National Aeronautics and
Space Administration

Headquarters
Washington, DC 20546-0001

May 18, 2018

Dear Colleague,

The draft Proposal Information Package (PIP) and Environmental Requirements Document (ERD) were written to support an Announcement of Opportunity (AO) for instruments for a potential Europa lander mission. For that use the information and parameters in a PIP and ERD are typically considered binding. For the Instrument Concepts for Europa Exploration 2 (ICEE 2) opportunity, the PIP and ERD serve a different purpose. They provide information and parameters that reflect our current best understanding of the mission concept. But that understanding continues to evolve with further study, and the ICEE 2 effort provides an invaluable opportunity for instrument developers to collaborate with the lander team on instrument accommodation to further advance that understanding. Under this scenario the PIP and ERD should be viewed as informational rather than binding as it is expected that some changes will arise because of this collaboration.

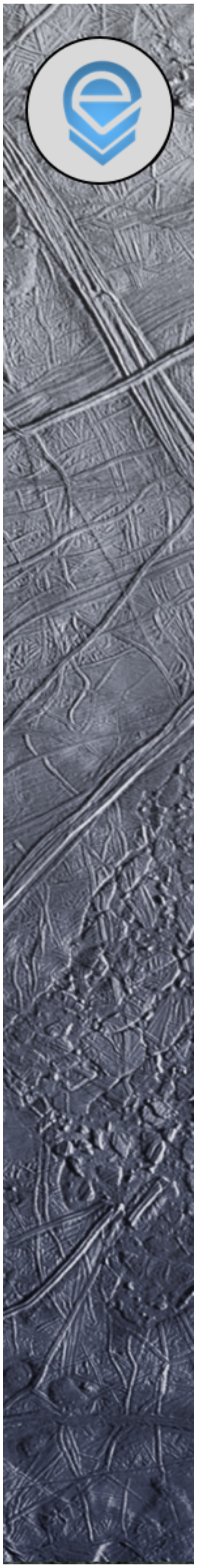
Some limitations in the draft PIP are in place because the resources needed to provide them were judged too excessive to accommodate. Specifically, the draft PIP does not support the following capabilities:

- Capability to deploy (i.e., place) of an instrument(s) on the surface using the robotic arm;
- Capability to locate instrument(s) outside of the vault (except for the Context Remote Sensing Instrument); and
- Capability to mount an instrument(s) on the robotic arm.

However, proposers may submit instrument concepts requiring these capabilities provided that a) the additional resources likely needed (additional survival heating, cabling, radiation shielding, etc.) come from the instrument's resource allocation and/or b) the proposal describes an implementation demonstrating that additional resources are unneeded.

Any further questions or requests for clarifications should be emailed to me. Good luck on your proposals!

Dr. Curt Niebur
Program Officer, ICEE 2



Europa Lander Project

Preliminary Environmental Requirements Document

Draft Release

Prepared by:

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Date

Paper copies of this document may not be current and should not be relied on for official purposes. The current version is available in PDMS, https://pdms.jpl.nasa.gov.
Pre-Decisional Information — For Planning and Discussion Purposes Only

May 23, 2018

JPL D-97633

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The research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

Approved:

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CHANGE LOG

DATE	SECTIONS CHANGED	REASON FOR CHANGE	REVISION
04/26/2018	ALL	New	Preliminary Draft Release

Table of Contents

1 Introduction

1.1 Purpose

This document defines environmental design and verification requirements for the Europa Lander Mission. Successful definition and implementation of the environmental requirements is expected to result in flight hardware that is fully compatible with all anticipated natural or induced ground, launch, and mission environments. This version is a Preliminary Draft which will be updated prior to the first official release.

The following definitions are used throughout this document:

"Shall" = required
"Should" = recommended
"Will" = planned to be carried out

1.2 Applicable Documents

The following documents form a part of this document to the extent specified herein. Unless otherwise specified, the current issue of the document applies.

