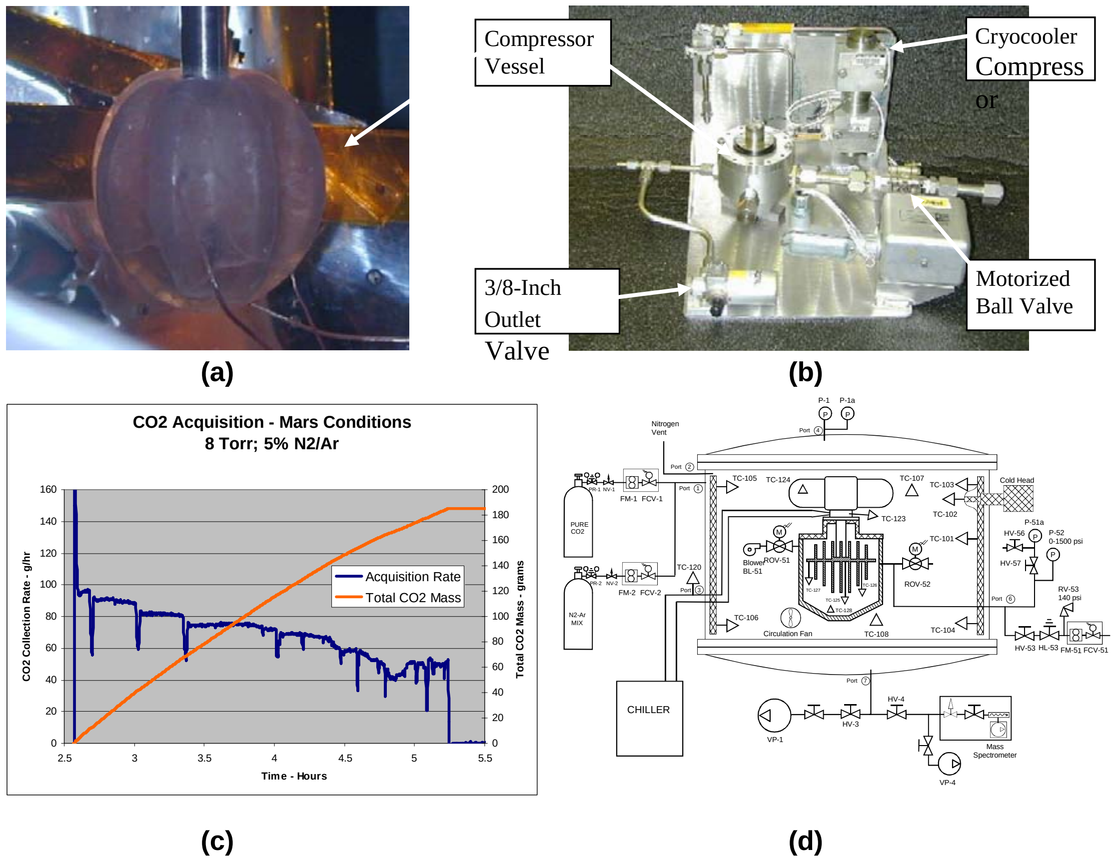
**Figure 7-18.** Solid material adsorption.

**Cryogenic separation (carbon dioxide freezer)**. During early adsorption bed/pump testing at JSC, the Mars atmospheric nighttime temperature (150 K/–123°C) was simulated by using warmed liquid nitrogen ($LN_2$) that was passed through a coil surrounding the adsorption pump. Because a temperature sensor in the adsorption pump was used to control the gas/$LN_2$ coolant feed-rate to achieve Mars temperatures, it was determined that some $CO_2$ froze in the adsorption pump when desorption results exceeded the theoretical adsorption capability of the material that was used. Upon closer examination, however, it was found that the temperature difference between solid and gaseous $CO_2$ at Mars atmospheric pressure was slightly lower than Mars nighttime temperatures such that it was theorized that with a low-power cryocooler, the $CO_2$ in the martian atmosphere could be frozen out, leaving other atmospheric constituents ($N_2$ and Ar) for further collection and separation. Work to evaluate this concept was performed by LMA (under contract to NASA) and Pioneer Astronautics (under an SBIR).

A $CO_2$ freezer (solidification pump) requires active cooling to lower the atmospheric gas temperature in the pump to below 150 K (–123°C). $CO_2$ will solidify at this temperature and at Mars pressures. The frozen $CO_2$ can then be heated in a controlled volume to supply $CO_2$ at any desired inlet pressure for subsequent processing. The solidification pump is attractive as an acquisition concept for future human and Mars robotic missions because it allows small-volume and high-pressure $CO_2$ delivery, and can potentially simplify system design and development costs by sharing cryocooler hardware with the \$O_2$/$CH_4$ propellant liquefaction and storage system.

This method of cryogenic $CO_2$ compression was demonstrated in the Mars Simulation Chamber at the Lockheed Martin facility in Denver, Colorado. This project examined the major solidification pump subcomponents, such as the acquisition pressure vessel, circulation blower, and heat exchanger configurations, as well as a variety of operating scenarios (see figure 7-19). To prevent building up a diffusion barrier of residual $N_2$ and Ar after $CO_2$ has been solidified, a bypass flow is established with a low-power blower to purge non-condensable gases from the freeze chamber.

Initial testing of the frozen $CO_2$ compression system demonstrated the viability of the concept as a low-power, high-pressure supply of $CO_2$. Operation of a complete system was demonstrated in this program, supplying over 80 grams in a 7-hour cycle. Supply pressure closely followed the vapor pressure of the imposed external temperature, generally over 200 psia (up to 300 psia). System operation is straightforward and results are very repeatable. Results of these initial tests provide an excellent basis for the design of larger compressor systems that would use highly power/cooling efficient cryocoolers and a two-chamber operation where heat that was rejected from the cryocooler that was used to cool one chamber is used to heat up the frozen $CO_2$ in the second chamber. The ISRU atmosphere collection subsystem mass, power, and volume that were described in section 6.2 are based on analytical and experimental results from this effort. While this concept was not baselined for the Mars ISRU system that was addressed in section 6.2, significant benefits to this approach should be evaluated, and further work in this area is highly recommended.




**(a)** Solid CO2 growth on copper heat exchanger; **(b)** Compressor system mounting plate; **(c)** CO2 capture as a function of time ; **(d)** Test system schematic

**Figure 7-19.** Cryogenic carbon dioxide compression.

### *7.4.1.2 Mars water collection and separation*

Robotic missions to Mars have shown that $H_2O$ can be found globally across the martian surface. In equatorial regions (±30°), the Viking missions measured 1% to 3% $H_2O$ by mass, and Mars Odyssey mission data suggest that there are regions with up to 8% to 10% $H_2O$ by mass in the top 1 m. Mars Odyssey data also suggest that the subsurface ice table may be within the top few meters in some localities in the mid latitudes (40° to 55°), near-surface subsurface ice tables may be widely prevalent at the high latitudes (55°–70°), and >50% water ice by mass is at or near the surface in the polar regions (+70°). Experts believe that hydrated minerals and gypsum may be widely available at $H_2O$ concentrations between 20% and 30% at sites of science exploration interest in the equatorial region as well. The extraction of $H_2O$ from martian soil raises several challenges that must be addressed, including the following:

- Operation for >300 days without crew support (life and autonomy concerns)
- Wide global and potentially local variations in $H_2O$ content, form, and depth (availability and content uncertainty concerns)
- Excavation of surface material down to 1.0 m in varying soil properties and rock distributions (soil penetration and motor/actuator force concerns)
- Movement of excavated material in bins, hoppers, and augers involving materials with varying $H_2O$ content and soil properties (bridging and clogging concerns)
- Thermal processing of Mars soil to extract $H_2O$ (system mass, power, processing time, and clogging concerns)
- Contaminant removal from extracted $H_2O$ ($H_2O$ processing corrosion/life concerns)

To mitigate these challenges, work is required to both better understand Mars soil/ $H_2O$ properties (global and local $H_2O$ concentrations and distribution, soil properties, and potential contaminants), and develop hardware to excavate, process, and extract $H_2O$ from Mars surface material.

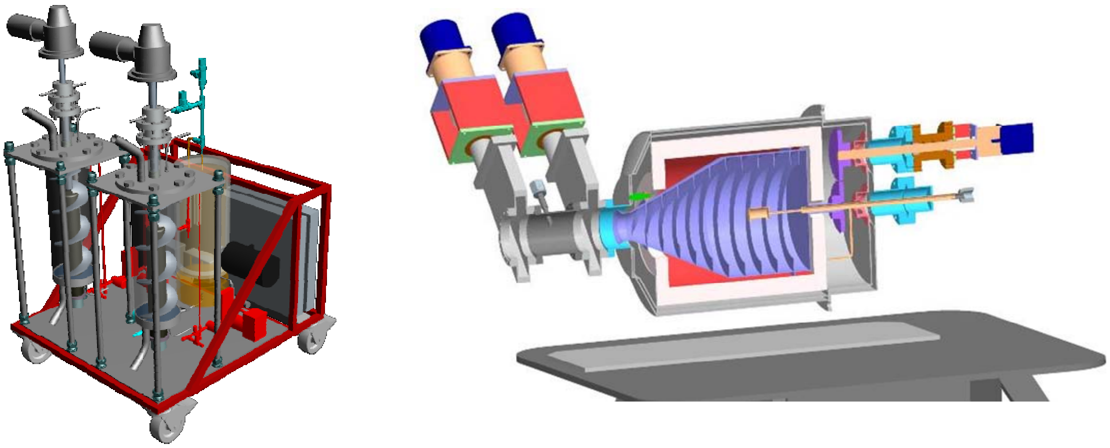
**Mars soil excavation and transport**. Excavation and material property experience and data that are derived from use of the arm/scoop on Mars Viking, wheel/soil interaction behavior and experience from the Sojourner and MERs, and potentially new excavation and surface material property data in the polar region from the Phoenix lander arm/scoop are good starting points for future Mars excavation and material transport development efforts. During the brief Mars DRM 5.0 study, a good amount of time was spent on trying to better understand and define Mars soil parameters for possible locations of scientific and human exploration interest including: $H_2O$ content, cohesion, internal friction angle, bulk density, compressive strength, and tool-soil adhesion. Because efforts are under way in modeling lunar material behavior for development of lunar excavation tools under the ETDP ISRU Project (see figure 7-20), computer models and laboratory test experience combined with estimated Mars soil parameters were used in section 6.2 to estimate Mars excavation tool and rover size, mass, and power. For Mars $H_2O$ extraction to be used for propellant production for crewed missions, excavation and soil processing systems must be designed to excavate and process 77 kg (3% $H_2O$ content) to 30 kg (8% $H_2O$ content) every hour. While lunar material is much more compacted and dry compared to expected Mars soil, there are enough similarities in design and operation that experience that is gained from current lunar excavation and material transport would benefit future Mar soil excavation and transport. Breadboard lunar ISRU subsystems are in the process of being built for the first end-to-end excavation-to-$O_2$ extraction and storage demonstration to occur at a field analog site on Mauna Kea, Hawaii in November 2008. Hardware is being built at a scale that is equivalent to making 250 to 1,000 kg of $O_2$ per year on the moon (70% operating time). Assuming a 1% extraction efficiency that is based on polar highland material properties and these production rates, current hardware is being designed to process 15 to 20 kg of material per hour, which is reasonably close to the 30 kg/hour that is associated with the 8% $H_2O$ content processing rate for Mars ISRU for propellant production. However, even the small excavators that are being examined are only used a fraction of the time, so higher excavation rates are possible. In preparing for this field demonstration, it was determined that while the volcanic material on Mauna Kea is a reasonable simulant for lunar processing, the relatively high $H_2O$ content can cause bridging and clogging of processes that are designed for low-$H_2O$-content materials. Lessons that are learned from evaluating lunar ISRU subsystems with simulants containing varying $H_2O$ amounts may be a good starting point for lunar/Mars hardware system compatibility evaluations. However, increased $H_2O$ content, gypsum/clay material properties, and deeper excavation on Mars for $H_2O$ extraction compared to excavation for lunar $O_2$ production are new challenges that must be addressed.

**(a)** **(b)** **(c)** **(d)**

(a) Surveyor arm/scoop replica and blade interaction tests to anchor lunar excavation models; **(b)** Small lunar rover scoop excavator/dump concept (Cratos); **(c)** Small lunar rover bucketwheel/dump concept (LMA); **(d)** Blade-soil interaction testing

**Figure 7-20.** Mars soil excavation and transport.

**Mars soil processing to extract water**. Mars soil processing hardware development and flight experience has been limited to small, single-use ovens that process only small amounts of soil material. For Mars $H_2O$ extraction to be useful for crewed missions, processing systems must be designed to operate for hundreds of cycles. Because efforts are under way in modeling and developing breadboard reactors for $O_2$ extraction from lunar regolith and feeding regolith into and out of regolith processing systems under the ETDP ISRU Project (see figure 7-21), computer models and laboratory test experience combined with estimated Mars soil and $H_2O$ content parameters were used in section 6.2 to estimate Mars $H_2O$ extraction system mass, volume, and power.



**(a)** ROxygen hydrogen reduction reactor with internal auger and water separation freezer; **(b)** PILOT rotating hydrogen reduction reactor with wall auger

**Figure 7-21.** Mars soil processing to extract water.

The lowest-risk lunar $O_2$ extraction from regolith process that is being developed is $H_2$ reduction from regolith. Because lunar regolith is a poor conductor of heat, this process uses augers or rotating reactors combined with $GH_2$ fluidization techniques to react $H_2$ with regolith to produce $H_2O$ at temperatures below 1,000°C. This $H_2O$ is then condensed and separated from the unreacted $H_2$ for subsequent electrolysis, and the $H_2$ is recycled for further $O_2$ extraction. To extract $H_2O$ from Mars soil, a similar process is envisioned that uses a carrier gas to remove $H_2O$ vapor from heated Mars soil at temperatures around 600°C, with subsequent $H_2O$ vapor removal and recycling of the carrier gas. While lunar material is much drier compared to the expected martian soil, there are enough similarities in design and operation of the lunar $H_2$ reduction reactors that experience that is gained from this effort should benefit future Mar soil processing and $H_2O$ extraction/separation development efforts. Breadboard lunar ISRU subsystems are in the process of being built for the first end-to-end excavation-to-$O_2$ extraction and storage demonstration to occur at a field analog site on Mauna Kea, Hawaii in November 2008. Hardware is being built at a scale that is equivalent to making 250 to 1,000 kg of $O_2$ per year on the moon. Assuming a 1% extraction efficiency that is based on polar highland material properties and these production rates, hardware is being designed to process 15 to 20 kg of material per hour, which is reasonably close to the 30 kg/hour that is associated with 8% $H_2O$ content processing rate for Mars ISRU for propellant production. In preparing for this demonstration, it was determined that while volcanic material on Mauna Kea is a reasonable simulant for lunar processing, the relatively high $H_2O$ content can cause bridging and clogging of processes that are designed for low-$H_2O$-content materials. As discussed in the section on Mars soil excavation, the lessons that would learned from evaluating lunar ISRU subsystems with simulants containing varying $H_2O$ amounts may be a good starting point for lunar/Mars hardware system compatibility evaluations.

#### *7.4.1.3 Carbon dioxide processing*

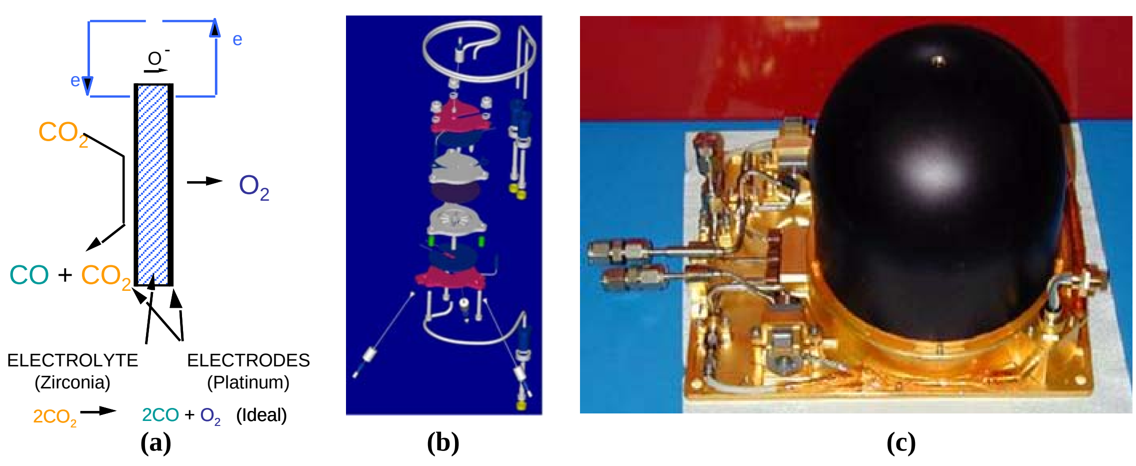
Conversion of atmospheric $CO_2$ into $O_2$ can be performed in a number of different ways, depending on the resources that are available and the products that are desired. The three processes that have been examined the most due to process simplicity or commonality with life support systems are: $CO_2$ electrolysis, Sabatier conversion of $CO_2$ to $CH_4$ and $H_2O$ (with subsequent $H_2O$ electrolysis), and RWGS conversion of $CO_2$ to CO and $H_2O$ (with subsequent $H_2O$ electrolysis). For both Sabatier and RWGS conversion of $CO_2$, $H_2$ is required. In the case of $O_2$ production using RWGS, the $H_2$ that is required is obtained from the subsequent $H_2O$ electrolysis, so $H_2$ is recycled. In the case of $O_2$ production using Sabatier, only half of the $H_2$ that is needed is recovered from the subsequent $H_2O$ electrolysis process. It should be noted that while other technologies and methods for $CO_2$ processing are possible and have been evaluated, these processes were considered be at too low of a TRL to be evaluated at a system level for mission applicability. These alternative low-TRL technologies include: molten carbonate electrolysis, non-aqueous electrolysis of $CO_2$, ionic liquid electrolysis, liquid $CO_2$ electrolysis, and lower-temperature mobile oxide ceramics.

**Carbon dioxide electrolysis (solid oxide carbon dioxide electrolysis)**. Carbon dioxide electrolysis involves the breakdown (or dissociation) of $CO_2$ into CO and $O_2$. A number of different material and electrode options and methods for supplying energy disassociate the $CO_2$ molecule; these are: glow discharge, RF electromagnetic radiation, thermal, and catalytic. The preferred method is a combined thermal/catalytic reactor using yttria-stabilized Zirconia (YSZ) with platinum (or platinum alloy) catalyst/electrodes, which is commonly known as SOCE. The SOCE process is fairly simple. $CO_2$ is supplied to the solid-state ceramic reactor where energy is supplied to the gas to disassociate the $CO_2$ molecule into $O_2$ ions and CO via a platinum electrode that is applied to the surface of the YSZ. The $O_2$ ions that are produced are conducted through a YSZ membrane with a voltage potential and combine there with another $O_2$ ion on the other side of the membrane to form an $O_2$ molecule. SOCE has a history of development for terrestrial FC and $O_2$ removal from air applications (medical and military); however, SOCE has, for the most part, been excluded from consideration for life support systems due to the production of CO.

SOCE operating at the 900°C to 1,000°C temperature range have been demonstrated at a very small scale with 40% to 50% $CO_2$ conversion efficiencies. While the SOCE process and hardware are fairly simple, there are four primary challenges that need to be addressed before SOCE can be used in future missions. These are

- Need to develop a ceramic-to-metal interface (internally or at the interface with the rest of the ISRU system) that can withstand repeated thermal cycling and high-temperature exposure to $O_2$ and CO
- Electrically and thermally efficient cell stack development and packaging
- Vibration and shock insensitive design and packaging
- Carbon monoxide/$CO_2$ separation and recycling of $CO_2$ to minimize atmosphere collection due to <50% $CO_2$ conversion.

Much of the work that was performed in this area was associated with investigating various aspects of sealing and packaging for use in the Mars environment. More durable and easier to manufacture metal alloy generator assemblies and manifolding concepts were evaluated by the University of Arizona (UofA) and Allied Signal, but with limited success before funding ran out. The UofA (under NASA JSC funding) developed a single-cell SOCE unit for the Mars in-situ propellant production precursor (MIP) flight experiment (see figure 7-22), and began development of the multi-cell SOCE unit for the Production of Resources On Mars In-Situ for Exploration (PROMISE) flight experiment. Even with these challenges, the SOCE was baselined for $O_2$-only Mars ISRU due to the relative simplicity, limited number of thermal cycles that would be expected (by using nuclear power), and recent advances in solid oxide FCs for terrestrial applications. The SOCE subsystem mass, power, and volume that are described in section 6.2 are based on analytical and experimental results from all of these efforts.




**(a)** $CO_2$ electrolysis process; **(b)** MIP single-cell stack configuration; **(c)** MIP $O_2$ generation flight unit

**Figure 7-22.** Carbon dioxide electrolysis.

**Sabatier**. Sabatier reactors catalytically convert $H_2$ and $CO_2$ into $CH_4$ and $H_2O$ in a self-sustaining, exothermic reaction. The Sabatier reaction is well characterized, and significant work has occurred to develop Sabatier reactors for life support and

Mars ISRU systems. While the technology is well in hand for life support system usage, and a Sabatier reactor was recently launched to the ISS for connection to the ISS life support system, further work in development of Sabatier reactors is required if they are to be used for Mars ISRU. Sabatier reactors for life support are based catalyst-bed reactors that are operating at low pressure (~10 psia) to eliminate leakage of $H_2$ or $CH_4$ into the crew's habitable/breathing volume. This approach, while low risk, is heavier and less efficient than a Sabatier reactor that operates at higher temperatures and uses thermal recuperation methods. Two approaches were previously examined: a low-risk catalyst bed with internal heat exchanger and a higher-risk/higher-efficiency Sabatier reactor/heat exchanger that uses MCATS technology.

The low-risk catalyst bed with internal heat exchanger Sabatier reactor was designed and built at JSC (see figure 7-23(a)). The reactor incorporated innovative features, such as regenerative preheating of the inlet $H_2$ and $CO_2$ gas flow, in an attempt to better control the thermal profile along the length of the reactor. Because the Sabatier reaction is temperature and product $H_2O$ sensitive, to achieve >99% conversion of $CO_2$, reactors are designed to have high temperatures near the inlet to achieve the bulk of the conversion quickly and then the temperature is reduced down the reactor length to complete conversion of the remaining $CO_2$. The reactor was designed to allow easy access to the internal components for configuration changes, and also has enhanced thermal data gathering features to better determine the internal thermal gradient down the axis of the reactor. The reactor was sized for a Mars robotic sample return mission that is also compliant with on-orbit life support system operating requirements. The planned range of testing parameters includes inlet flow rates of 50 to 250 grams per hour (g/hr) of $CO_2$ at inlet pressures of 10 to 50 psia. While the reactor was built, funding was not available for subsequent testing.


**(a)** **(b)** **(c)**

**(a)** JSC Sabatier reactor for ISRU; **(b)** KSC RWGS testbed; **(c)** PNNL MCATS Sabatier & RWGS reactors

**Figure 7-23.** Reverse water gas shift.

The advantage that MCATS technology has for chemical processing systems, such as Sabatier, over conventional catalyst bed and heat exchanger technology is that its small channel size allows for rapid heat and mass transport, it has non-equilibrium chemical products, its surface forces dominate over gravity forces, and it has a high productivity per unit volume. This allows reactors and systems to have high thermal integration and energy efficiency and allows separate units to be assembled into systems for redundancy and rapid recycling or multi-step processing. The challenge to MCATS technology is that conventional catalyst and reaction rate behavior is no longer applicable, so some trial-and-error testing is required to optimize reactions, heat exchange, and product separation/reactant recycling into compact reactors and heat exchange units. For the MCATS Sabatier reactor, PNNL took advantage of the high heat and mass transfer of microchannels, used a very active catalyst, and used an integrated heat exchanger to provide a method for accurate temperature control. By varying catalysts, $CO_2$:$H_2$ inlet ratios, flow rates, and reactor temperatures, PNNL could evaluate $CO_2$ conversion percentage, $H_2$ conversion percentage, catalyst $CH_4$ production sensitivity (vs. other hydrocarbon products), and $VH_4$ production rates. The final one-eighth-scale Sabatier reactor that was built and tested was an order of magnitude smaller in volume and mass compared to conventional Sabatier reactors. It was found that as $CO_2$ conversion increases, the production of $CH_4$ decreases; so from a system perspective, the final reactor that was built and tested maximized $CH_4$ production at an 83% $CO_2$ conversion efficiency and a 350°C operating temperature. The final one-eighth-scale Sabatier reactor that was built and tested was an order of magnitude smaller in volume and mass compared to conventional Sabatier reactors (see figure 7-

23(c)). Sabatier reactor subsystem mass, power, and volume in section 6.2 are based on analytical and experimental results from this effort.

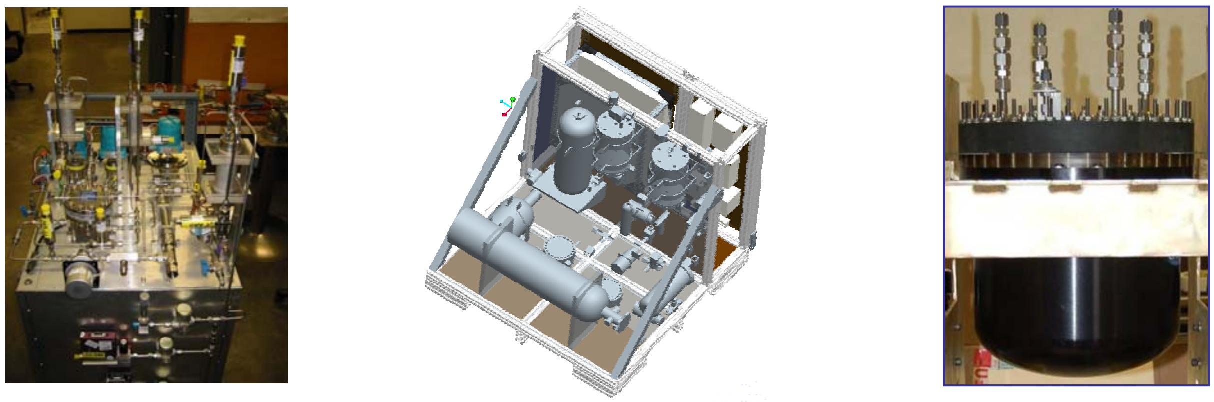
**Reverse water gas shift**. Like Sabatier, the RWGS process has been well known since the mid-1800s. The RWGS reactor operates by taking $H_2$ and $CO_2$ and combining them in an endothermic catalytic reaction ($\Delta H = +9$ kcal/mole) to form $H_2O$ and CO. The process, which uses a copper catalyst, seems to be most efficient at about 400°C. Like SOCE, RWGS has, for the most part, been excluded from consideration for life support systems due to the production of CO. Like Sabatier, two approaches were previously examined: a low-risk catalyst bed with recirculation and a higher-risk/higher-efficiency RWGS reactor/heat exchanger that uses MCATS technology.

Building on the work of Pioneer Astronautics, KSC fabricated a testbed that allowed further development of RWGS technologies (see figure 7-23(b)) The testbed explored technologies to improve its efficiency, provide efficient gas separation methodologies, and develop autonomous process control technology. The testbed achieved $H_2$ conversions on the order of the 98%; however, each pass through the reactor converted only approximately 20% of the reactants. Therefore, a recycling system that separates the $H_2O$ and CO from the unreacted $CO_2$ and $H_2$ is required. This is the major drawback to the catalyst bed RWGS reactor approach.

As mentioned under micro-channel $CO_2$ adsorption and Sabatier reactors, MCATS technology has significant advantages over conventional separation and chemical processing system designs. Further MCATS advantages over conventional RWGS catalystic bed reactors includes rapid removal of $H_2O$ vapor from product streams to help shift the non-equilibrium RWGS reaction to the left to produce more $H_2O$, and the ability to perform multiple RWGS reaction stages in a low mass/compact unit to increase $CO_2$ conversion well above 20% for conventional, single-pass catalyst reactors. The challenges that are to be addressed by PNNL in developing an MCATS RWGS reactor include relatively low catalytic activity at high throughput (conversion efficiency), catalyst stability at high temperatures (up to 800°C), the need for high catalyst loading, and reduced pressure drop. PNNL first examined different catalysts in single micro-channels, and then down-selected a few options for evaluation of multichannel performance before designing the integrated RWGS/heat recuperation reactor unit. The most promising RWGS catalyst from this evaluation was 6% Ru/$ZrO_2$-CeO that is coated on FeCrAlY foam (an inter-metallic alloy from Porvair), which was used in subsequent reactor unit development and testing. Like the MCATS Sabatier reactor project, the MCATS RWGS reactor has been one-eighth scale since integration, and combined operation with the MCATS Sabatier reactor was also part of the contracted effort. The RWGS reactor unit that was built (see figure 7-23(c)) was tested over a variety of temperatures, $H_2$-to-$CO_2$ ratios, and contact times. The $H_2$-to-$CO_2$ ratio was limited to 1.2:1 to prevent the possibility of coking the reactor at reduced $H_2$ levels. Carbon dioxide conversion from 40% to over 60% was achieved with selectivity to CO of >99.99% and minimal pressure drop. Performance was much better than expected, and equated to almost one-half-scale production rates. Like the MCATS Sabatier reactor, the MCATS RWGS reactor was an order-of-magnitude smaller and lighter than conventional RWGS reactors. RWGS reactor subsystem mass, power, and volume, which are discussed in section 6.2, are based on analytical and experimental results from this effort.

#### *7.4.1.4 Water processing*

Water processing involves $H_2O$ separation for reactant streams, $H_2O$ cleanup to remove contaminants, and $H_2O$ electrolysis to convert $H_2O$ into $O_2$ and $H_2$. Water processing is required for both $O_2$-only production on Mars using RWGS as well as $O_2$ and $CH_4$ production on Mars using Sabatier. Water electrolysis is a mature technology that is used in multiple terrestrial applications (such as $O_2$ production on U.S. Navy submarines and as a laboratory device to provide $H_2$); a $H_2O$ electrolysis unit was recently delivered to ISS for use in life support system operations. However, these systems are relatively low pressure, and challenges still exist for efficient $H_2O$ product separation from product streams and $H_2O$ cleanup before electrolysis. A moderate-pressure anode-feed $H_2O$ electrolysis system (150 psia) with integrated $H_2/CH_4$ separation unit was built at JSC for Mars ISRU but never completed testing. The two $H_2O$ electrolyzers from this previous effort are now undergoing repackaging as part of the end-to-end lunar $O_2$ extraction from regolith system development effort that is now ongoing in the ETDP ISRU Project. A on water processing unit is also being built by LMA that uses a slightly higher-pressure cathode-feed on electrolysis unit (400 psi) for this lunar ISRU $O_2$ system development and demonstration effort (see figure 7-24). Water processing subsystem mass, power, and volume, which are addressed in section 6.2, are based on analytical and experimental results from this effort.



**(a)** **(b)** **(c)**

(a) Water processing unit built at JSC for Mars ISRU; **(b)** Water processing unit being built at JSC for lunar oxygen production; **(c)** Giner water electrolysis unit for LMA lunar oxygen production

**Figure 7-24.** Water processing.

#### *7.4.1.5 System integration and testing*

Development of individual subsystems for the Mars ISRU, which is listed above, is important. However, equally important is integration and optimization of these technologies and subsystems into a complete end-to-end plant. Besides the RWGS/$H_2O$ electrolysis subsystem that was built and tested by KSC (see figure 7-23(b)), a first generation Sabatier/$H_2O$ electrolysis unit with Mars atmospheric $CO_2$ sorption pump acquisition and $O_2$ liquefaction and storage unit was built and tested under simulated Mars surface pressure, temperature, and atmospheric constituent conditions at JSC in 1998. Based on lessons that were learned from building and testing this integrated system, a second-generation system using a $CO_2$ freezer, an advanced Sabatier reactor, and an advanced $H_2O$ processor (each was discussed previously) was under development when the project was cancelled. Besides these ground development units, a very subscale ISRU robotic precursor flight demonstration unit (the MIP) was built, tested, and certified for flight as part of the Mars 01 Surveyor Lander mission (see figure 7-25). Mars environment simulation chambers at JSC are no longer operational.

**(a)** **(b)** **(c)** **(d)**

**(a)** 1st generation sabatier/water electrolysis testbed; **(b)** Sabatier/water electrolysis testbed in 20 ft diameter Mars environment simulation chamber at JSC; **(c)** MIP ISRU flight demonstration unit; **(d)** MIP in 5 ft diameter Mars environment simulation chamber at JSC

**Figure 7-25.** Mars in-situ propellant production hardware.

#### *7.4.1.6 Conclusion and recommendations*

The use of non-terrestrial resources can substantially benefit a variety of future space activities by dramatically reducing the amount of material that must be transported from Earth to a planetary surface. ISRU is a critical component of long-term, largely self-sufficient outpost operations. For human missions to Mars, ISRU systems for propellant production may be enabling.

A significant amount of work has been performed to develop technologies and hardware for Mars atmosphere-based ISRU for $O_2$ and propellant production. Work is just beginning on soil excavation and processing technologies as part of the lunar

ISRU technology, hardware, and system development project in the ESMD ETDP that is highly relevant to Mars soil excavation and soil processing to extract $H_2O$ hardware. Processes for extracting resources from the Mars atmosphere are believed to be much simpler and have, therefore, been baselined because $O_2$ could be produced through straightforward reduction of the atmosphere, which is primarily $CO_2$. While potentially very beneficial, actual hardware and system technology and hardware needs for Mars soil excavation and processing require further study. Collaboration with science mission soil and $H_2O$ Characterization missions on Mars, such as Phoenix, and future missions will be extremely valuable as well. Table 7-8 summarizes the current TRL and how much time it would take to reach TRL 6 if a highly focused effort were started.

It is recommended that although ISRU options were baselined in the study, the different technology and subsystem options that were identified in this section be developed to TRL 6, and that these technologies be tested in integrated systems before downselection for final flight development. It is also recommended that future Mars science robotic missions and ESMD $H_2O$ resource assessment and processing development activities be coordinated, thus leading to a possible joint robotic precursor mission.

**Table 7-8.** In-Situ Resource Utilization Technology Readiness

| Technology | Current TRL | Dev. Time to TRL 6 (years)* | Dev. For Lunar Campaign (Y/N) |
|---|---|---|---|
| Mars Atm. Acquisition and Separation | | | |
| $CO_2$ Freezing | 3–4 | 2 | N |
| Microchannel adsorption bed | 3 | 2 | N (for ISRU) |
| Mars Water Acquisition | | | |
| Excavation unit | 2–3 | 3 | Y (similar) |
| Hauler/dumper mechanism | 2–3 | 3 | Y (not yet) |
| Surface mobility unit | 5–9 | | Y |
| Soil processing reactor | 2–3 | 4 | Y |
| Carbon Dioxide Processing | | | |
| $CO_2$ electrolysis | 4 | 3.5 | N (for ISRU) |
| Microchannel Sabatier reactor | 4 | 2 | N (for ISRU) |
| Microchannel RWGS reactor | 4 | 2 | N (for ISRU) |
| Integrated Sabatier/RWGS | 4 | 1.5 | N |
| Water Processing | | | |
| Water electrolysis | 6–9 | 1.5 | Y |
| Microchannel water/gas separator | 4 | 2 | N (for ISRU) |

*This is for TRL at the individual component/subsystem level. Further time is required for system development.

### *7.4.1.7 References*

[1] Section 6.2 In-Situ Consumable Production.
[2] Sanders, G. B., Trevathan, J. R., Kaplan, D. I., Peters, T. A., Baird, R. S., Cook, J. S., McClean M. L., Pauly, K., "In-Situ Consumable Production (ISCP) for Mars Robotic and Human Exploration at the NASA/Johnson Space Center", 00ICES-168, 30th International Conference on Environmental Systems, Toulouse, France, July, 2000.
[3] Sanders, G. B., Trevathan, J. R., Peters, T. A., Baird, R. S., Larson, W. E., Lueck, D. E., Parrish, C. F., "Preparing for Robotic & Human Exploration Missions Which Incorporate In-Situ Resource Utilization", AIAA 2000-5317, Space 2000 Conference and Exposition. Long Beach, CA., Sept., 2000.
[4] Brooks, et. al, "Microchannel In Situ Propellant Production System Project Final Report", NASA JSC Contract NNJ05HB58C by Battelle-Pacific Northwest National Laboratory, Richland, WA., March, 2006.
[5] Larson, W. E., Lueck, D. E., Parrish, C. F., Sanders, G. B., Trevathan, J. R., Peters, T. A., Baird, R. S., "Technology Development for Human Exploration Beyond Low Earth Orbit in the New Millennium", IAA-01-IAA.13.3.07, 52nd International Astronautical Congress, Toulouse, France, Oct., 2001.
[6] Rapp, D., "Accessible Water on Mars", JPL Contract D-31343-Rev.7, July, 2006.
[7] Perko, H. A., Nelson, J. D., Green, J R., "Mars Soil Mechanical Properties and Suitability of Mars Soil Simulants", Journal of Aerospace Engineering, July, 2006.
[8] Perko, H. A., Nelson, J. D., Green, J R., "Mars Soil Mechanics Investigation", CUS Project 53-1947, NASA Contranct/Grant NAG5-11280 Supp1, 2004

### 7.4.2 Cryogenic fluid management

Cryogenic fluid management (CFM) is an important technical area that is needed for the successful development of Mars architectures. The first and foremost challenge is the storability of $LH_2$ and $LO_2$ propellants for long durations. Note that the longest flight of stored cyrogens is Titan Centaur-5, where the propellants were stored in orbit for 9 hours. These propellants have very low boiling points – well below the environment temperatures of Earth orbit, Mars transit, Mars orbit, or Mars surface – as such, the tanks must be regularly vented to prevent overpressurization. Such venting will cause unacceptable propellant losses for the long-duration missions being that are considered to Mars. In lieu of venting, active cooling or refrigeration can be integrated into the tanks to preserve propellants. Present lunar architectures (NASA, 2005) [27] will help pave the way, if they maintain the 95- or 14-day loiter requirement. Present thought, however, is that loiter time will be reduced to perhaps 4 days, minimizing the thermal control requirements and necessary technology development of advanced cryogenic storage concepts. Most aspects of long-term cryogenic storage technology nevertheless exist at some state, mainly from the development of advanced dewars for life support and satellite instrument purposes. Central to space telescope performance is the ability to mitigate molecular movement, which is possible only at cryogenic temperatures that are achieved with cryogenics. Thick MLI systems have been applied to cryogenic dewars; also, active cooling components such as cryocoolers have been integrated. These are rapidly advancing in capability and state-of-the-art, and are gradually replaced cryogenic dewars for space telescope applications. Still, these developments have not been applied to cryogenic propellant applications, particularly to the size of the tanks that would be needed for this Mars architecture. Furthermore, there have been no significant advances in $LH_2$ temperature cryocoolers near the sizes that would be needed for zero boil-off cryogen storage.

Besides the thermal control aspects, other CFM development issues that would ensure safe and reliable cryogenic storage and supply to the propulsion systems include: liquid acquisition and transfer, to ensure vapor-free propellant supply to the engine as well as to a second tank, and mass gauging, to ensure reliable propellant quantity information. These three cryogenic areas have been under development for the present CFM program, which is part of the ETDP. The purpose of that effort is to mitigate the substantial risks that are associated with cryogenic propellants in support of lunar mission architectures by the year 2011. Note that all the technical elements under development by CFM are applicable to Mars mission scenarios.

Other systems that would benefit from advanced cryogenic propellant storage systems include ISRU, if in-situ cryogenic propellant production is part of it. Advanced storage systems include large-scale, flight-rated cryocooler development, which is central to large-scale liquefaction efforts. Furthermore, this same development would benefit concepts for efficient long-duration storage of FC reactants.

#### *7.4.2.1 Cryogenic fluid management goals*

The top-level exploration cryogenic fluid management goals are:

1. *Long-term (5 years+) in-space storage of cryogenic propellants and fuel cell reactants.* The authors of the NASA Vision for Space Exploration (NASA, 2004) [28] state that “To conduct an effective and exciting program of exploration, discover, we must overcome the limitations of space, time, and energy, as well as various space hazards” (pg 15). Applying this to cryogenic propellants means that we must develop technologies and capabilities to store cryogens in space for very long durations to ensure significant mission flexibility and operability. Today’s active cooling devices, cryocoolers, have 10- to 15-year design lifetimes. Such long-term storage is enabled through active cooling systems that are combined with a robust insulation design.

2. *Long-term (5 years+) Mar surface storage and liquefaction of cryogenic propellants and fuel cell reactants.* Primary to high-performance, long-term thermal control is excellent insulation, which includes radiation foils under a vacuum jacket on each cryogenic tank; solar protection via articulating shades to reduce the exposure temperatures while on the surface, which is key to minimizing the active cooling system requirements; and a robust active cooling system that will be coupled to the vacuum-jacketed tanks to remove the heat that enters those tanks. The propellants will be cooled to eliminate boil-off with a large, distributed cooling system. It is anticipated that reverse Brayton cycle cryocoolers will be used for this function. In addition, a large liquefaction station will be

[27] “NASA's Exploration Systems Architecture Study -- Final Report.” NASA TM 2005-215062, November 2005.
[28] The Vision for Space Exploration, National Aeronautics and Space Administration, February 2004.

required to liquefy gaseous reactants or propellants, in support of the in-situ propellant production process. Such a system will be staged to provide added capacity at liquid methane ($LCH_4$) and $LO_2$ temperatures. The envisioned tank storage elements all have proven long-term lifetimes.

3. *Substantially reduce mass and volume resources of the cryogenic stages*. Studies have shown that active cooling from zero boil-off and reduced boil-off systems will begin to reduce mass and volume after several months in orbit for $H_2$ systems, and after several weeks in orbit for $LO_2$ systems. This will profoundly affect the Mars missions that are being considered. Without active cooling, the tanks must be oversized to accommodate tank boil-off, which will more that double the size of the tanks for the mission scenarios that are under consideration. Other advanced concepts to reduce mass and volume include advanced passive approaches, including sunshades, planet shades, and surface shades, which have been shown to reduce the MLI temperature substantially, along with the corresponding active cooling requirements. Other important elements are the use of common insulation to create a cold box that will reduce insulation mass and reduce expose to the warm environment. Note that the best cryogenic storage options begin with large tanks, as these will reduce tank support and plumbing heat while consolidating the cold cryogens together, thereby reducing the exposed tank surface area.

4. *Provide vapor-free liquid cryogen supply to engines*. Incorporate liquid acquisition devices to fill a start basket or other device to ensure that propellant delivery to the engine is vapor free. If stage propulsion is used several times, liquid acquisition devices have been shown to reduce mass when compared to traditional settling and venting.

5. *Ensure cryogenic propellant mass quantity error is less than 1*%. Provide accurate cryogenic mass gauging to ensure that propellant mass quantity information is available at all times. This is especially significant for vented tanks or for stages that have multiple propellant burns, since propellant inventory information will be less accurate.

**Thermal control issues regarding Mars environment**

Primary to high-performance insulation for propellant preservation is a hard vacuum to protect from exposure to convective and gas conduction heating while on the surface. Such insulation must also protect the tanks from solar, albedo, and IR heating. Mars has a 10-torr atmosphere, while hard vacuum is considered to be $10^{-6}$ torr. Accordingly, a vacuum jacket must be considered. Such a jacket would be prohibitively heavy if it were designed for the 760-torr atmosphere on Earth, unless a load-bearing insulation system was used that is not presently available. However, since a vacuum jacket is not required while on the Earth (the cryogenic propellant tanks can be topped up until lift-off), a Mars-specific vacuum jacket could be incorporated, provided it was vented while on Earth and was exposed to space vacuum in flight. On evacuation, the jacket design must provide for vacuum isolation prior to entry into the Mars atmosphere.

The solar exposure on Mars will also be significant. The daytime peak air temperature is approximately 260 K, while the night temperature is approximately 200 K (Mars Pathfinder-measured air temperatures). The surface of Mars is warmer; its equatorial temperature is approximately 300 K. These can be reduced via articulating shades to block sunshine from reaching the cryogenic tanks, by protecting the tanks from surface exposure (using a surface-insulating cover), and by exposing warm surfaces to deep-space temperatures. The shades will have several layers of aluminized Mylar with a “v”-shaped orientation to isolate a view of space on the interior Mylar sheets. This will reduce the surface temperature of the Mylar, which will, in turn, reduce the tank outer vacuum shell temperature, thereby reducing heat into the tank. This will serve to reduce the exposure temperature significantly while on the surface, which is key to minimizing the active cooling system requirements. It is likely that these sunshades would decrease the peak daytime temperatures significantly below the peak temperature of 260 K. A model of shades on the lunar surface, which estimates that the peak surface temperatures near the poles could be reduced from 210 K to 105 K, as shown in figure 7-26. If one extrapolates that to a Mars mission, the temperatures will also be reduced, although perhaps not as much since Mars is farther from the sun.

| Shield Configuration | T, K |
|---|---|
| With out shields | 210 |
| With shields 3 sides | 171 |
| With shields 4 sides | 146 |
| V-Groove Shield | 105 |

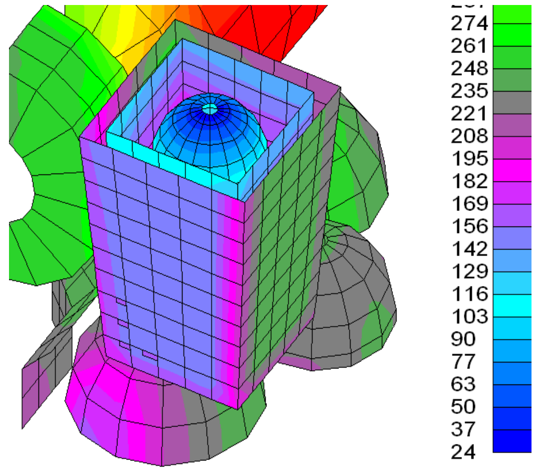



**Figure 7-26.** Cryogenic tank temperature with thermal shades on lunar surface.

There is a possibility that the Mars surface could be used to help protect the cryogenic tanks if mounds of Mars dirt were formed. The soil within the mounded area would not be exposed to portions of the solar day cycle, thus functioning like a sunshade. It could also be used for heat rejection from cryocoolers. This could be coupled with a counter-flow heat exchanger and other elements to provide geothermal heat to the habitat or for other systems or components.

Another Mars environment factor for cryogenic thermal control is dust. Insulation performance is predicted based on the surface temperatures on the outer vacuum jacket, which is a function of its absorptivity and emmissivity. Absorptivity will degrade with dust. Lunar mission data showed that a dusty surface had an absorptivity of 0.4, a substantial degradation from 0.1. This property can also be degraded by oxidation, although oxidation is expected to be minimal on Mars. It is possible that a blower would be required to occasionally remove dust from these cryogenic tank surfaces to reduce the absorptivity that will reduce the surface temperature. Note that radiation heat transfer is proportional to the fourth power of this temperature.

Note also that the CFM program has several near-term planned activities that will advance the technology of shade and active distributed cooling elements.

#### *7.4.2.2 Cryogenic fluid management system descriptions*

The CFM systems that are anticipated for Mars missions include advanced thermal control and associated tank pressure control, liquid acquisition, mass gauging, and fluid transfer. The first and most important system to be discussed is thermal control.

**Thermal control**

Mars missions that are under consideration would benefit from incorporation of high-$I_{sp}$ propellants such as $LH_2$ and $LO_2$, even with their accompanying boil-off losses that are necessary to maintain a steady tank pressure. A recent paper (Plachta 2007) [29] addresses a cryogenic propellant boil-off reduction system that would minimize or eliminate boil-off. Concepts to do so were considered under the In-space Cryogenic Propellant Depot Project (Howell) [30]. Specific to this project was an investigation of cryocooler integration concepts for relatively large depot-sized propellant tanks, which are relevant to Mars mission-size tanks. One concept proved promising; it served to efficiently move heat to the cryocooler even over long distances via a compressed He loop. Analysis shows that when compared to passive-only cryogenic storage, the cryogenic boil-off reduction system begins to reduce system mass if durations are as low as 40 days for $LH_2$ and 14 days for $LO_2$. In addition, a method of cooling $LH_2$ tanks is presented that precludes the development issues that are associated with $LH_2$ temperature cryocoolers.

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[29] Plachta, et. al., Cryogenic Boil-off Reduction System, 2007, presented at the 2007 Cryogenic Engineering Conference, Chattanooga, TN, to be published in "Advances in Cryogenic Engineering," Volume 53.
[30] Howell, to be published.

Earlier studies (Plachta 2002) [31] indicate that application of active cooling, coupled with an adequate passive thermal control system, shows mass savings when compared to a benchmark passive-only systems for missions that are longer than 1 week for $LO_2$ and several months for $LH_2$. However, that analysis omitted the design integration of the cryocooler to the propellant tank. Only simple temperature gradient integration losses were assumed, which is not adequate if the cryocooler is located more than a short distance from the propellant tank. There also was no concept design included for cooling over large areas. Rather, it was assumed that the heat would enter the tank and be removed by a cryocooler that was coupled to a heat exchanger that, in turn, was submerged in the tank.

The boil-off reduction system design was performed under the In-Space Cryogenic Propellant Depot Project, which envisioned long-term storage as a requirement for large depot-sized tanks and had similar diameters to those planned for the Mars architectures. A literature search was performed to identify the interface components concepts that might be incorporated into the thermal control design of a properly insulated and actively cooled cryogenic tank, specifically to thermally connect cryocoolers to propellant tanks and radiators. Concepts considered included wide-area heat pipes, thermal switches and diodes, distributed cooling, using a micro-electromechanical system (MEMS)-type approach in a gas-manifolded cryogenic cooling system, as well as cooled shield designs. The selected design is a closed-loop compressed He system, the cryogenic boil-off reduction system (CBRS).

This design is a distributed cooling system that uses a chilled gaseous helium (GHe) loop, as shown in figure 7-27. This design is attractive for several reasons. At high pressures, He has a high conductivity and density, giving it convective heat transfer properties similar to those of a liquid; and because of its extremely low boiling point, He can be used to cool at $LH_2$ temperatures. Since the heat leak through MLI is relatively small, small gas mass flows can be used to remove heat. Therefore, small-diameter tubing can be used without causing excessive pressure drops. As the tubing is small, the shield it is mounted on is also thin: This thin shield can simply be Al foil that is reinforced to prevent tearing. The small tubing that is coupled to a thin foil shield yields a mass penalty of about 10 layers of MLI. This is useful to cool the $LH_2$ MLI, which serves to decrease its boil-off rate substantially. Slightly larger-diameter tubes are needed to remove the heat leak that enters the tank from penetrations, such as plumbing and supports. These have inherently greater heat fluxes when compared to the MLI. Those tubes, however, do not have to cover large surface areas.

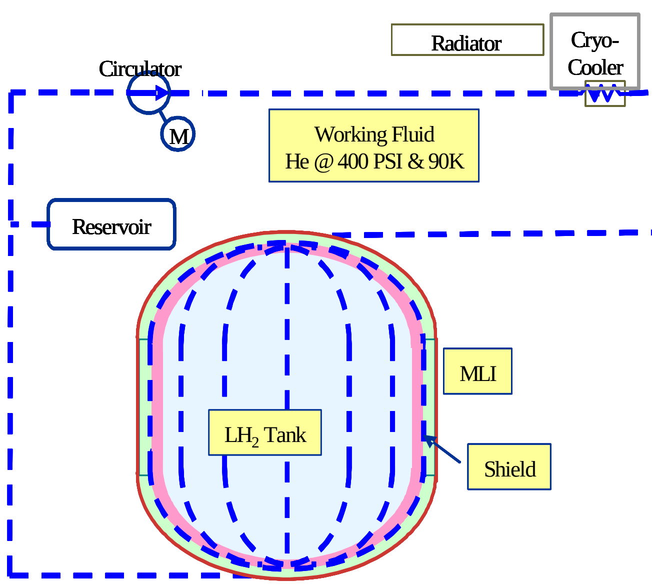



**Figure 7-27.** Schematic of cryogenic boil-off reduction system for a single liquid hydrogen tank.

[31]Plachta, D.W., Kittel, P.: "An Update to Zero Boil-Off Cryogenic Propellant Storage Analysis Applied to Upper Stages or Depots in a LEO Environment," NASA TM 2003-211691, July 2002

***Parametric trades***

All of the results that are shown use the thermal storage mass. For the passive cases, this is the sum of tank, insulation, propellant, boil-off, and tank/insulation growth masses. For the CBRS cases, it is the sum of tank, insulation, propellant, reduced boil-off ($LH_2$ only), cryocooler, solar array, radiator, and integration component masses including the shield, tube, circulator, He reservoir, and heat exchanger. Redundant components were assumed for the cryocooler, circulator, tubing, and reservoir.

The first graph that is shown in figure 7-28 includes passive and CBRS thermal storage mass predictions for $H_2$ as a function of storage duration, which is shown in days. This was done at a tank diameter of 6 m. From this graph and the additional runs with the other propellant and tank sizes that were considered, the equal mass lines were constructed; these are shown in figure 7-29. These are the storage durations where the passive and CBRS masses are equal; durations longer than these are predicted to reduce mass for CBRS; shorter durations would benefit if passive storage were used.

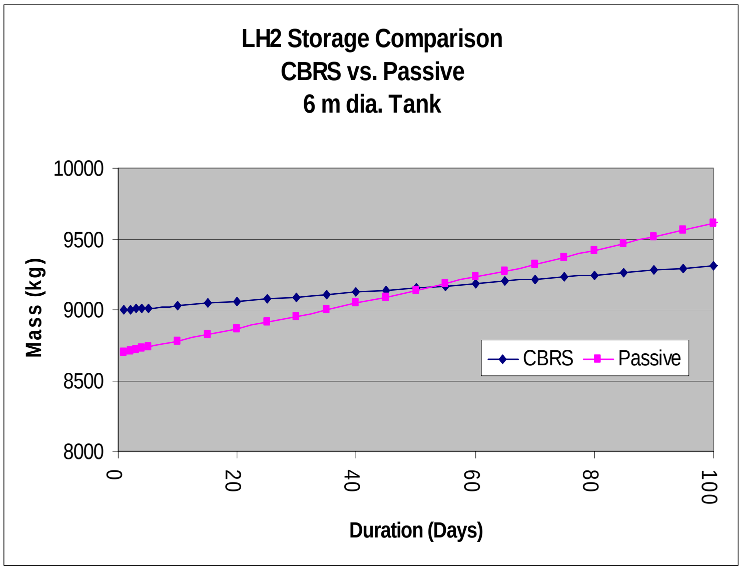



**Figure 7-28.** Cryogenic boil-off-off reduction system performance.

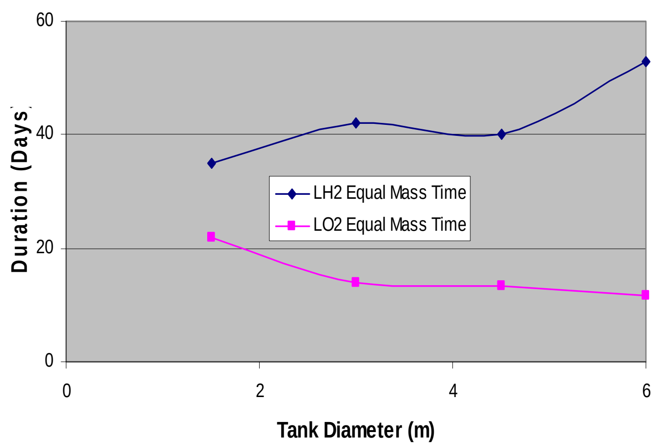



**Figure 7-29.** Mission duration when passive and boil-off reduction system masses are equal.

***Discussion***

With the reduced boil-off approach for $H_2$, which uses a higher temperature and higher Carnot efficiency cryocooler at roughly $LO_2$ temperatures coupled to the distributed cooling system, we find that the duration when active cooling should be considered is approximately 60 days for the larger tanks that would be needed for Mars architectures. Note that this was not compared to a zero boil-off system for $LH_2$, and at this point it is uncertain which system is projected to reduce mass more. It is possible that the CBRS system, if it were coupled with a shade, would be more efficient. Note that its inherent advantage is that the technology readiness of this is much higher than that of $LH_2$ cryocoolers, which are projected to be very expensive.

Notice that the equal mass curve in figure 7-30 changes slope for the $LH_2$ case just before the 4.5-m-diameter tank size. This is the point at which the tank heating rate increases above 20 W, and the model switches from the Brake (Brake, et al 2002) [32] cryocooler relationship to the Air Force Research Laboratory (AFRL) cryocooler sizing relationship. This increases the cryocooler mass.

An important assumption that was used was that the cryocooler is located 10 m from the propellant tank to accommodate configuration issues. This provides for significant flexibility in its location. Note that the model is relatively unaffected by this assumption. For instance, if the length is 15 m, the He temperature increases by a relatively small 0.07 K if the tube size is not changed (it decreases otherwise). This design aspect can be taken advantage of by plumbing the He tubing around several tanks together. The fuel ($LH_2$ and/or $LCH_4$) and oxidizer tanks, if any, could share the same system, as could a chilled He bottle for pressurant supply. This reduces mass further as it decreases the overall number of CBRS components and creates an economy of scale for other components.

As such, this concept could be used with a common insulation system for a grouping of tanks when applied to a Mars lander. CBRS could provide a chilled He enclosure that would reduce the total insulation mass as well as the number of per-tank seams and penetrations. The cooling system for each tank would be replaced with a single, but redundant, CBRS. The different zones could be manifolded together with valving to ensure proper control.

**Thermal control summary**

The indications are that optimized passive thermal control approaches, such as thick MLI and solar/planet shade development, when coupled with the CBRS, is projected to reduce mass and offers tremendous flexibility for the Mars architectures. While it is unclear at this time whether this system saves mass over the zero boil-off approach for $LH_2$ storage, it is certain that it would require much less development, money, and time to prepare this system for flight primarily because it does not require flight $LH_2$ temperature cryocoolers, which have not been developed and would be expensive.

***Liquid acquisition devices and transfer***

When transferring propellant in space from tank to tank or from tank to engine inlet, it is most efficient to transfer single-phase liquid. In the gravity field of Earth or under acceleration, propellant transfer is fairly simple. Single-phase fluid is transferred to a propellant engine or tank by opening a valve at the bottom of the propellant tank. In low gravity, where fluid is not centered over a tank outlet, withdrawing single-phase fluid becomes a challenge. A variety of propellant management devices (PMDs) are required to ensure single-phase flow. One type of PMD, the screen-channel-type liquid acquisition device (LAD), uses capillary flow and surface tension to acquire liquid. Capillary-flow LADs have been well characterized for storable propellants (i.e., propellants that are liquids at room temperature) for in-space propulsion needs, but there are little available data for cryogenic LADs. Over the past several years, GRC and MSFC have been working on a joint program to advance the knowledge of the thermodynamic and fluid behavior of cryogenic LADs.

Present technology efforts are being performed under the ETDP in support of lunar architectures. An experimental program was conducted to determine bubble point and the outflow characteristics of a screen channel LAD for $LO_2$. Two different Dutch twill screen meshes were studied – 200 × 1400 and 325 × 2300. In a bubble point test, the pressure retention capability of a screen is demonstrated. Test configurations are shown in figures 7-30 and 7-31.

[32]Walter Brake, H.J.M., Wiegerinck, G.F.M.: "Low-Power Cryocooler Survey," *Cryogenics* 42 (2002) pp. 705-718.

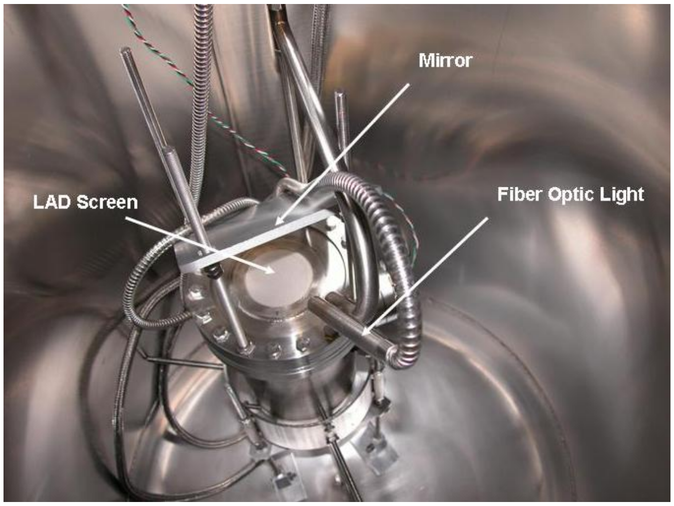



**Figure 7-30.** Bubble-point test hardware installed in test dewar.


**Figure 7-31.** Screen channel test hardware installed in test dewar.

Another experiment was conducted to evaluate the surface tension LAD technology that would be applied to cryogenic tanks for in-space use. Although the principles of surface tension are the same for both storable and cryogenic liquids, and the LAD components (screens, sponges, vanes, etc.) should be similar, additional challenges are inherent in the cryogen application. The most significant challenge arises from the problems that would be caused by heat leakage into the cryogen.

**Future directions**

To develop LAD devices for Mars missions, fundamental data on various potential propellants (including $LH_2$, $LCH_4$, and $LO_2$) need to be developed for external and autogenous pressurants. Other issues that need to be considered include:

- Determine the effects of vibration on the performance of the LADs
- Determine the effect of autogenous/non-autogenous pressurants on the LADs
- Develop/validate robust analytical models to predict the performance of cryogenic LADS
- Develop and test flight LAD designs to validate LAD manufacturing techniques and LAD performance at flow rates that are expected for a depot
- Develop/validate techniques to minimize vaporization inside the LAD channel that is caused by incident heating through tank wall/lines and changes in tank pressure. These techniques could include the use of heat sinks from recirculators, active cryocoolers, or gas in the thermodynamic vent
- Develop a low-g experiment to anchor models with flight data
- Continue gathering fundamental data on various potential propellants (including $LH_2$, $LCH_4$, and $LO_2$)
- Perform preliminary heat entrapment testing with $LN_2$

Note that part of this development has begun as part of the CFM program. In addition, cryogenic LADs are projected to be incorporated in the lunar lander program, which will provide key flight data.

**Transfer**

Effective and regular human exploration of the solar system will require refueling in microgravity with large quantities of cryogenic propellants. Figure 7-32 shows an artist's concept of space refueling in LEO. Although modest quantities of non-cryogenic propellant are transferred routinely, the unique properties of cryogens and the much larger quantities of required propellant make the prior techniques ineffective for microgravity cryogenic refueling (Chato 2000) [33].

Cryogenic fluid transfer allows for the reuse of hardware that is already in orbit, thus reducing lift mass. Stages that are initially filled on orbit can eliminate many of the systems and structural mass that would be required to support and maintain cryogens on the launch pad. Transfer allows tanks on the mission vehicle to be insulated only for the mission rather than the months of loiter that would be required to assemble a stage on orbit. The valving and hardware for cryogenic transfer should

[33]Chato, D. J; "Technologies for Refueling Spacecraft On-Orbit" AIAA 2000-5107, September 2000.

be substantially simpler and safer than drop tank designs (10- to 20-cm disconnects that can be checked for leakage vs. 43 cm shuttle external tank (ET)-style valves that must seal instantaneously when the pyrotechnic devices fire to drop the tanks). (Chato, 1991).[34]


**Figure 7-32.** Artist's concept of an in-space cryogenic propellant depot.

The present Mars Architecture for the NTR calls for the core stage to be launched first (see figure 7-33), followed by the in-line tanks, and drop tanks. These could loiter in orbit for 4 months. This scenario will cause much of the $H_2$ to boil off unless active cooling is used to prevent it. However, cryogen transfer from a depot offers another opportunity. The depot could be launched just prior to the end of the loiter period and used to top off the tanks just prior to use.

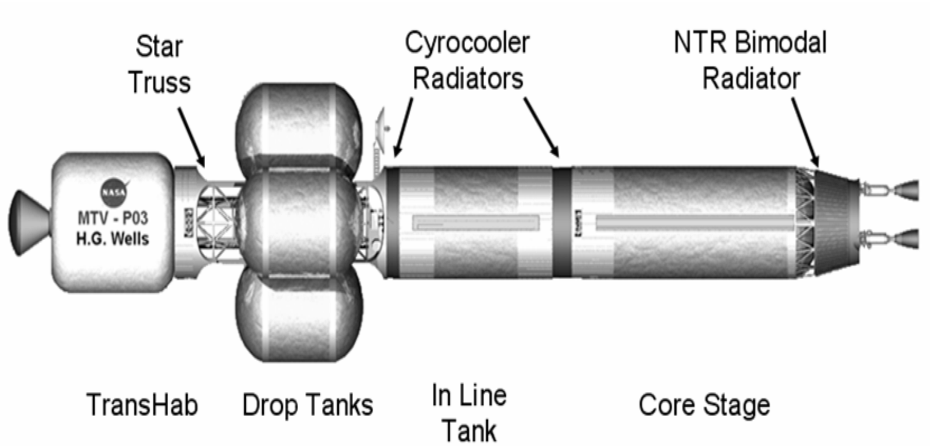



**Figure 7-33.** Artist's concept of the nuclear thermal rocket.

If the Mars Architecture uses the ESAS[1] EDS, it reaches orbit roughly 32% empty, having used significant propellant just to reach a stable circular orbit. If a depot were available, the tanks could be refilled prior to departure, thereby significantly

[34]Chato, D. J., "Cryogenic Transfer Options For Exploration Missions," AIAA 91-354, September 1991.

increasing the EDS payload. Alternately, the size of the EDS could be reduced to only contain the 68% of propellant that would be required for the Earth departure burn. Cost to the EDS would be mainly the weight of the transfer line and its associated valves. Since the tanker is capable of completing its mission prior to the arrival of the crew, it is possible that it could be built to uncrewed commercial standards, thus significantly reducing the amount of margin and redundancy and, consequently, reducing both weight and cost.

Even if no tanker is used, a transfer system that would be between larger and smaller tanks on the vehicle could provide substantial benefit. Cryogenic storage is more efficient in larger tanks, so storing the propellant in the core stage of the NTR or the EDS for a chemical propulsion concept should decrease the total boil-off loss. In this case, the more advanced thermal control (zero boil-off or reduced boil-off) would be used on the larger tank, and cryogens would be transferred into the smaller tanks for top-off prior to use.

Transfer operations will also be required for ISRU of propellants. Although it is conceivable that the in-situ cryogens could be liquefied in the tank that feeds the engine, this would mean carrying the mass of the liquefaction system on the ascent vehicle or, alternatively, developing an interface and separation system between the two. Another issue is that ISRU is planning to produce cryogenic supply of $O_2$ for non-propulsive vehicle elements as well as propulsive elements for the life support system and power reactants. A more practical approach is to have a liquefaction station that can liquefy the propellant in a system that remains on the surface and then transfer it to the ascent vehicle or to the non-propulsive cryogenic tanks.

Advanced architecture studies (Troutman, et al, 2002) [35] (Troutman, et al, 2003) [,36] also show significant benefits to refueling at L1 prior to traveling deeper into space. Tankers from the lunar surface that would be filled with in-situ propellant (or even just $LO_2$, which is the heaviest propellant) rendezvousing with Mars departure stages (MDSs) at L1 could significantly reduce the mass MDS as well.

***Mass gauging***

The objective of low-gravity cryogenic mass gauging technology development is to produce an accurate, robust method of gauging the liquid quantity in a propellant tank in low gravity without the need to settle the propellant with an ancillary propulsive system. The goals of the In-space Cryogenic Propellant Depot Project was to mature low-gravity cryogenic propellant tank mass gauging technology to a TRL 4/6 that would be based on three promising mass gauging concepts: the optical mass gauge (OMG), the RF mass gauge, and the pressure-volume-temperature (PVT) mass gauge. When that project ended after Phase I, the ETDP continued to develop such gauges with similar goals.

Several models and tests were conducted on these mass gauging concepts. Laboratory and system testing on RF gauging proved fruitful. There was excellent agreement between the measured and calculated Eigen mode frequencies, which shows that the computer simulations will be a valuable tool in predicting the low-g performance of an RF mass gauge. The simulation tool can be used to aid in the design and development of an RF mass gauge, and will be used to construct a database of modal frequencies as a function of fill level using various liquid configurations, such as a settled tank or wet wall configurations. From such a database, the goal will then be to develop an RF mass gauging algorithm that can accurately predict the tank fill level using a few select Eigen mode frequency measurements.

Optical mass gauging modeling and testing was also performed. Certain trends have become evident. The tank wall finish seems to play a large role in the overall effectiveness of this method. The integrated reflectance over all angles affects the measurement as a lower wall reflectance provides for more signal loss at the wall and every reflection. This may be overcome by shifting to more absorptive regions of the spectra. However, this will require more input power from the light-emitting diodes (LEDs) to maintain a similar signal-to-noise ratio. Another factor of the wall finish is how Lambertian or diffuse of a reflection the wall provides. The more diffuse the reflection, the more even the sampling of the tank volume. This will provide for less variation in readings as the tank is tilted as the location of the bubble will become less of a factor in the reading. More testing will give answers to these issues.

[35] Troutman P. A, et. al. "Orbital Aggregation and Space Infrastructure Systems," IAC-02-IAA.13.2.06, October 2002.
[36] Troutman P. A, et. al. "Revolutionary Concepts for Human Outer Planet Exploration (HOPE)" Space Technology and Applications Int. Forum 2003 AIP Conference Proceedings Volume 654; no. 1; pp 821-828, February 2003.

The PVT method of liquid quantity gauging in low-g is based on calculations assuming conservation of pressurant gas within the propellant tank and the pressurant supply bottle (such as depicted in figure 7-34). This method is currently used to gauge the remaining amounts of storable propellants on board the space shuttle's orbital maneuvering system and on Earth-orbiting communications satellites. There is interest in applying this method to cryogenic propellant tanks since the method requires minimal additional hardware or instrumentation. A PVT gauging experiment with $LO_2$ has been completed at GRC using a large-scale cryogenic test tank with an attached cold, high-pressure He supply bottle.

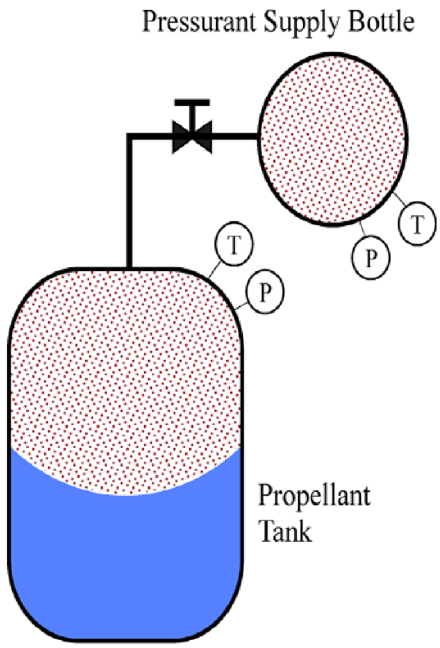



**Figure 7-34.** Basic pressure-volume-temperature gauging hardware configuration.

To use PVT with cryogenic fluids, a non-condensable pressurant gas (He) is required. With cryogens, a significant amount of propellant vapor would be mixed with the pressurant gas in the tank ullage. This condition, along with the high sensitivity of propellant vapor pressure to the temperature of a cryogenic propellant makes the PVT method susceptible to substantially greater measurement uncertainty than is the case with less-volatile propellants. An uncertainty analysis that was applied to example cases of $LO_2$ tanks indicated that the PVT method will be feasible for $LO_2$. (Van Dresar, 2004) [37] A previous experiment with $LN_2$, which has similar properties to $LO_2$, further demonstrated PVT gauging feasibility (Van Dresar, 2006[38]).

**Summary**

The CFM program is advancing mass gauging technology, and the developments that are being made are significant. It appears that the most promising concepts offer significant hope for achieving high accuracy, particularly when coupled with other methods such as inventory monitoring and settled gauging.

**Acknowledgments**

I would like to acknowledge Dr. David Chato and Dr. Neil Van Dresar of the CFM team (NASA GRC) for their support in this technical effort.

[37] Van Dresar, Neil T., "An uncertainty analysis of the PVT gauging method applied to sub-critical cryogenic propellant tanks," *Cryogenics*, 44 (2004) pp. 515-523.
[38] Van Dresar, Neil T., "PVT gauging with liquid nitrogen," *Cryogenics*, 46 (2006) pp. 118-125.

### 7.4.3 Communications and navigation

Advances in cross-cutting technological information systems and automation have the potential to substantially improve the performance, increase the reliability, and reduce the cost of the systems that would be required for future human exploration of Mars. Technology development can decrease delay rates and increase system on-line availability through fault-trend analysis and management. Development can also substantially decrease the operational effect of the need for space-based system maintenance through the application of highly automated system architectures. Technological advancements are needed in automation and robotics, high-rate communications, and data systems, processors, and recorders.

The Mars network that would be required to support human Mars missions would be an extension of the current NASA space networks and the planned lunar network. Mars mission designers do not envision the need for new basic capabilities or increased capacity. The fundamental differences between the lunar and Mars architectures are due to a few factors:

- 1,000-fold increase in maximum distance (400M km vs. 400K km)
- Mars environment including atmosphere
- Choice of a circular aerostationary orbit vs. an elliptical, eccentric orbit
- Spectrum requirements for deep-space (Category B) vs. near-Earth (Category A) mission

The distance drives the design to X-band (vs. S-band for near-Earth use) and increases the difficulty of closing the communication link by 1,000,000-fold in proportion to the square of the distance. The environment of Mars drives the configuration of antennas for descending and ascending vehicles and link budgets through the atmosphere. The aerostationary orbit is highly advantageous for communications purposes, providing continuous coverage of one-third of the planet with a single asset, but sacrifices navigation utility compared to the lunar orbit.

Table 7-9 identifies the key technologies that would be required to extend lunar capabilities to meet martian requirements. This list assumes that the technologies that are identified in the lunar architecture are developed.

**Table 7-9.** Technology Enhancements for the Mars Communications and Navigation

| Capability/ Technology | Key Performance Metric(s) | Lunar Capability | Mars Capability |
|---|---|---|---|
| Integrated Communications/ Navigation Transceiver | Signal data recorder (SDR)/transceiver with integrated comm/nav functions that is easily reprogrammable through software uploads and works for S-, L-, and K-bands. Includes development support for surface transponders. | Initial capability for S-, L- (GPS), and Ka-bands | Modify design for X-band (vs. S-band) |
| Common On-board GN&C System | Enables verification and validation of integrated navigation systems with avionics hardware and other instrumentation within the spacecraft, communications and navigation relays, and lunar surface systems. | Initial commonality based on product line approach | Evolved commonality with tighter integration to lower system cost |
| Autonomous Landing and Hazard Avoidance Technology | Need autonomous landing and hazard avoidance system, including terrain-relative navigation that operates in all lighting conditions, including darkness. Need 100-m accuracy at 3$\sigma$ certainty. Need 0.5-m hazard recognition and avoidance. | EDL capability for descent in vacuum into permanently shadowed crater | Enhanced EDL capability for descent through Mars atmosphere |
| Delay-tolerant Networking (DTN) | Enable surface elements to be interconnected by Internet protocol (IP) across extended distances, overcoming eclipses and link interruptions such as crater and mountain blockage, providing enhanced net throughput with constrained relay capabilities and assets. | DTN capable of supporting 1.5 light-second lunar delays | DTN capable of supporting >44 light-minute Mars delays |
| Advance Antennas | Lightweight, dual-beam HGA to enable communications with both a habitat and a rover that are separated by 250 km via a relay orbiter. | 1-m diameter, S-/Ka-band, fixed antenna with 100° gimbals | 6-m diameter, X-/Ka-band, inflatable or deployable |
| Surface Wireless Network (Mars communication terminal) | Need to support 15 simultaneous users with an aggregate bandwidth of 80 Mbps at extended ranges to the horizon; need to support minimum data rates of 16 kbps and maximum data rates of 20 Mbps; able to convert between WLAN and CNS protocol stacks; able to support time synchronization service to all surface elements. | 5.6-km range 802.16 links capable of 80 Mbps data rate at low user power with mesh network | 10-km range with advances in mobile communication standards at less mass and power |
| Surface Network User Radios | IP-based radios to link humans, robots, habitat, power stations, ISRUs, rovers, and science packages together for loss-of-signal (LOS) applications. | 1-kg EVA suit radio; 320-kg 802.16 base station; separate S-band navigation radio | Reduce mass and power by 50%; integrated navigation and contingency voice channels |
| High-power Dual-band Traveling Wave Tube Amplifiers | Need TWTAs with high output power and high reliability. | 20-W TWTA at S-/Ka-bands, 7-year design life | 500 W at Ka-band, 100 W at X-band; 14-year design life |
| Advanced Optical Communications | Need optical terminals to provide high-bandwidth links for surface-to-surface, surface-to-space, and direct-to-Earth communications. Spacecraft transceiver provides bi-directional communications; transmits up to 100 Mbps. Ground telescope requires photo counting detectors at 1.5 microns; two-way ranging with centimeter-class precision; and atomic clock synchronization. | Not required; recommended demonstration at 400K km | Highly recommended at 400M km |

### 7.4.4 Supportability and maintainability

Among the challenges that are facing human Mars missions will be the development and implementation of robust supportability concepts. In the current context, the term “supportability” has a rather broad scope that includes system maintenance, maintenance-related processes, maintainability design issues, crew support functions including provisioning and overhead tasks, and other issues that fall within the scope of integrated logistics support. Supportability issues will be so important to mission success that they must be an integral part of the operations concept and, in fact, will be a key factor in the development of hardware design requirements.

Resupply capability will be extremely limited, or nonexistent. All resources that would be required to support the mission must be pre-positioned or carried with the crew (with the exception of in-situ generated resources). These missions will also face mass and volume limits that will bound sparing options and strategies. These two constraints highlight the need for, and challenge of, a self-sufficient supportability approach. It will be necessary for the crews of these missions to have at hand all of the resources that are necessary to sustain critical spacecraft systems and support equipment for the duration of their time away from Earth. This capability must be provided while minimizing associated mass and volume requirements. The top-level exploration supportability goals are:

1. *Provide system operational availability that meets requirements*. Operational availability is a factor that describes the amount of time that a system can perform its function as a fraction of total time – including downtime for maintenance.[1]

2. *Enable robust, autonomous maintenance capabilities*. Crews on these missions must be able to perform all of the required system maintenance with minimal direct support from Earth-based experts, and with the physical resources that they carry with them at the beginning of the mission or with resources that were pre-positioned at the destination, when feasible.

3. *Substantially reduce mass and volume resources required to support maintenance and repair*. It will be impractical to carry all of the spares that would be necessary to deal with any potential hardware failure if sparing is implemented at the LRU/ORU level, as is the predominant case for the IS). The mass and volume allocation that this would require would be excessive. A maintenance concept must be implemented that minimizes this need.

4. *Reduce crew time required for overhead tasks*. The ISS crew currently performs a number of tasks that are tedious and time-consuming. Examples include inventory management, trash management, and housekeeping activities such as filter cleaning and biocidal wiping of surfaces. New approaches should be explored that minimize or eliminate the need for crews to perform tasks of these types.

5. *Reduce mass and volume allocation required for crew support*. Concepts must be developed that significantly reduce the mass and volume of crew provisions that would be required to enable and support the function of human crews. As one example, the current approach to providing clothing for ISS crew members will not be practical for long-duration exploration missions. ISS crew members wear an article of clothing for several days then discard it. The mass and volume of clothing supply that would be required by this approach would be excessive for a 3-year mission to Mars. Consequently, it will be necessary to incorporate some type of laundering capability to enable reuse of clothing over a much longer period of time.

These supportability goals will be achieved through the definition and implementation of a consistent supportability concept, appropriate design requirements, and necessary operational and physical capabilities (i.e., training, procedures, facilities, and equipment).

#### *7.4.4.1 Exploration supportability concept*

The cornerstone of the exploration supportability concept is that missions beyond LEO must be independent of support from Earth because of the extended, or nonexistent, supply chain. The crews of these missions must have all of the resources and capabilities that will be necessary to enable them to succeed fully and complete the mission without direct intervention from Earth-based supporting personnel.

This self-reliance will be achieved, in part, by increased emphasis on maintenance by repair rather than replacement. A repair-centered maintenance approach would only be effective, however, when it is strategically coupled with a hardware design that is specifically structured as part of the supportability concept.

**Maintenance**. Robust, autonomous maintenance capabilities will be enabled by implementation of the following concepts and capabilities:

1. *Repair rather than replace*. It is preferred to repair failed hardware items rather than simply remove and replace them. This concept is particularly important for LRUs, ORUs and shop replaceable units (SRUs) that have high failure rates and large masses or volumes. This avoids the use of large quantities of relatively bulky and massive spares.

2. *Replace at the lowest practical hardware level*. The objective is to minimize the mass of spares consumed. An example would be to remove and replace an integrated circuit that has a mass of grams rather than a complete avionics LRU that has a mass of several kilograms.

3. *Comprehensive on-board failure diagnosis*. Failure diagnosis should identify the cause of the failure to the level of maintenance. To the extent possible, these capabilities should be built into the systems. When this is not practical, standalone diagnostic equipment can be used.

4. *Fabricate structural and mechanical replacement parts rather than manifesting unique spares*. Processes are being developed that permit the fabrication of parts from feedstock material that would be carried from Earth or, eventually, produced from in-situ resources. This allows manifesting of an appropriate mass of feedstock material rather than a large selection of unique, prefabricated parts. Manifesting prefabricated parts incurs the risk of carrying excess mass in the form of parts that are never needed and of carrying an insufficient number of parts when there is an unanticipated high-demand rate.

5. *Implement a comprehensive preventive maintenance approach*. An effective preventive maintenance program can help to avoid the occurrence of system failures and loss of availability. In addition, preventive maintenance can delay wearout, thus reducing the need to stock replacement parts. Extensive pre-mission study is required to define realistic schedules for preventive maintenance that allows for in-flight adaptability based on real-time experience.

6. *Enable utilization of common LRUs, SRUs, piece parts, and components across an entire vehicle set.* This will allow spare parts that would be carried on one vehicle to be used on another vehicle, or for system elements from one vehicle to be scavenged for use on another vehicle in critical situations. Interchangeability yields flexibility.

7. *Use reconfigurable hardware.* Using hardware that can be reconfigured to perform different functions as a mission progresses reduces the overall mass of hardware that is carried and minimizes the number of unique spares that are required. Optimally, a single generic part, such as a circuit card, can be easily reconfigured to perform in multiple locations.

**Crew support functions**. Examples of ways in which the crew time that would be required for overhead tasks and the mass for crew support can be reduced include the following:

1. *Launder clothes*. Mass reductions can be realized if clothes are laundered and used multiple times.

2. *Make inventory management transparent to the crew*. Comprehensive and current inventory information is important to crew efficiency. The current manual barcode scanning method that is employed on ISS is cumbersome. A better approach might be the use of radio frequency identification (RFID) tags – active or passive – or use of machine-readable markings. Effective implementation of this technology may require some accommodation in vehicle hardware and system design to allow effective placement of sensors and transmission of RF signals, and to ensure noninterference with other systems.

3. *Recycle waste products*. Mass efficiency will be enhanced if waste products such as packaging and failed hardware can be recycled and reused.

#### *7.4.4.2 Maintainability design themes/requirements*
Design and operational themes are derived from experience with prior and current human space flight programs with consideration of future constraints. Emphasis is on ease of maintenance, standardization and commonality of hardware, and cognizance of issues that would be specific to operations during space flight. The design themes that have emerged to enable the maintenance concept that was described above include:

1. *Design for maintainability, graceful degradation, upgrades, and adaptation*. For a spacecraft that must be maintained entirely by its crew, design for ease of maintenance is crucial. Systems should also be designed in such a way that they can continue to provide reasonable levels of functionality even after some failures have occurred. Systems should be able to accommodate upgrades – either hardware or software – without requiring total redesign. Finally, designers should seek opportunities to design hardware in such a way that it can perform a variety of

functions in different mission phases. This can reduce the total amount of hardware that would be required and simplify its support.

2. *Design and build for maintainability in the operational environment*. The spacecraft structure will be subject to pressure and thermal differentials that can cause dimensional changes. These dimensional changes can affect clearances between parts, thereby making removal and replacement difficult or impossible. The design must consider these changes and how they will affect the ability of a crew to perform maintenance tasks. Working in a weightless environment provides some advantages, but it also requires consideration of how a crew member will maintain stability and be able to apply loads required for tasks. For example, although a specific number of closeout fasteners may be necessary to secure hardware for dynamic phases of flight, far fewer fasteners may be necessary during the much longer, quiescent periods. The number of required fasteners should be minimized whenever possible.

3. *Require commonality and standardization at hardware levels among major architecture elements*. Mission architectures may require multiple elements such as crew transport vehicles, landers, SHABs, and surface vehicles. Every effort should be made to standardize hardware at all levels (LRU, SRU, component) among all architecture elements. This will simplify provisioning of spares, reduce the number of unique tools, and enable substitution between elements. As noted, this applies to hardware at all levels, including avionics circuit card assemblies; electronic components; other assemblies such as pumps, power supplies, and fans; fasteners; connectors; and other piece parts.

4. *All hardware to be maintained should be internal – minimize extravehicular activity*. EVA increases crew risk, is time-consuming, and imposes additional hardware design requirements. To the maximum extent possible, all hardware that may require maintenance should be located inside the vehicle in a pressurized environment to avoid the necessity of performing EVA maintenance operations.

5. *Eliminate avionics line replaceable unit boxes – implement rack-mounted boards*. Eliminating the boxes that are typically associated with avionics LRUs offers potential mass savings and facilitates access to the individual circuit cards for maintenance. Adoption of this approach, however, also necessitates consideration of cooling efficiency, physical isolation of redundant system elements, and the mass of cabling that would be required if avionics are centralized.

6. *Do not combine Imperial and SI [System International] hardware*. All hardware should be designed using a single system of units of measure (SI preferred) to avoid the need for multiple tool sets.

7. *Provide robust diagnostics and post-repair verification*. Efficient maintenance operations require quick, unambiguous fault isolation to the designated repair level. This can be accomplished with built-in-test (BIT) capabilities or with standalone test equipment. Whether via BIT or standalone test equipment, the hardware must be designed to be "testable."

8. *Design systems to operate in a "keep-alive" mode with minimal power*. In situations when power availability has been degraded or when power must be conserved, it is important that other spacecraft systems can remain functional with a minimal power demand. In this condition, the system may not perform its function but retain the capability to do so when additional power is provided. This is similar to interplanetary probes that revert to a "survival mode" during severe radiation events to protect (by power off) vulnerable hardware.

9. *Design systems to enable isolation of faulty components to preclude loss of entire system*. Systems should be designed so that single failures do not cause total loss of function.

10. *Design systems so that pre-maintenance hazard isolation is restricted to the item that is being maintained*. When power, pressurized gas, coolant, or other potentially hazardous resources are isolated from system hardware elements to make them safe for maintenance, isolation should be limited to the smallest possible set of hardware to minimize impacts to overall system availability.

#### *7.4.4.3 Supportability technology development*

Enabling crew autonomy while maintaining future spacecraft systems will require capabilities that go significantly beyond those that are currently available to the crews of ISS. The thrust of technology efforts will be to provide tools and processes that enable crews to repair failed hardware rather than simply rely on remove-and-replace approaches to maintenance. Specific examples of this approach include component-level repair of electronics and on-demand manufacturing of structural and mechanical replacement components.

Component-level repair of electronics is currently practiced by the U.S. Navy in operational environments.[2] This serves as an excellent model for future in-space capabilities. Soldering repairs have been performed on board ISS, and active research is currently under way to fully understand potential effects of reduced gravity on the soldering process.[3] Although the current technology level that would be applied to these operations relies heavily on manual soldering, the introduction of automated soldering rework systems will significantly reduce skill and training requirements for the maintainers.

Since accurate diagnosis and repair of an electronics failure can require significant amounts of time, a useful approach might be to carry a limited selection of spare circuit cards, replace the failed circuit card with a spare, and then repair the failed card when time permits. This would allow quick restoration of system function or redundancy without making the repair process an urgent operation.

This strategy will be leveraged to the greatest degree with implementation of standardization and commonality of piece parts and components across systems. The growing application of programmable logic devices offers the potential to significantly reduce the number and variety of active electronic components and, possibly, the number and variety of distinct circuit cards that would be contained within the avionics systems. This would further minimize sparing requirements, and can greatly simplify fault diagnosis by minimizing the variety of hardware configurations. Further flexibility and robustness can be achieved through use of hardware and systems that can serve multiple functions by reconfiguration of hardware or software.

The introduction of MEMS and nanotechnology into space systems will require consideration of the support strategy that would be best suited for these technologies. Issues that are associated with their inherent reliability, packaging, and handling will determine whether system reliability is achieved by incorporating large numbers of redundant components without replacement of failed items, or whether more traditional replacement of failed components will be appropriate or even possible.

On-demand manufacturing capabilities can provide the means to produce structural and mechanical replacement parts as needed instead of carrying unique spare parts. With this approach it will only be necessary to estimate the mass of the components that will be required rather than specifically the number of those components that will be required. This reduces the risk that would be associated with the possibility that inadequate spares of a particular type will be available. It also reduces the risk that an excess number of specific spares will be carried, thus wasting mass and volume capacity.[4,5]

On-demand manufacturing requires complete geometrical information about the part that is to be fabricated. This information is used to generate a near-net-shape part with one of a variety of additive manufacturing processes and then to make it conform with specified dimensions, tolerances, and surface finishes with subtractive manufacturing processes. Processes for metrology and quality assurance are also required.

Processes are available that can generate parts from a wide range of engineering alloys and nonmetallic materials. Development challenges that must be addressed for space applications include reducing the mass and volume of the processing equipment while expanding its capabilities to produce highly complex components. Process and safety issues that would be associated with operation in reduced-g environments must also be considered. For example, feedstock materials must not pose health risks to humans or safety risks to other systems hardware. Chips that are removed during subtractive processing must be controlled, and cooling of the workpiece during machining will be a challenge.

#### *7.4.4.4 Testing and evolution*

Before the concepts and technologies that were described in the preceding paragraphs are implemented on future missions, they must be tested in appropriate analog environments and during near-term space missions. Experience that has been gained from ISS operations can help to focus future technology development efforts and illuminate design issues. Ongoing maintenance and repair activities will highlight issues that must be addressed to provide increased self-sufficiency. Additionally, ISS may be used as a testbed for development and validation of these technologies and concepts. Recent experience has demonstrated that when resupply opportunities to station are severely constrained, the crew can effectively perform maintenance of systems and payload hardware at levels that were not originally planned.

Similarly, a lunar outpost facility provides an ideal venue in which to test maintenance technologies and operational concepts that may be used during a Mars surface mission phase. These technologies and concepts can be implemented in a non-critical-path approach to better assess their applicability and to identify opportunities for further improvement.

### 7.4.5 Cross-cutting systems critical challenges and technology needs summary

Table 7-10 summarizes the cross-cutting systems challenges, and table 7-11 summarizes the cross-cutting systems testing venues that are applicable to a future Mars mission.

**Table 7-10.** Cross-Cutting Systems Challenges

| |
|---|
| **Current Knowledge or Capability Gaps** |
| • Mars surface composition and knowledge<br>• Mars dust environment |
| **Technology Needs** |
| In-situ Resource Utilization<br>• Production of $O_2$, $H_2O$, and buffer gases from the martian atmosphere<br>• Mars atmosphere acquisition and separation<br>• Mars $H_2O$ acquisition<br>• Carbon dioxide processing<br>• Water processing<br>• System performance and reliability for long periods<br>• Autonomous system operation<br>Cryogenic Fluid Management<br>• Ability to store cryogenic $H_2$, $O_2$, and $CH_4$ for long periods with no minimal losses<br>• Thermal management<br>• Liquid acquisition devices<br>• Cryogenic fluid transfer<br>• Mass gauging |
| **Technology Needs** |
| Information and Communication<br>• High-bandwidth communications<br>• Integrated communication/navigation transceiver<br>• Common on-board GN&C system<br>• Digital-tolerant networking<br>• Advanced antennas<br>• Surface wireless network<br>• Surface network radios<br>• High-power, dual-band TWTA<br>• Advanced optical communications<br>Supportability and Maintainability<br>• Ability to maintain systems in situ<br>• Ability to perform repair at the lowest component level<br>• Ability to manufacture spare parts in situ<br>• Advanced diagnosis and forecasting capabilities |

**Table 7-11.** Cross-Cutting Systems Testing Venues

| Venue | In-Situ Resource Utilization | Cryogenic Fluid Management | Information & Communication | Supportability & Maintainability | |
|---|---|---|---|---|---|
| **A. Earth Surface** | • Technology development<br>• Laboratory demonstration of system performance | • Technology development<br>• Laboratory demonstration of system performance | • Technology development<br>• Laboratory demonstration of system performance | • Technology development<br>• Laboratory demonstration of system performance | • |
| **B. Earth Atmosphere** | • N/A | • N/A | • N/A | • N/A | • |
| **C. Earth Orbit** | • N/A | • Testing of long-term storage of cryogenic fluids in zero-g | • N/A | • Supportability concepts tested and demonstrated at ISS | • |
| **D. Lunar Transit and Orbit** | • N/A | • | • Verification and validation of communication and navigation systems | • N/A | • |
| **E. Lunar Surface** | • Storage, maintenance, and use of locally produced $O_2$ | • Long-term storage of cryogenic fluids in hypo-gravity harsh environments | • Simultaneous system communications<br>• Surface communication and navigation<br>• High bandwidth | • Supportability concepts used<br>• Resupply minimized | • |
| **F. Deep Space Transit** | • N/A | • | • Verification and validation of communication and navigation systems | • N/A | • |
| **G. Mars Orbit** | • N/A | • N/A | • Verification and validation of communication and navigation systems<br>• High bandwidth | • N/A | • |
| **H. Mars Atmosphere** | • N/A | • N/A | • N/A | • N/A | • |
| **I. Mars Surface** | • Production and use of locally produced $O_2$ | • Storage of cryogenic fluids for long periods | • Verification and validation of communication and navigation systems<br>• High bandwidth | • N/A | • |

## *7.5 Risk Mitigation Strategies*

The human exploration of Mars will be a complex undertaking. It is an enterprise that will confirm the potential for humans to leave their home planet and make their way outward into the cosmos. Although just a small step on a cosmic scale, it will be a significant one for humans, because it will require leaving Earth on a long mission with very limited return capability. The commitment to launch is a commitment to several years away from Earth, and there is a very narrow window within which return is possible. This is the most radical difference between Mars exploration and previous lunar explorations. Successful implementation of the human exploration of Mars will require a thorough and in-depth technology development program coupled with a rigorous risk mitigation strategy.

Precursor activities consist of both technology investment and test and validation that are required to produce the technical readiness to develop human missions to Mars. In addition to leveraging the technical advances that would be expected from the lunar human missions and the ongoing robotic Mars Science Program, new, unique precursor activities must be initiated to pave the road to the required capability readiness.

Although no specific timetable has been established for the first human mission to Mars, a notional date of the early 2030s was used as an example date for assessments by the Mars architecture study team. For the initiation of Mars human missions in the early 2030s, mission development will be initiated in the 2020–2030 time period. Thus, precursor activities will need to be conducted in the mid-2010s to early 2020s. This is illustrated in figure 7-35, which notionally depicts a generic series of precursor activities: system studies, technology developments, and validation tests, including possible robotic precursor flights to Mars. This sequence may culminate with large-scale precursors early in the third decade of the 21st Century to validate design approaches.

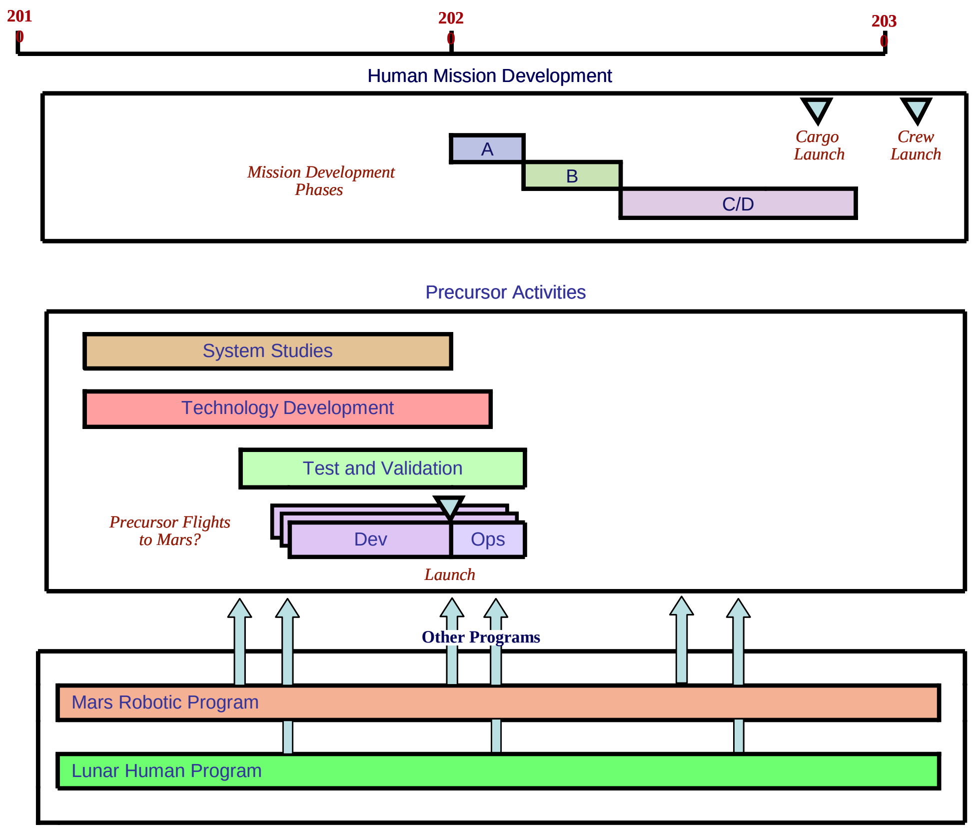



**Figure 7-35.** Notional precursor development schedule.

### 7.5.1 Low-Earth-orbit-based research

The MAWG spent most effort during 2007 on the applicability of lunar and Mars robotic missions and less effort on Earth and ISS testing objectives. Some preliminary efforts were spent on LEO testing. An overview of the types of data that would be needed and testing that is valuable to conduct on ISS is provided in table 7-12.

**Table 7-12.** Mars Forward Testing on International Space Station and Near-Earth Orbit

| |
|---|
| **Human Health and Performance** |
| • Development of zero-g countermeasure protocols<br>• Maximize the number of crews and the duration of their missions at ISS to certify 180- to 200-day zero-g Mars transits<br>• Radiation monitoring and shielding technologies<br>• Advanced medical care techniques<br>• Life support system operation and maintenance |
| **Space Transportation** |
| • Safe and reliable delivery and return of crew<br>• Automated rendezvous and docking<br>• Certification of low-speed entry guidance system and TPS |
| **Cross Cutting and Miscellaneous** |
| • Supportability and maintainability concepts, including repair at low levels<br>• Long-term system performance<br>• Understanding and validation of gravity-sensitive phenomena |

### 7.5.2 Lunar missions

Missions to the moon represent the logical first step in exploration beyond LEO, leading to human missions to Mars. The moon is a natural body with reduced gravity (one-sixth that of Earth) and has a total area roughly equivalent to the continent of Africa. It is relatively close, only a few days away; and is a natural research laboratory orbiting planet Earth.

Lunar missions and preparations are essential prior to accomplishing piloted missions to Mars. The moon is relatively accessible and return can be accomplished at any time, unlike the eventual missions to Mars. This close proximity and enhanced risk posture allow the moon to serve as a vital stepping-stone to the more difficult Mars missions. The topography and environment of the moon can be used to simulate martian conditions and remote operations. As part of this approach, it is important to test and operate actual equipment and systems that are to be used for the Mars missions. The only way to prove that the equipment and systems are truly reliable is to test their functions and operate them over long periods of time in realistic environments.

The moon provides a testing environment of human performance to ensure the safety of the crew. The issue of human performance after long exposure to zero gravity, and the effectiveness of countermeasures to long-term exposure to zero and reduced gravity, must be well understood before sending crews to Mars. The degree of autonomy that would be required in systems and equipment is better assessed after understanding crew adaptability to reduced-gravity environments. Simulations of human stay times can be demonstrated between time spent on the lunar surface and ISS. Crew members adapt in facilities on the moon, performing tasks similar to those required at Mars. These crew members also experience the psychological effects and isolation that are experienced by crews traveling to and from Mars. Operational concepts are developed to make best use of the systems and crew on the planetary surfaces. An overview of the types of data that would be needed and testing that is valuable to conduct on the moon is provided in table 7-13.

### 7.5.3 Mars robotic missions

The need for testing of systems at Mars and making environmental measurements at Mars is expressed in several of the previous sections in this report. Entry, descent, and landing challenges for human flights to Mars, for instance, extend today's robotic program capability of one metric ton of useful mass on the surface to 40 metric tons. In situ resources used at Mars, both for breathing oxygen and for propellants, represent difficult engineering capabilities which have never been tested in a foreign, hostile environment. Much learning about these two challenges, to name a few, can be obtained from single or multiple test flights to Mars itself. Such test flights will lie beyond the fiscal limits of the robotic science program and must therefore be considered within the precursor programs in the Exploration Systems Mission Directorate.

**Table 7-13.** Mars Forward Testing on or Near the Moon

| |
|---|
| **Human Health and Performance** |
| • Development of hypo-gravity countermeasure protocols<br>• Certification of 500-day Mars surface stays by extrapolation from the lunar one-sixth-g environment<br>• Radiation monitoring, protection, and shielding technologies<br>• Advanced medical care techniques<br>• Closed-loop life support system operation and maintenance<br>• Long-term food storage |
| **Space Transportation** |
| • Safe and reliable delivery and return of crew to remote destinations<br>• Heavy-lift launch vehicle performance, reliability, and availability<br>• Automated rendezvous and docking<br>• Certification of high-speed Earth return, including Mars speed, entry guidance system, and TPS<br>• Advanced $LO_2/CH_4$ engine performance<br>• Hazard avoidance and precision landing |
| **Surface Systems** |
| • Extravehicular activity system performance<br>• Surface mobility system performance<br>• Power system performance<br>• Habitation system deployment and operation<br>• Generation and use of locally produced consumables<br>• Dust mitigation techniques<br>• Science strategies and remote field operations<br>• Deep-drilling techniques and operations<br>• Advanced remote laboratory capabilities |
| **Cross Cutting and Miscellaneous** |
| • Supportability and maintainability concepts, including repair at low levels<br>• Long-term system performance in harsh, deep-space environments<br>• Planetary protection protocols<br>• Long-term storage and management of cryogenic fluids<br>• Operational approaches commensurate with Mars mission time delays |

Recent mission studies have shown that a single Ares V launch to Mars can carry as such as 40 t of mass to the planet. This allows a partial, but ample, scale of instrumented demonstrations of aerocapture to orbit and aero-entry to landing. Masses between 8 and 12 t can be landed on the surface with a single Ares V launch. These heavy landers can host a large variety of tests on the surface of Mars, including subsurface drilling to, say, 10 m depth at several locations from a repositioning lander and from quarter-scale ISRU prospecting and productions plants. Many, many measurements of climate, soil, and dust properties and surface chemical composition can be performed. If desired, a single Ares V launch can host a sample return from one or more Mars surface locations. This will allow chemical measurements of the surface element to extreme accuracy by the best instruments on Earth. Such missions to Mars should be planned for no later than the middle of the third decade of the 21$^{st}$ Century as the last phase of precursors that would be needed to enable human travel in the fourth decade of the 21$^{st}$ Century. An overview of the types of data that would be needed and testing that is valuable to conduct on Mars via robotic missions is provided in table 7-14.

### 7.5.4 Precursor activity ties to human mission risk reduction

Ultimately, the precursor activities must be adjusted and optimized to effect a maximal risk reduction for the anticipated human missions to Mars. Each technical discipline must be assigned a risk-reduction maturity curve, which would be directly tied to the risk-reducing effects that are associated with each respective development and test activity. Currently, adequate risk development maturity profiles are not available for analysis and recommendations of the key prioritized precursor activities that should be carried for the respective disciplines. Therefore, this report carries all activities that are recommended at this time by discipline specialists. As maturity curves are developed, recommendations for all of the precursor activities will need to be adjusted and prioritized to make sure that the dollar expenditures improve to maximal extent the reduction of total mission risk.

**Table 7-14.** Mars Forward Testing on Mars via Robotic Missions

| |
|---|
| **Human Health and Performance** |
| • Radiation-shielding technologies<br>• Mars environmental characteristics, including surface radiation, dust, and toxicity |
| **Space Transportation** |
| • Demonstration of human-scalable entry, descent, and landing systems<br>• Precision landing of large payloads (10s of metric tons)<br>• Use of locally produced propellants |
| **Surface Systems** |
| • |
| **Cross Cutting and Miscellaneous** |
| • Mars surface environment, including radiation, dust, toxicity, subsurface, presence of $H_2O$, etc.<br>• Sample return |

An example is provided here for a notional risk-mitigation maturity curve for ascent propulsion from the martian surface to low Mars orbit. This is a crucial, but relatively high-risk, step in our astronauts' return home from the visit to Mars.

Figure 7-36 is a notional illustration of the maturity curves of $CH_4$ fuel-based, pressure-fed and pump-fed engine approaches to Mars ascent. Note that with little experience with this unknown rocketry approach, the probability of success today is quite low, around 0.5. As tests on ground and in flight are conducted across a decade-length development program, the risk falls off and the predicted probability of success rises to above 0.95. A more rapid maturity growth is predicted for pressure-fed systems than the required pump-fed systems. Although the chart at this time does not introduce the specifics of testing and flight use that would be needed for increasing maturity, the lunar program would certainly represent an ideal way to introduce mission experience into the pump-fed technology that would be needed for Mars. It represents a prime opportunity in the lunar program for feed forward to Mars.

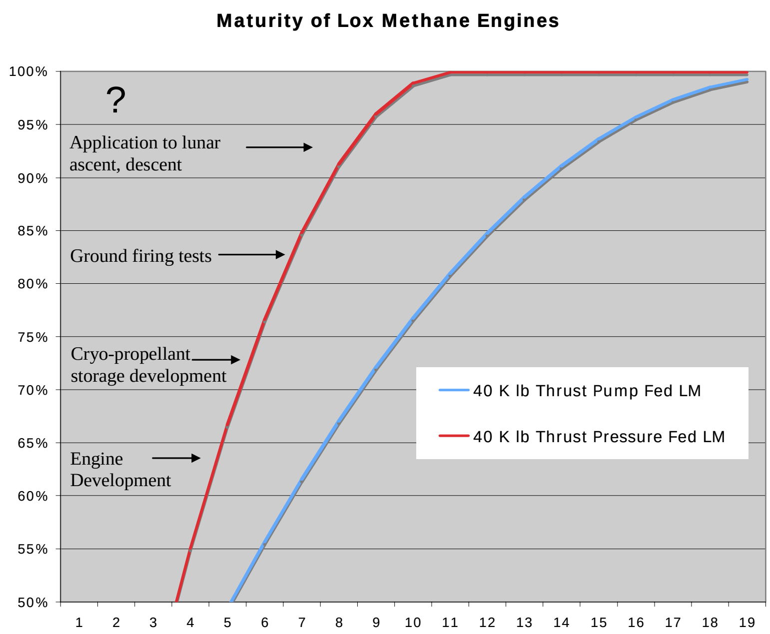



**Figure 7-36.** Example of a technology maturity curve.

To properly use the maturity curves, the project must qualitatively link each test to an equivalent mission value. For example, to continue on the vein of the $LO_2$ $CH_4$ engines, one must first look at the action that would be performed before one could do the analysis that would be required to determine whether performing the active cooling flight test is worthwhile. It is possible to do this by having a risk engineer work with an expert in $LO_2$ engines and have them work out what the qualitative returns on the test will be; from there, an equivalent mission value can be assigned to the active cooling flight test

The example for ascent propulsion that was provided above represents the way in which maturity gains; their impact on risk reduction must eventually be associated with each technical area on which rests the safety of human missions. This type of analysis should be used to guide a determination of the components of an optimized precursor program for martian human flight. If we compute, for instance, the probability of a safe return of astronauts on a mission to Mars today, built without the support of a decade of dedicated precursor activities, that probability would be well below 0.5. A reasonable goal would be for the precursor activities of the decade to drive the maturity curves for each technical discipline to such higher values that the probability of safe astronaut return is above 8.5 or 0.9. At this point, it would be reasonable to begin serious mission development for a human mission. A full-scale mission development program would likely continue for an additional decade, and would likely produce even higher levels of maturity for the technical disciplines. At such point, at the time of human launch, the predicted probability of astronaut safe return to Earth might be greater than 0.95

Another risk-reduction concept that emerged late in this study that would require some precursor activities is the achievement of high degrees of reliability without major increases in mass associated with subsystem- or system-level redundancy and/or sparing. Component-level repair can, in some cases, theoretically achieve sufficiently high levels of reliability while avoiding major mass growth; but the skills, tools, design, and procedures to enable this approach must be developed, tested, and demonstrated.

## *7.6 Technology Contingencies, Descopes, and Fallbacks*

The Mars Design Reference Architecture 5.0 relies on a number of technology developments to provide benefits in performance or cost, or to reduce risk. To enable a resilient and low-risk development program, contingencies, descopes, and fallbacks need to be defined up front to provide tools and options to the program managers and to enable more realistic cost analyses that include contingency scenarios. The following definitions will be used here:

- *Contingency* – an alternate approach that can be used in the event of a programmatic disruption that does not reduce performance or require a fundamental change to the architecture or a technology that was used. An example would be that an alternate star tracker can be used that has similar performance and cost in the event that the primary star tracker source is disrupted or fails to meet cost and schedule allocations.

- *Fallback* – an alternate technology or design that can be used to replace the primary approach in the event that the primary approach proves to be no longer viable from the standpoint of cost, schedule, development risk, in-flight risk, or performance. A fallback typically would entail some negative impact to the program, as referenced in the original specification; e.g., higher cost, schedule delay, higher mass, or reduced performance. An example would be that a surface nuclear power system no longer could become programmatically viable, and the program uses a surface solar power system as a fallback, giving up some benefits in performance or resistance to dust storms.

- *Descope* – a programmatic or technology change that reduces cost, schedule, risk, or performance, but impacts the program such that it no longer meets some of the Level 1 goals or requirements. An example would be to reduce the crew size from six to four to fit within the mass and cost constraints of the program.

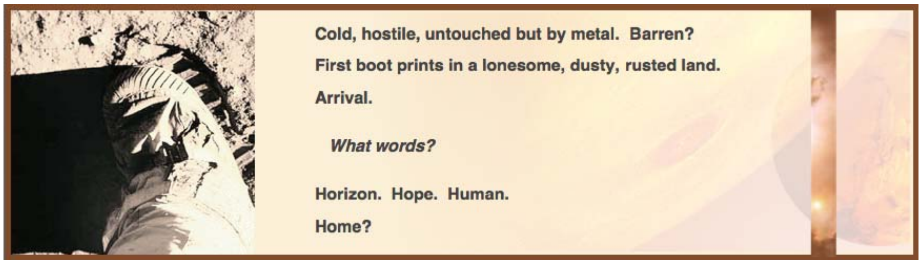
Mars DRA 5.0 is resilient in having a menu of contingencies, fallbacks, and descopes that could be exercised, if needed, to respond to disruptions to the program beyond that which could be covered through cost, schedule, mass, or performance reserves. Table 7-15 provides a list of such options.

**Table 7-15.** Example Contingencies, Fallbacks, and Descope Options

| | Benefits | Impacts | Comments |
|---|---|---|---|
| **Contingencies** | | | |
| $LO_2$/kerosene propulsion for landers and MAV | Lower development cost and risk. Heritage for pump-fed engines. Lower in-flight risk for long-term cryo fuel storage, especially for pre-deploy. Easier thermal design. | Slight lower $I_{sp}$. | |
| Aerobraking used to lower Mars apoapsis for MTV | Much lower propulsive MOI delta-V. | Two to 3 months for aerobraking taken from surface mission. Probably some moderate impacts to spacecraft configuration. | Aerobraking might be possible in <2 months |
| **Fallbacks** | | | |
| Surface power is solar with RPS emergency power | Lower development cost and risk. | Lower overall power available. ISRU for propellant may be problematic. Only have emergency survival power during severe dust storms. | |
| Blunt-body entry vehicle for landers | Probably lower development cost and risk. Better structural load paths. Probably simpler implementation in jettisoning heat shield and backshell. | Lower internal volume. Less configuration flexibility. Possibly less heritage from lunar lander. | Might use inflatable hypercone or other deployable aerobrake |
| Chemical in-space propulsion (TMI, MOI, and TEI) | Lower development cost and risk | Higher launch mass | $LO_2/LH_2$ or $LO_2/LCH_4$ |
| **Descopes** | | | |
| Reduced crew size (e.g., four) | Lower mass and cost. | Reduced crew capability and redundancy | |
| Launch opportunities alternate between cargo and crew | One-half the launch rate. Significantly lower sustaining cost. | Crews sent to Mars only at every other Mars opportunity. Lose benefits of overlapping redundant elements. | Uses pre-deploy architecture |

# 8 OUTREACH: STRATEGIC PUBLIC ENGAGEMENT OPPORTUNITIES

## *8.1 Overview*




The human journey to Mars is a venture in possibility that is sustainable only by the courage, capability, and commitment of the people from around the world over many decades. Each robotic and human-precursor mission in a series will build, step by challenging step, the infrastructure for the first three human missions to the Red Planet. While this dramatic voyage beyond our home planet will stretch technological capabilities, crew health and safety tolerances, time and budget constraints, and the will of hundreds of thousands, we, as humans, strive to go because Mars has much to tell us about the possibility of life beyond Earth, climate changes on both planets, and the limits of survival and sustainability.

The human journey to Mars does not just refer to the brave, select few who will dare to go, or just to the thousands more in the space industry who will, over the course of their dedicated careers, build, launch, and monitor the missions. This voyage belongs to the people of planet Earth, who collectively will not only be witnesses to history, but can increasingly be participants in it. We are already experiencing the beginning of a virtual human presence on Mars, first through the connection between mission team members and their rovers, landers, and orbiters, which have become robotic extensions of themselves, and second between these robotic partners and the public, who have followed their paths, seeing Mars through their "eyes." Advances in Internet and broadband technologies are enabling people not just to access information, but increasingly to have dynamic experiences and make contributions of their own.

From the first rockets to leave the atmosphere of Earth to planetary exploration, the first lunar landings, the space station, and future visits and extended stays visits on the moon and Mars, these are the decades in which human beings are stepping out beyond our home world. Seen from the vantage of human history, it is a civilization's endeavor, *our* civilization's endeavor, as monumental and lasting in the timeline of human history as the pyramids, the Great Wall of China, and other world-heritage sites that remind us of a collective human capacity to achieve what lies at the limits of imagination, skill, and toil. In many senses, it will take a civilization to build this capacity to explore other worlds, and it is how we will be remembered.

Given the large investments that would be required and the risks that would be incurred in pursuit of human missions to Mars, public commitment over several decades will be critical to mission success. Because the public is a primary stakeholder, a traditional outreach program is insufficient. The term "public engagement" is important philosophically; it differs from "outreach" in that it is, by nature, two-directional and implies that it is no longer just about reaching out. Active participation and communication back into the program is extremely important to enable the public to take part authentically in discovery and exploration. While many technical decisions must be made by mission experts who will carefully consider and select among options that best support safe arrivals, that process can be made more transparent. Special opportunities must be created where public input can be included without increasing mission risk, including decisions that would be related to the type of "public engagement payloads" that are of interest, and a public role in their selection (e.g., the most popular request currently is for a microphone so people can hear how Mars sounds). Sharing the adventure with video feeds and interactive sessions with astronauts from the moon and Mars can only happen if public engagement is considered early in mission design and among the principle requirements for mission success (e.g., decisions that are related to increasing bandwidth and reserving mass, power, and space for additional payloads that may not be strictly scientific or life-supporting).

This kind of active public participation at all stages will be a radical change in the way space exploration is conducted, and in some ways, will be as bold a vision as the venture to go beyond Earth. It will take a dedicated public engagement program to create strategic pathways for enabling increasingly sophisticated and informed public participation in the human exploration of Mars. NASA has long been a *civilian* space agency, but it now has the opportunity to become increasingly a *citizen* space agency, a modern transformation worthy of epic exploration in the 21st Century, and of this momentous era in human history.

## 8.2 Guiding Principles for Public Engagement

To create a rich environment for public engagement, five guiding principles related to the human exploration of Mars include: story, participation, inclusion, connectivity, and transparency. These are discussed below.

### 8.2.1 Story

Being able to formulate a story has, since ancient times, been central to human understanding of the world around us and to communicating knowledge from one generation to the next. What is already inherent in sending humans to Mars are the chapters and story arc of exploration, which consist of:

- **Reconnaissance**, the initial context that establishes a characterization of Mars
- **Intensive investigation and sampling**, which represent the rising action, the buildup of challenges on the way to the ultimate experience of touching upon other worlds
- **Human precursor and human exploration,** which are phases that provide the climactic moments within the story of moon and Mars landings
- **Preparation for a later sustained human presence**, which provides the resolution for a crowning achievement…and maybe even the beginning context of the next story in our evolution toward living and working in space!

The epic elements are already in place. For every story, however, structure and information are not enough. Inherent meaning is created as listeners and learners follow a journey outside of what is known, and synthesize that experience in a way that brings greater self and societal understanding. The goal is to have citizens involved in co-creating the story of our age, having the opportunity to provide input on what is meaningful and worth pursuing within acceptable risks and costs, and of benefit to humankind in terms of new knowledge, technological innovations, and human capabilities. It is not about the public being told the story, but rather about the public living within the experience of it, too. Surveys have helped clarify what is essential to the public, and that can inform the journey we will share as a global community. Progressive opportunities will be created in synchronization with each mission phase or “chapter” to ensure deepening citizen knowledge and participation over time.

### 8.2.2 Participation

For a strong public engagement program, good concepts that would be embedded within mission plans are those that would allow the public to follow along and experience the adventure as it happens. The best concepts, however, are those that allow the public to participate actively in the process of discovery. Three methods of ensuring direct public participation are to:

- reserve mass, power, space, etc. for specialized, public-engagement payloads that are driven by public interest and selected with public input
- make small modifications to technologies/systems that are required for human missions that may also result in large public-engagement value (e.g., cameras, infrastructure for “live shows” and data relays, etc.)
- incorporate requirements for bandwidth and other technologies into mission planning, including expansion of DSN, spacecraft relay, and related capabilities that support live programming from the moon and Mars.

Although participation for society-as-a-collective is a goal, the way to achieve this is through specially tailored participative experiences for specific audiences, including educators and students, the media (print, broadcast, and Internet), science centers, museums, planetaria, libraries, community centers, civic and service organizations, special-interest groups, other governmental organizations that have a stake, industry, and universities.

### 8.2.3 Connectivity: “humanity linked to humanity”

While public-engagement activities should be tailored to meet the needs of individual audiences, they should also be designed to encourage partnerships that connect one group to another – i.e., industry to schools, museums to universities, media to civic organizations, and all manner of networks – to provide the richest interactions, the sharing of knowledge, enhanced technical literacy, and a connection to others.

This principle aligns with what is already taking place online, as virtual communities are establishing the means to share knowledge and make contributions, taking advantage of social networks and the “social mind” to solve problems and provide information that would go beyond the limits of any one individual. This kind of capacity can be very advantageous, as the amount of data that is returned from Mars grows beyond the capacity of science teams to analyze in great depth given time and resources. A current example is the way in which the online community creates far more three-dimensional images of the martian surface using rover data than the teams have time to process. Student teams that are linked in online virtual teams have also begun to be trained in analyzing Mars data sets to study topics that are interesting and important, but are farther down on the priority list for experts in the field (e.g., rock abundance counts). In interaction with one another through virtual teams, students have also begun to survey, and present for the consideration of instrument teams, potentially interesting sites for further spectroscopic or other analysis. Providing an outlet and infrastructure for this kind of connectivity will thus remain a key component of connectivity.

### 8.2.4 Inclusion

This journey beyond our home planet belongs to everyone. Whether it is accomplished by one nation, or several nations united together in the effort, the lasting significance of this endeavor is of global scope. Greater opportunities for encouraging those who have been traditionally underrepresented in science and engineering fields should remain key in all public-engagement programming to ensure that NASA can attract and retain human talent and ingenuity from across the nation, among citizens of all backgrounds. Listening to the viewpoints and ways of knowing of indigenous peoples worldwide can also help guide a thoughtful approach to the design of exploration that supports survival and sustainability on this world, as well as on others. The moon, and to a lesser extent Mars, are considered sacred bodies to some cultures, so this kind of dialog is vital to avoid unnecessary misunderstandings and to design a program that is respectful of all traditions. Regardless of specific cultural traditions, the moon is iconic to all of humanity and, together with the sun, is one of the first celestial bodies pointed out to young children the world over. Given that a personal relationship with the moon is part of everyone’s nighttime experience, people care about the moon and are beginning to care about Mars as familiarity with it, and its potential as a human destination, increases. Those sentiments cannot be ignored for the moon and Mars as that might be considered, from the public perspective, part of the global commons, no matter which nations first visit them.

Language that is used in mission planning, preparation, and communications can either welcome participation or be a barrier to it. Thus, in reference to missions and exploration, it is desireable to defer from using terminology such as “colonization,” which has specific, negative connotations among many peoples, in favor of establishing “communities,” “outposts,” and “settlements.”

### 8.2.5 Transparency

Space exploration does not occur without risk and without any impact. Part of the public’s interest will no doubt center on issues such as planetary protection, the use of nuclear power, costs, and risks. It will be important to continue to promote the philosophy that is outlined in the current Mars Exploration Risk Communication Plan and cross-correlated with the Mars Exploration Program Public Engagement Plan:

- Stay responsibly open, candid, and honest
- Share information freely and as soon as possible
- Use plain language
- Continue being transparent, especially when sharing information about risks, benefits, and programmatic changes and failures
- Actively seek as many perspectives as possible
- Be sensitive to cultural differences
- Listen to colleagues, critics, and supporters
- Be clear where NASA can/is willing to take input
- Based on input, be open to modifications or new options

Likely questions where public input would continue to be valuable include:

- Why should humans explore Mars? What themes does the public care about (e.g., science themes, economic potential)? What would it take to have a human mission to Mars worth doing (above robotic missions only)?

- How much cost is acceptable? Over what timeframe? As what percentage of the NASA budget? As what percentage of the U.S. budget?
- What kind of timeframe is acceptable? (Next decade? Next century? Would there be a loss of interest over a long timeframe, particularly in response to deferments or setbacks?)
- What are measures of success from a public perspective? How much failure is acceptable? How would the public respond to failures of precursor missions? How would the public react to losses of human life on the journey to and from, or during time spent on, the moon and Mars?
- What should the role of industry and commerce be in a space economy? What is desired as a return on investment in terms of spin-offs and other benefits that might improve life on Earth?
- What is the public sentiment toward power-source options and planetary-protection topics? What questions do members of the public have?

## *8.3 Public Engagement Strategy*

### 8.3.1 Central organizing theme

While beyond the scope of this study, a detailed plan for public engagement must be created that is based on formative analyses of the ways in which the national and global public would like to participate in the adventure. Without this public input, it is premature to select definitively an action plan for public engagement. At the same time, what likely binds Earth, moon, and Mars exploration is a **central organizing theme** that is both immediate and compelling in human terms: **survival and sustainability** on any of these worlds. Already, that theme is likely to dominate public life in considering conditions on our home planet over the next decades. Additionally, in an informal public-opinion survey that neutrally asked “Why should we explore Mars?” of Internet visitors, one-third of respondents (all ages, genders, education levels) replied with fears for conditions on Earth and hopes of a second home. Attention to survival and sustainability, therefore, responds to public interests and not only enhances the possibilities for continued human and robotic exploration of Mars, but also provides an important opportunity to engage the public in improving life and well-being here on Earth.

### 8.3.2 Public engagement outcome

By the time humans set foot on Mars, the desired **public engagement outcome** is that **citizen scientists would gain new knowledge and use technology for sustainable living and personal exploration as members of a spacefaring society.** Beyond space exploration, this outcome is important to a vibrant and healthy $21^{st}$-Century economy. Both science and engineering enrollment is suffering in the United States, and ensuring greater technical literacy for all members of society is increasingly important for informed decision-making in all of the areas in which science and technology relates to our lives, whether in health, transportation, energy, commerce, or space. A nation of discoverers and inventors is the ultimate goal for national well-being, and the excitement of space exploration can be an initial hook for that capability to grow so that it becomes an intrinsic and inextricable part of our society and culture.

### 8.3.3 Public engagement topical strands

In fact, to say that the human exploration of Mars is a civilization endeavor means that there can no longer be a strict separation between the majority of citizens and the “rocket scientists” who specialize in space careers. At the same time, public input cannot be random and whimsical; the emphasis has to be on informed public participation in a manner that complements and enhances NASA goals. To enable citizens to gain the expertise that would necessary to become full-fledged members of an increasingly spacefaring society, a progressive pathway for participative learning experiences should be created. For cohesion, these progressive learning experiences, which would be designed to build knowledgeable public participation, will center on **three major strands of public engagement related to survival and sustainability on the Earth, moon, and Mars: namely, science, technology, and society.** These three strands are directly correlated with the desired public engagement outcome: citizen scientists who are gaining new knowledge (science) and using technology for sustainable living and personal exploration (technology) as members of a spacefaring society (society).

#### *8.3.3.1 Science: acquiring place-based knowledge through imaging and data analysis*

Whether on Earth, the moon, or Mars, understanding one’s environment is key to survival and sustainability. Imaging and data analysis – through laboratory analysis, field studies, or remote sensing – can reveal important information about life, climate, geology, and overall habitability. However, data analysis can often be intimidating and abstract to the general public. Research studies show that place-based methodologies increase an interest in learning, because an experiential, multidisciplinary, hands-on approach focuses first on the local environment, which is most relevant to life, home, and community. With this connection, meaningful relationships and learning experiences can be more easily extended to other bioregions on and beyond Earth. (In addition, including Earth science is important because it, and not moon or Mars science, is in the National Science Education Standards.)

#### *8.3.3.2 Technology: developing a human-robotic partnership through innovations*

Human communities have always relied on innovations and inventions for survival and sustainability, and have now gone far beyond the creation of mere tools. Robots that are tended by humans are currently providing a virtual experience of other worlds, and are extensions of the humans that tend them. Building a habitat and the means of survival on other worlds relates very closely to what is important in people's everyday lives: food and drink, shelter, tools, health, transportation, communication and companionship, resources, and energy. While the more technical names can be substituted for the more prosaic (e.g., Mars SHAB =shelter; CEV/MAV=transportation; DSN=communication), these are essential human needs. The ability to sign on students, industry, academia, and the wider public into designing, building, and using technologies that are related to survival and sustainability will directly link members of the public to understanding the scope of the challenges, even while they are increasingly invested in providing ideas and solutions for use on Earth or elsewhere.

#### *8.3.3.3 Society: building a shared human experience*

When we go to Mars, we will not be sending automatons that collect and analyze data by rote and return with only information. We will also be sending people who will take their hopes, fears, humor, frustrations, imagination, courage, language, and customs with them. In addition to individual characteristics, perspectives, and emotions, human culture is a product of place and time, of influences such as topography, soundscape, what we eat, how we move, what our background experiences have been, and of how we influence, and are influenced by, the popular culture. On the moon and Mars, all of these factors will come into play as astronauts relate what it is like to be on another world. As a recent public survey reflected, "Robots can blaze trails through space, but only people can tell us how space feels." The public is directly involved in the evolution of what it means to be not just increasingly global citizens, but citizens of a spacefaring society. Being part of this society means documenting and preserving the experience in ways that connect scientific discoveries and technological innovations with history, geography, weather patterns, local resources, and artistic and literary expression. While folklore seems a quaint and antiquated term, it is about who we are at any given time, and its study involves skills that are very familiar to scientists: observing, recording, finding characteristic patterns, mapping, and deriving meaning. An integration of the hard sciences with the arts, humanities, and social sciences can open the door for those who would not otherwise be as exposed to the excitement of scientific discovery and technological innovation. Its inclusion is central to the question of how we, as a society, will choose to balance (in a similar manner to technical trade spaces) environmental, economic, and social equity in considering space exploration among national and world priorities…and even respond to the eventual question that arises from our science-fiction culture: "red Mars, green Mars, or blue Mars?"

### 8.3.4 Public engagement pathways

Public engagement activities in each of the three topical strands of science, technology, and society will deepen and expand in concert with missions that are pursued during the technical phases of Mars exploration: reconnaissance, intensive investigation, sampling, human precursors (moon), human exploration (Mars), and a sustained human presence. **Public engagement pathways will progressively develop citizen capabilities in each of the three topical strands** to the extent that participation in the human exploration of Mars is no longer remote, and is instead able to be embedded within the context of people's communities and lives. Table 8-1 provides a sample of public engagement pathways.

#### *8.3.4.1 Pathway 1 – Science: acquiring place-based knowledge through imaging*

By the time humans reach Mars, the desired public engagement outcome for the science strand is that citizens will be gaining new knowledge according to their interests, and through opportunities to participate in exploration and discovery. To achieve thus, a sample pathway of progressively sophisticated public-engagement opportunities is provided in the following subsections.

**Table 8-1.** Sample Public Engagement Pathways

| | Central Organizing Theme: Survival & Sustainability for the Earth, Moon, and Mars | | | | | | |
|---|---|---|---|---|---|---|---|
| | Pathways to a Public Engagement Outcome By Mars Exploration Phase | | | | | | Public Engagement Outcome |
| | Reconnaissance | Intensive Investigation | Sampling | Human Precursor (Moon) | Human Exploration (Mars) | Sustained Human Presence | |
| | *Heritage Public Engagement* | *Expanded Heritage* | *Potential New Efforts* | | | | |
| SCIENCE: Acquiring Place-based Knowledge through Imaging & Data Analysis | Earth/Mars Comparisons Rock Around World<br>Mars Student Imaging & Analysis | Public "Earthonaut" Investigations in Local Environments<br>Earth, Moon, & Mars Data-mining Credentialing | Earthonaut Local Soil Comparisons to Mars Soil Simulants "Returned"<br>Earth, Moon, & Mars Data-mining | Citizen-Directed Investigations on the Moon | Citizen-Directed Investigations on Mars | | Citizen Scientists gaining new knowledge... |
| TECHNOLOGY: Developing a Human-Robotic Partnership through Inventions & Innovations | FIRST, LEGO, BEST, Weather Balloons | Public Design Challenges: Survivor Rover<br>Habitat Technologies for Humanity | Public Design Challenges: Human Helper Robots<br>"Invention Convention" Design Challenges for Earth Habitats (food, transpo, energy etc.) | Citizen-Selected Public Engagement Payloads for the Moon<br>Bioplex Dome Testbed for Citizen/Industry Research related to Outpost Living on the Moon & Mars | Citizen-Selected Public Engagement Payloads for Mars<br>Habitat Technologies Integrated on the Earth, Moon, & Mars | | ... and utilizing technology for sustainable living & personal exploration... |
| SOCIETY: Building a Shared Human Experience through the Arts, Humanities, & Social Sciences | Citizen Think Tanks<br>Imagine Mars Project IMAX/Nova programs<br>Sundial Messages, Send Name to Mars | Citizen Councils<br>Virtual Field Trips & Simulations for Outpost Living<br>Time Capsules for Mars Created | Citizen Council Input for Sample Returns<br>Reality Earth Earth Classroom | Citizen Council Input Integrated for Moon<br>Reality Moon Lunar Classroom | Citizen Council Input Integrated for Mars<br>Reality Mars Martian Classroom<br>Time Capsules for Mars Opened | | ...as part of a Space-faring Society. |

**Science near-term: reconnaissance through intensive investigation**

Existing heritage includes the Mars Student Imaging Project, which has had nearly 20,000 students (largely middle school) involved in targeting a camera on the Mars Odyssey orbiter and analyzing the resulting image. The Mars Exploration Student Data Teams involve students (largely high school) in virtual teams that analyze more sophisticated data sets and make recommendations to the science teams. Both of these ongoing programs have a common infrastructure that includes data from multiple missions and provide a solid base for involving students in a way that provides increasingly sophisticated levels of participation over time. For the general public, some orbital camera teams allow people to make image-acquisition suggestions. All of these efforts have room to grow, and there is an opportunity to increase citizen knowledge and capability with data-mining opportunities that are related to remote sensing of the Earth, moon, and Mars. By applying techniques that are familiar in a gaming environment (e.g., progressive levels of stature built through experience and skill), for instance, a fun and educational way of credentialing the public to identify interesting data that are worthy of further study could be created. That capability is increasingly important in an environment where the amount of data is so extensive that applying more talent and a “world mind” would be useful contributions to NASA goals. To move beyond this baseline of public involvement in robotic exploration, a program could be created enabling students and the public to mimic human exploration by becoming “Earthonauts.” Earthonauts would, with the participation of local university and other experts, analyze their local environments through place-based learning methods. This kind of a program would set up knowledge that will foster both Earth-based awareness and later comparisons to the moon and Mars. It would also put students and the public more in touch with role models and industry and university recruiting efforts that would help to build a workforce for space exploration.

**Science mid-term: sampling through human precursors**

With greater numbers of the public “qualified” through gaming techniques to analyze data, a more formalized program for data-mining could be put in place, with cooperation occurring among virtual teams across the country and mentoring opportunities between learners and those with greater expertise. With a lot more eyes on the data, the public would be directly engaged in discovery, much as has sporadically occurred (e.g., a member of the public identified a dust devil during the Mars Pathfinder mission), and could also be more actively involved in data processing (e.g., as has occurred with public creating three-dimensional images of Mars data). As these technical and critical-thinking skills are widely applicable to a number of high-tech industries, it also contributes to the development of a skilled workforce. Earthonaut experience could be brought to bear in comparing local soil samples that would be collected around the country with samples of Mars soil simulant “returned” through the mail by companies that crush analog volcanic rocks for NASA use.

**Science long-term: human exploration through sustained human presence**
With the creation of a citizen base that is knowledgeable and qualified to participate in ongoing NASA research, citizen-directed investigations could be conducted on the moon as a test for later investigations on Mars. Given that astronauts will be on the surface of Mars for some 500 days, there will be ample opportunity and even a need to develop tasks that alleviate boredom and create connections back to Earth to mitigate the psychological strain of being physically separated from Earth and humankind. As mission success requirements are met, the public could become more involved in suggesting rocks to pick up and analyze, safe but "just-out-of-sight" landscapes to explore, sunsets and views of the martian moons to collect, and other studies of general interest. Live broadcast sessions that allow the public to be on Mars virtually, "alongside" the astronauts as they conduct their work, would provide a direct sense of what it is like to explore another world.

#### *8.3.4.2 Pathway 2 – Technology: developing a human-robotic partnership through innovations*
By the time humans reach Mars, the desired public engagement outcome for the technology strand is that citizens will be using technology for sustainable living and personal exploration on the Earth, moon, and Mars. To achieve this, a sample pathway of progressively sophisticated public engagement opportunities is provided in the following subsections.

**Technology near-term: reconnaissance through intensive investigation**
Currently, NASA participates in student robotics competitions (e.g., FIRST, BEST), and some missions have enabled student experiments on weather balloons and other platforms. While these efforts have an impact, they require a great deal of instructional aid and reach a small, albeit important, group. To involve more people at different levels of engineering skill and across a more diverse range of fields that are relevant to human exploration, a "Habitat Technologies for Humanity" program would enable citizens to learn the skills that would be necessary to build low-cost technologies (e.g., solar-powered $H_2O$ purifiers) that could be delivered to communities facing what might be considered "outpost-like conditions." While providing public services to the global community, the program would provide students and citizens with a fundamental baseline experience in engineering that could then be expanded upon. As the heritage continued to expand from the Mars Exploration Program Public Engagement Plan, one design challenge could create a "survivor rover" experience wherein individuals or university and industry teams could compete to create micro-rovers that survived after being dropped from a height (e.g., by crane, helicopter, etc.). In addition to media prospects, the challenge would also help the public understand the difficult risks and challenges of landing payloads safely, thus helping to manage public expectations.

**Technology mid-term: Sampling through human precursors**
In this period, more complex engineering opportunities could be enhanced through "invention convention" design challenges for Earth habitats and outpost living that build on the heritage of a "Habitat Technologies for Humanity" program. The design challenges would focus on innovations for better providing necessities, including food and drink, shelter, tools, health, transportation, communications, resources, and power. As an incentive, the winning inventions would be considered for potential addition to human flights. These inventions would range from simple to complex. To take the next step beyond the survivor rover design challenge for micro-rovers, students, citizens, and industry teams could design rovers to conduct tasks that would assist astronauts in carrying out their daily tasks. These human-helper robots could be tested in Earth/Mars analog sites, with the competitions that are similar in organization to the DARPA Grand Challenge Race Challenge, in which teams design long-range autonomous vehicles.

**Technology long-term: human exploration through sustained human presence**
Once the public has developed more extensive engineering experience, further research experiences could be created for a bioplex dome testbed that would replicate, at least in proximity, the setting for exploration on the moon and Mars. While the public at large will not be able to leave the surface of Earth for a long time, this site could be a draw for real-time or virtual visitors and media alike. While astronauts are on the moon, parallel experiments could be taking place in the bioplex dome testbed, especially as citizen-selected public engagement payloads for the moon would be in place there. Lessons that would be learned from this experience would then be used as a baseline for the further development of habitat technologies and citizen-selected public engagement payloads for Mars.

#### *8.3.4.3 Pathway 3 – Society: building a shared human experience*
By the time humans reach Mars, the desired public engagement outcome for the society strand is that citizens will feel that they are truly participating members of a spacefaring society. To achieve this, a sample pathway of progressively sophisticated public-engagement opportunities is provided in the following subsections.

**Society near-term: reconnaissance through intensive investigation**
One of the most popular programs for the current Mars Exploration Program Public Engagement Plan is "Imagine Mars," in which student teams take a look at their home communities, evaluate what is effective and what is not, and then design a community on Mars, learning scientific and engineering concepts along the way. This program has been particularly effective in underserved communities, including hurricane-affected students in Louisiana and in Department of Housing and Urban Development (HUD)-sponsored after-school programs. This program could initially be expanded to more areas around the country, and later connected to virtual field trips to Earth/Mars analog sites on our planet where simulations of "outpost living" would be conducted based their ideas and designs. That would also enable astronauts to practice and test, from a technical perspective, real-time programming for eventual live events from the moon and Mars. To ensure public input, citizen think tanks would initially be created to allow people to access knowledge about Mars and the plans to send humans as well as to establish initial dialog. Those think tanks would be elevated into citizen councils, which would be made up of members who have passed various "knowledge grade levels" to provide input on technical topics that are related to the moon and Mars and societal topics that are related to extending a human presence to other worlds. Communities could record their ideas and create real-time or virtual time capsules to be opened when humans first land on Mars.

**Society mid-term: sampling through human precursors**
During this period, it will be important to enable citizen councils to provide informed input that would be related to the societal aspects of sample returns, as well as ways of disseminating accurate public information. From a social studies perspective, citizen councils nationally, and perhaps globally, could be connected in open dialog in the interest of sharing cultural perspectives. One potential outcome would be the creation of a record of humanity that would eventually be placed on Mars. This digitized payload would give citizens worldwide an opportunity to decide what is important as symbols of humanity, much as a smaller committee did for Voyager. Astronauts who are selected for moon missions could connect with the citizen councils through "town hall meetings" from the moon. Other live programming would include "Reality Moon" and "Lunar Classroom," both of which would be a way of allowing people to go along with the astronauts as they conduct their work on the surface and simultaneously learn as new discoveries are made. This co-collaboration would be further enhanced through citizen-selected, NASA-vetted public engagement payloads for the moon in which the public has a direct interest.

**Society long-term: human exploration through sustained human presence**
When humans reach Mars, it will be the culmination of many years of focused effort with more public participation than ever before. In this historic moment of a shared journey fulfilled, people throughout the world will be connected in opening and sharing the contents of their earlier time capsules, which would provide a reminder of how far humans have come historically and otherwise since the vision was first pursued. Following the experiences on the moon, astronauts would connect with citizens back on Earth through Reality Mars, Mars Classroom, town hall meetings, and other formats that would bring a real experience of the dream to life.

## 8.4 Schedule and Benchmarks

By following pathways in each of the three areas of science, technology, and society, the intended outcome is to achieve a state of participation in which citizen scientists would gain new knowledge by using technologies for sustainable living and personal discovery as citizens of a spacefaring society. A critique of this planned course of action for public engagement might be that the goal is too ambitious. Yet reaching beyond what is currently possible helps draw us inexorably toward higher achievements than might otherwise be possible. Setting the bar high and then seeking to reach for it is, after all, what will eventually take humankind to the moon and Mars. The idea of building toward a spacefaring society of knowledgeable and participative citizen scientists and citizen engineers is very much in alignment with the great contemporary Mars mission engineering spirit of "never say no, say how."

A detailed Public Engagement Plan for the Human Exploration of Mars will need to be created, but several overarching benchmarks will help to support the achievement of an ambitious public engagement outcome commensurate with landing humans on Mars. Table 8-2 depicts a potential benchmark.

**Table 8-2** Outreach Schedule Benchmarks

| SCHEDULE | BENCHMARKS FOR CREATING AUTHENTIC CITIZEN CAPABILITY AND A ROBUST TECHNICAL INFRASTRUCTURE THAT ENABLES PUBLIC PARTICIPATION | | |
|---|---|---|---|
| | Place-based Knowledge (Science) | Human-robotic Partnership (Technology) | Shared Experiences (Society) |
| 2008–2009 | • Conduct intensive research on public opinions and interests, potential partners, measurable outcomes, and evaluation methods<br>• Define and characterize the current lunar/Mars science and engineering communities and future needs<br>• Create a comprehensive plan that is based on that public engagement and pipeline research<br>• Create an initial set of online public information materials<br>• Invest in any initial strategic leveraging opportunities with existing NASA, Mars Public Engagement, or other efforts | | |
| 2010–2014 | Develop user-friendly software and other tools, and demonstrate that they enable students and citizens to work with and share data | Develop user-friendly software and other tools and demonstrate that they enable students and citizens to work with and share data<br><br>Bandwidth requirements in mission planning for high-resolution interactive public participation | Proof of Concept: Citizen think tanks demonstrate effective dialog and input<br><br>Earth virtual field trips are selected and "live programming" is tested<br><br>Time capsules are created nationwide |
| 2015–2019 | Demonstrated success in citizens conducting and contributing research on their home environments/ habitats<br><br>Examples of "gaming-certified" citizens mining data and making discoveries with moon and Mars remote-sensing data | Citizen-built "human-helper" rovers working here on Earth, and at least one public "invention convention" concept selected for lunar outposts<br><br>Public engagement payloads in announcement of opportunity (AO) for human missions to Mars<br><br>Bandwidth requirements met in expansion of DSN, spacecraft relay, and other capabilities | Reality Earth/Earth Classroom Programming tested, with astronauts practicing live programs from remote Mars analog sites<br><br>Citizen councils demonstrate ability to provide informed input on technical topics, including sample returns, and on societal topics that are related to extending a human presence to other worlds |
| 2020–2024 | Proof of concept: First citizen science results from the moon | Citizen/industry/academic payloads incorporated in lunar missions and public-engagement payloads in AO for human missions to Mars<br><br>New technologies tested in Earth-based biodome as part of design challenges | Reality Moon and Lunar Classroom programming with astronaut "correspondents" demonstrated<br><br>Input from citizen councils incorporated into lunar missions and astronauts connect with citizen councils through "town hall meetings" from the moon |
| 2025–2029 | Expanded citizen science investigations on the moon and concept Validation for Mars | Citizen/industry/academic payloads in use on the moon | Reality Moon and Lunar Classroom enable interactive citizen-science investigations<br><br>Citizen councils consider societal topics that are related to extending a human presence specifically to Mars |
| 2030–2034 | Citizen science investigations selected for Mars | Citizen/industry/academic payloads selected for Mars<br><br>Additional DSN, spacecraft relay, and related capabilities in place for live programming from Mars | Astronauts selected for Mars missions practice live programming from remote Mars analog locations on Earth<br><br>Input from citizen councils incorporated into Mars missions |
| 2035–2039 | Citizen science investigations on Mars<br><br>Leave-behind citizen science payloads created (e.g., astronaut-enabled work or leave-behind payloads) | Technologies created through public/industry/academic partnerships in use on Mars | Reality Mars and Mars Classroom Programming begun with astronaut "correspondents" who also connect with citizen councils through "town hall meetings" from Mars<br><br>Mars time capsules opened when humans land on Mars |
| 2040– | Public Engagement Outcome Achieved | | |
| | Citizen scientists gaining new knowledge... | ...and using technology for sustainable living and personal exploration... | ...as members of a spacefaring society. |

## 8.5 Recommendations

Recommendations are focused on actions that must be taken at high levels within the agency and are beyond the purview and authority of any public-engagement staff who would ultimately implement the components of a Public Engagement Plan for the Human Exploration of Mars.

### 8.5.1 Programmatic organization

To succeed, a public engagement effort must be highly organized and effective at keeping contributors from many different parties, both inside NASA and beyond, focused on the same goals and outcomes. The current Mars Exploration Program has successfully demonstrated a programmatic approach, where core efforts such as the Mars Student Imaging Project and Mars Museum Visualization Alliance continue to build a participant base, both in numbers and capabilities of participants, over the life of all of robotic missions to Mars. Current mission content is brought into these ongoing programs, but the infrastructure is common, so there is no “reinventing the wheel” from one mission to the next, resulting in cost-savings and multiple leveraging opportunities. The longevity of the program enables the development of lasting relationships with partner networks that increase in depth, sophistication, and reach over time. That concept can be usefully expanded for the human exploration of Mars. Essentially, what is needed is a systems approach to public engagement that not only brings into synergy the efforts of each of the directorates and external partners, but also creates a joined moon-Mars program for public participation. The two destinations are linked not only in content, but also in consequence. Success, failure, risk, and cost for one will impact the other. Management should be located within a single directorate to ensure that the contributions, while distributed across the agency and beyond, are strategic, and not going in a multitude of directions. That said, the intent is not to create a monolithic effort, but rather a program where contributions from many sources can be streamlined toward a single set of clearly articulated goals and evaluated for effectiveness in meeting them. However individually well-run, a myriad of disconnected public engagement activities, especially those that are created anew, mission-by-mission, ends up serving few and does not allow a buildup of long-lasting capability or truly national reach.

### 8.5.2 Changes to the announcement of opportunity process for public engagement

A systems approach for public engagement would also mean that necessary changes be made to the current AO process. The original intent for ensuring the inclusion of education and public outreach in competitive proposals was sound: to involve mission scientists and engineers in giving back to the taxpaying public, who essentially invests a portion of national assets in space exploration. However, the current process introduces inefficiencies and, thus, prevents the largest return on that investment. Mission, instrument, and other proposals are currently selected on technical merit, and the accompanying education and public engagement efforts vary widely in reach and effectiveness. Often, the best public engagement ideas are found in proposals that did not win for technical reasons. Additionally, when there is only one winner among tens of proposals, much effort in developing plans with potential external partners who are willing to bring in-kind and other assets to the table is lost. In considering changes to the AO process, it remains important to ensure that a percentage of funding is reserved for public engagement purposes so that the public can be involved in, and benefit from, the research and ventures that it sponsors. In fact, returning to the 1% to 2% level of funding for public engagement only seems appropriate given the amount of national investment that the public would be asked to make in sending humans to Mars. That level of funding needs to begin at least by the year 2010, after a plan is put in place during the 2008–2009 timeframe. An uninterrupted funding stream to continue the heritage and momentum to feed forward is critical. Regardless of the amount of funding, however, a better approach to allocation would be to direct a proportion of public engagement funding to an infrastructure for ongoing, proven programs, and then to conduct separate, competitive selections (of various sizes and outcomes) for public engagement, specifically geared to achieve goals set out in an overarching plan. That would allow infusion of new ideas and partners, yet keep all participants marching toward the same collective outcomes. Following the principles of transparency, inclusion, and participation, a role could also be designed in the new process for the public to provide input on which efforts would be most meaningful to them.

### 8.5.3 Public engagement payloads for high-bandwidth programming

Opportunities for citizens and students to work with cameras and other instruments are already in place (e.g., student use of THEMIS on Odyssey, the “People’s Camera Program” through HiRISE on Mars Reconnaissance Orbiter), and small payloads that were designed for public engagement value have also been included on missions (e.g., “Send Your Name to Mars” DVDs). For future robotic and human missions to Mars, this trend can be enhanced by making public engagement payloads part of design requirements for mission success, and by ensuring that any dual-use opportunities are leveraged (e.g., the way in which MER calibration targets double as sundials for students or the video camera on MSL will be leveraged for public engagement purposes). By providing the public a role in predetermining which potential payloads are of greatest interest (e.g., a microphone), design requirements could specify the parameters of the payloads prior to competition. More industry partners beyond those that are typically involved in space missions could then be included (e.g., in the microphone example, radio- and TV-related companies that could also bring their networks and assets to bear). That said, the concept of what constitutes a payload does not necessarily need to be technically challenging and expensive. Well-designed yet "low-tech" student experiments, for example, are already being accepted for inclusion in activities on ISS. Beyond specific public engagement payloads, images and video will become increasingly important to sharing the adventure of moon and Mars exploration by humans and their robotic partners. Beyond the scope of any public engagement budget, a communications infrastructure that enhances the capabilities of DSN and related broadband technologies supporting content delivery nationwide is essential to supporting eventual live programs and virtual experiences (e.g., based on gaming environments) from the moon and Mars.

#### 8.5.4 Workforce pipeline development

The ultimate in public participation is that people who are reached through public engagement programs would find careers within NASA, space-related industry, and academia. It is important to ensure that NASA will have the required capacity to conduct missions to the moon and Mars, which will require a broad range of disciplines (engineering, science, management) and interdisciplinary research and cooperation.

Ensuring an adequate scientific and engineering workforce is a long-term issue, and is one that cannot be achieved by government alone. It requires participation from the private sector and the educational community, and is a cross-cutting issue that is faced by the lunar community as well. Updating the recommendations that were originally proposed in the Mars Exploration Roadmap (2005), the following steps must be taken early:

- Characterize the size, composition, and health of the current scientific and engineering workforce
- Determine specifically how many people will be needed, and in what careers (in 5-year increments or by program milestone), to support Mars (and lunar) exploration
- Develop strategies to assure a capable workforce by placing special emphasis on developing a capability in subfields that need particular attention (e.g., entry, descent, and landing at Mars)
- Assess and remediate barriers to entry in key fields, with industry and academia involved

That knowledge largely has to be developed by the technical community; but once it is understood, public engagement efforts can be applied to attracting and retaining the best talent, creating education and communications programs for progressively sophisticated participation in those areas.

# 9 ACRONYMS AND ABBREVIATIONS

| | |
|---|---|
| A/L | airlock |
| AC | aerocapture |
| ADD | architecture design document |
| ADP | Advanced Development Project |
| AEDL | aerocapture and entry, descent, and landing |
| AFC | alkaline fuel cell |
| AFL | p. 48 |
| AFRL | Air Force Research Laboratory |
| AG | artificial gravity |
| Al | aluminum |
| ALARA | as low as reasonably achievable |
| ALHAT | Autonomous Landing and Hazard Avoidance Technology |
| ALSEP | Apollo lunar surface experiments package |
| AM | Arsia Mons |
| amu | air movement unit |
| ANL | Argonne National Laboratory |
| AO | announcement of opportunity |
| Ar | argon |
| ARA | Ablative Research Associates |
| ARC | Ames Research Center |
| ARMD | Aeronautics Research Mission Directorate |
| ASE | airborne support equipment |
| ASRG | advanced Stirling radioisotope generator |
| ATK | Alliant Techsystems Inc. |
| ATO | abort to orbit |
| ATP | authority to proceed |
| ATR | advanced test reactor |
| AU | astronomical unit |
| BCF | bioconcentration factor |
| BET | best-estimated trajectory |
| BFO | blood-forming organ |
| BHP | behavioral health and performance |
| BIT | built-in test |
| BL | barrel length |
| BLDT | balloon-launched decelerator test |
| BNTR | bimodal nuclear thermal rocket |
| BPA | Boeing phenolic ablator |
| BRI-18 | Boeing Reusable Insulation |
| BSM | booster separation motor |
| C | carbon |
| | commuter |
| C3I | command, control, communications, and intelligence |

| | |
|---|---|
| $C_L$ | lift coefficient |
| CAP | CEV Aerosciences Project |
| CB | Chasma Boreale |
| CBRS | cryogenic boil-off reduction system |
| CD-0 | Critical Decision-0 |
| CDR | Critical Design Review |
| CEV | crew exploration vehicle |
| CFD | computational fluid dynamics |
| CFM | cryogenic fluid management |
| $CH_4$ | methane |
| CHP | crew health and performance |
| CI | confidence interval |
| CL | confidence level |
| CM | Centauri Montes |
| CME | coronal mass ejection |
| CNS | central nervous system |
| | Communications and Navigation System |
| CO | carbon monoxide |
| $CO_2$ | carbon dioxide |
| COSPAR | Committee on Space Research |
| CTF | contained test facility |
| CVD | chemical vapor deposition |
| CxP | Constellation Program |
| D | diameter |
| DAF | device assembly facility |
| DAV | descent/ascent vehicle |
| DC | direct current |
| DDT&E | design, development, test, and evaluation |
| DE | demonstration engine |
| DEM | digital elevation model |
| DFE | direct from Earth |
| DM | docking module |
| DIAL | differential absorption LIDAR |
| DIPS | dynamic isotope power system |
| DoD | Department of Defense |
| DOE | Department of Energy |
| DOF | degree-of-freedom |
| DPT | Decadal Planning Team |
| DRA | design reference architecture |
| DRI | Desert Research Institute |
| DRM | design reference mission |
| DSN | Deep Space Network |
| DSO | detailed supplementary objective |
| DTE | direct to Earth |
| DTN | delay-tolerant networking |
| DTO | detailed test objective |
| DWC | dry-weight contingency |

| | |
|---|---|
| EAST | electric air shock tube |
| ECLS | environmental control and life support |
| ECLSS | environmental control and life support system |
| EDL | entry, descent, and landing |
| EDS | Earth departure stage |
| EELV | evolved expendable launch vehicle |
| EI | entry interface |
| EIRP | effective isotropic radiated power |
| EM | electromagnetic |
| EMAD | engine maintenance assembly/disassembly |
| EMT | emergency medical technician |
| EMU | extravehicular mobility unit |
| EOR&D | Earth orbit rendezvous and docking |
| ERV | Earth return vehicle |
| ESA | European Space Agency |
| ESAS | Exploration Systems Architecture Study |
| ESMD | Exploration Systems Mission Directorate |
| ET | external tank |
| ETDP | Exploration Technology Development Program |
| ETO | Earth to orbit |
| ETS | effluent treatment system |
| EV | extravehicular |
| EVA | extravehicular activity |
| | |
| FC | fuel cell |
| FCT | Flight Control Team |
| FE | flight-type engine<br>fuel element |
| FIAT | p. 309 |
| FOM | figure of merit |
| FPA | flight path angle |
| FPR | flight performance reserve |
| FRSI | flexible reusable surface insulation |
| FSP | fission surface power |
| FSPS | fission surface power station |
| FTIR | Fourier transform infrared spectrometer |
| FY | fiscal year |
| | |
| G/T | gain/temperature |
| G&A | General and Administrative |
| GaAs/Ge | gallium arsenide/Germanium |
| GC | p. 30 |
| GCM | general circular model |
| GCR | galactic cosmic radiation |
| $GH_2$ | gaseous hydrogen |
| GHe | gaseous helium |
| GN&C | guidance, navigation, and control |
| $GO_2$ | gaseous oxygen |
| GPHS | general-purpose heat source |
| GPR | ground-penetrating radar |
| GPS | global positioning system |
| GR | general theory of relativity; general relativity |
| GR&A | ground rules and assumptions |
| GRC | Glenn Research Center |
| GS | ground system |
| GSE | ground support equipment |
| GTF | ground test facility |
| | |
| $H_2$ | hydrogen |
| $H_2O$ | water |
| $H_2O_2$ | hydrogen peroxide |
| $H_2S$ | hydrogen sulfide |
| HCN | hydrogen cyanide |
| He | helium |
| HEM-SAG | Human Exploration of Mars-Science Analysis Group |
| HFEF | hot fuel examination facility |
| HGA | high-gain antenna |
| HLLV | heavy-lift launch vehicle |
| HPF | Hazardous Processing Facility |
| HRP | Human Research Program |
| HSRM | human science reference mission |
| HTPB | hydroxy-terminated polybutadiene |
| HUD | Department of Housing and Urban Development |
| HYPAS | Hybrid Predictor-corrector Aerocapture Scheme |
| | |
| ID | internal diameter |
| IFM | in-flight maintenance |
| IMLEO | initial mass in low-Earth orbit |
| IMU | inertial measurement unit |
| INL | Idaho National Laboratory |
| IOC | initial operational capability |
| IP | Internet protocol |
| IPP | Innovative Partnership Program |
| IR | infrared |
| $I_{sp}$ | specific impulse |
| ISPP | p. 340 |
| ISRU | in-situ resource utilization |
| ISS | International Space Station |
| IV | intravehicular |
| IVHM | integrated vehicle health monitoring |
| | |
| JAXA | Japan Aerospace Exploration Agency |
| JIMO | Jupiter Icy Moons Orbiter |
| JPL | Jet Propulsion Laboratory |
| JSC | Johnson Space Center |
| | |
| KDP A | Key Decision Point/Phase A |
| $klb_f$ | pounds force |
| KSC | Kennedy Space Center |
| kWe | kilowatt (electrical) |

| | |
|---|---|
| L | length |
| L/D | lift-to-drag |
| LAD | liquid acquisition device |
| LANL | Los Alamos National Laboratory |
| LAT | Lunar Architecture Team |
| LC-39 | Launch Complex 39 |
| LCC | life cycle cost |
| $LCH_4$ | liquid methane |
| LCT | lunar communications terminal |
| LED | light-emitting diode |
| LEO | low-Earth orbit |
| LET | linear-energy transfer |
| $LH_2$ | liquid hydrogen |
| Li | lithium |
| LIBS | laser-induced breakdown spectroscopy |
| LIDAR | laser imaging detection and ranging |
| LMA | Lockheed Martin Aeronautics |
| $LN_2$ | liquid nitrogen |
| $LO_2$ | liquid oxygen |
| LOC | loss of crew |
| LOM | loss of mission |
| LOS | loss of signal |
| LRS | lunar relay satellite |
| LRU | line replaceable unit |
| LRV | lunar rover vehicle (Apollo) |
| LV | lunar vehicle |
| LVA | launch vehicle analysis |
| LW | p. 27 |
| MADS | modular auxiliary data system |
| MARSAT | Mars aerostationary relay satellite |
| MAT | Mars Architecture Team |
| MAV | Mars ascent vehicle |
| MAWG | Mars Architecture Working Group |
| MCATS | microchannel chemical and thermal systems |
| MCC | Mission Control Center |
| MCS | maintenance control system |
| MCT | Mars communication terminal |
| MDS | Mars departure stage |
| MEA | membrane-electrode-assembly |
| MEIT | multi-element integration test |
| MEL | Master Equipment List |
| MEMS | micro-electromechanical system |
| MEPAG | Mars Exploration Program Analysis Group |
| MER | Mars exploration rover |
| MET | metrology |
| MFC | materials and fuel complex |
| MGCM | Mars global circulation model |
| MGS | Mars geosynchronous satellite |
| MH | Mars habitat |
| MIA | Mars ISRU Architecture |
| MIP | Mars in-situ propellant production precursor |
| ML3N | p. 48 |
| MLI | multilayer insulation |
| MM | mesoscale model |
| MMH/NTO | monomethyl hydrazine nitrogen tetroxide |
| MMOD | micrometeoroid and orbital debris |
| MN | Mars Network |
| MOC | Mars orbital capture (p. 70) |
| MOD | modification |
| MOI | Mars orbit insertion |
| MOLA | Mars orbiter laser altimeter |
| MPF | Mars Pathfinder |
| MPS | main propulsion system |
| MR | mixture ratio |
| MRO | Mars Reconnaissance Orbiter |
| MRS | Mars relay satellite |
| MSFC | Marshall Space Flight Center |
| MSL | Mars Science Laboratory |
| MR | mixture ratio |
| MSR | Mars Sample Return |
| MT | Mars transit (p. 99) |
| MTV | Mars transfer vehicle |
| MXH | p. 271 |
| $N_2$ | nitrogen |
| NAFCOM | NASA/Air Force Cost Model |
| NAS | National Academy of Sciences |
| NCRP | National Council on Radiation Protection |
| NEPA | National Environmental Policy Act of 1969 |
| NERVA | Nuclear Engine for Rocket Vehicle Applications |
| NEXT | NASA Exploration Team |
| NF | Nili Fossae |
| NH | ammonia |
| NOAA | National Oceanic and Atmospheric Administration |
| $^{237}Np$ | Neptunium-237 |
| NPD | NASA Policy Directive |
| NPF | nuclear processing facility |
| NPLD | p. 23 |
| NPR | NASA Procedural Requirements |
| NR | naval reactor |
| NRC | National Research Council |
| NRX-XE | NERVA experimental engine |
| NSF | National Science Foundation |
| NSPS | nuclear surface power system |
| NTP | nuclear thermal propulsion |
| NTR | nuclear thermal rocket |
| NTS | Nevada Test Site |

| | |
|---|---|
| $O_2$ | oxygen |
| $O_3$ | ozone |
| O/F | outfitting |
| OD | outer diameter |
| OMG | optical mass gauge |
| OML | outer mold line |
| ORU | orbital replacement unit |
| OSF | offline stacking facility |
| %REID | percent radiation-exposure-induced death |
| P-STAR | Propulsion Sizing, Thermal, Accountability, and Weight Relationship First Order Modeling Tool |
| PA | parachute assembly |
| PBAN | polybutadiene acrylic acid acrylonitril terpolymer |
| PBL | planetary boundary layer |
| PC | propulsive capture |
| PDR | Preliminary Design Review |
| PEL | permissible exposure limit |
| PEMFC | proton exchange membrane fuel cell |
| PFC | primary fuel cell |
| PI | p. 283 |
| PICA | phenolic impregnated carbon ablator |
| PIE | post-irradiation examination |
| PK | pharmacokinetics |
| PLOC | probability of loss of crew |
| PLOM | probability of loss of mission |
| PLSS | portable life support system |
| PMAD | power management and distribution |
| PMD | propellant management device |
| PNNL | Pacific Northwest National Laboratory |
| PNT | position, navigation, and time |
| POD | point of departure |
| POST | Program to Optimize Simulated Trajectories |
| pptv | p. 27 |
| PRA | probabilistic risk assessment |
| PRD | Program Requirements Document |
| PROMISE | Production of Resources On Mars In-Situ for Exploration |
| psia | pounds per square inch, absolute |
| $^{238}Pu$ | plutonium-238 |
| $PuO_2$ | plutonium dioxide |
| PV | photovoltaic |
| PVA | photovoltaic array |
| PVT | pressure-volume-temperature |
| R/T | roll through |
| R&D | rendezvous and docking |
| R&R | rest and relaxation |
| RCAP | rapid cycle adsorption pump |
| RCS | reaction control system |
| RF | radio frequency |
| RFC | regenerative fuel cell |
| RFID | radio frequency identification |
| RHU | radioisotope heater unit |
| RM | redundancy management |
| RPS | radioisotope power system |
| RSRB | reusable solid rocket booster |
| RSS | root sum square |
| RTG | radioisotope thermoelectric generator |
| Ru | Rutherfordium |
| RWGS | reverse water gas shift |
| S/C | spacecraft |
| SAFE | subsurface active filtration of exhaust |
| SBIR | Small Business Innovative Research |
| SCAWG | Space Communications Architecture Working Group |
| SDR | signal data recorder |
| SEI | Space Exploration Initiative |
| SEM | scanning electron microscope |
| SEP | Strong Equivalence Principle<br>sun-Earth-probe |
| SHAB | surface habitat |
| SI | System International |
| SINDA | Simplified Improved Numerical Difference Analyzer |
| SLA | Super Lightweight Ablator |
| SMD | Science Mission Directorate |
| $SO_2$ | sulfur dioxide |
| SOCE | solid oxide $CO_2$ electrolysis |
| SOMD | Space Operations Mission Directorate |
| SOTA | state-of-the-art |
| SPE | solar particle event |
| SPS | surface power system |
| SRB | solid rocket booster |
| SRU | shop replaceable unit |
| SSB | Space Studies Board |
| SSPF | Space Station Processing Facility |
| STAB | p. 309 |
| STScI | Space Telescope Science Institute |
| SW | p. 27 |
| t | metric ton |
| T/W | thrust-to-weight |
| TBR | to be resolved |
| TC | telecommuter |
| TDRSS | Tracking and Data Relay Satellite System |
| TDU | technology demonstration unit |
| TEI | trans-Earth injection |
| TES | thermal enclosure system |
| THEMIS | thermal emission imaging system |
| TLI | trans-lunar injection |
| TMI | trans-Mars injection |
| TOF | time of flight |
| TPA | turbopump assembly |

| | |
|---|---|
| TPS | thermal protection system |
| TRL | Technology Readiness Level |
| TRN | terrain-relative navigation |
| TT&C | telemetry, tracking, and communications |
| TWTA | traveling wave tube amplifier |
| | |
| $^{235}U$ | enriched uranium |
| UAV | uncrewed aerospace vehicle |
| $UC_2$ | uranium carbide |
| UHF | ultra-high frequency |
| ULF | ultra-low frequency |
| $UO_2$ | uranium dioxide |
| UofA | University of Arizona |
| | |
| VAB | Vertical Assembly Building |
| VIF | Vertical Integration Facility |
| VUV | vacuum ultraviolet |
| | |
| W | tungsten |
| WE | water electrolysis |
| WEI | work efficiency index |
| WLAN | Wireless Local Area Network |
| | |
| XRD | x-ray diffraction |
| | |
| YSZ | yttria-stabilized Zirconia |
| | |
| ZBO | zero-boiloff |
| ZBR | zone of minimum biological risk |
| ZrC | zirconium carbide |

# APPENDIX A: MEMBERSHIP

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13. ABSTRACT *(Maximum 200 words)*

This appendix to the Mars Design Reference Architecture describes the systems and operations that would be used for the first three missions to explore the surface of Mars by humans. These missions would occur on three consecutive trajectory opportunities within the next several decades. This minimum set was chosen for this reference architecture because the development time and cost to achieve the basic capability to carry out a single human Mars mission are of a magnitude that a single mission, or a pair of missions, is difficult to justify. Moreover, three consecutive missions would require approximately 10 years to complete; a period of time that is sufficient to achieve basic program goals and acquire a significant amount of knowledge and experience, making this a likely point in time to consider new goals and improved architectures to achieve them. These first three human Mars missions are also assumed to have been preceded by a sufficient number of test and demonstration missions on Earth, in the International Space Station, in Earth orbit, on the moon, and at Mars (by robotic precursors) to achieve a level of confidence in the architecture such that the risk to the human crews is considered acceptable.

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