1.2.1 Government Documents

MIL-STD 461C	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference, 4 August 1986.
MIL-STD-461F	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference, 10 December 2007.
MIL-STD-462	Electromagnetic Interference Characteristics, Measurement of.
MIL-STD-704F	Aircraft Electric Power Characteristics

1.2.2 Launch System User Guides

Falcon Heavy	Falcon Heavy Payload Planner's Guide, [**TBD**]
Delta IV Heavy	Delta IV Launch Services User's Guide, United Launch Alliance, June 2013.
SLS	Space Launch System Payload Planner's Guide, [**TBD**].

1.3 NASA Reference Documents

The following reference documents include various NASA guidelines which are called out in this document for further information on design or test guidance. Others are NASA standards, upon which some of the requirements in this document are based.

NASA-HDBK-4002A	Mitigating In-Space Charging Effects – A Guideline, Mar. 3, 2011
NASA-STD-4003A	Electrical Bonding for NASA Launch Vehicles, Spacecraft, Payloads, and Flight Equipment, January 19, 2016.
NASA-STD-5001B	Structural Design and Test Factors of Safety for Spaceflight Hardware, August 6, 2014.
NASA-STD-7001A	Payload Vibroacoustic Test Criteria, January 20, 2011.
NASA-STD-7003A	Pyroshock Test Criteria, December 20, 2011.
NASA-HDBK-7004C	Force Limited Vibration Testing Handbook, November 30, 2012.
NASA-HDBK-7008	Spacecraft Dynamic Environments Testing, June 12, 2014.
GSFC-STD-7000A	General Environmental Verification Specification, April 22, 2013

1.4 Project Reference Documents

The following is a list of relevant project documents. Not all documents have been released.

JPL D-97628	Europa Lander Safety and Mission Assurance Plan
JPL D-97629	Europa Lander Parts Program Requirements
JPL D-97630	Europa Lander Reliability Assurance Requirements

JPL D-97634 Europa Lander Radiation Control Plan (RCP)
JPL D-97635 Europa Lander Electromagnetic Environmental Control Plan
JPL D-97636 Europa Lander Surface Charging/iESD Control Plan
JPL D-97653 Europa Lander Planetary Protection Plan
JPL D-xxxxx Europa Lander Test and Analysis Matrix (TAM)
JPL D-xxxxx Europa Lander Temperature Requirements Table (TRT)

1.5 Project Background

1.5.1 Europa Lander Flight System Components and Nomenclature

The Europa Lander flight system is comprised of a collection of elements. Single system elements are referred to as a 'stage.' Multiple 'stages' combined into a single element constitute a 'vehicle.' There are four stages in the complete Europa Lander flight system. Table 1.5 -1 summarizes the vehicle and stage configurations during key mission events. Light shading denotes vehicles comprised of multiple stages and darker shading indicates individual stages.

Table 1.5-1 Constituent stages and vehicles for the Europa Lander flight system during key mission events. [TBR]

<table>
<tr><th>Launch</th><th>Deorbit</th><th>Descent</th><th>Surface Ops</th></tr>
<tr><td rowspan="4">Cruise Vehicle</td><td>Carrier and Relay Stage</td><td>Carrier and Relay Stage</td><td>Carrier and Relay Stage</td></tr>
<tr><td rowspan="3">Deorbit Vehicle</td><td>Deorbit Stage</td><td>Deorbit Stage</td></tr>
<tr><td rowspan="2">Powered Descent Vehicle</td><td>Descent Stage</td></tr>
<tr><td>Lander</td></tr>
</table>

Each stage may be considered its own 'system' with regards to system-level or assembly-level test criteria based on environment applicability. The Europa Lander Test and Analysis Matrix (JPL D-xxxxx; **TBD**) will specify system-level configurations for environmental tests and analyses.

1.5.2 Europa Lander Launch Vehicle

The baseline launch vehicle for Europa Lander mission is the Space Launch System (SLS) Block 1-B. In general, reliable launch environment estimates are not currently available for this vehicle. As needed, estimates of environments based available data from the Evolved Expendable Launch Vehicle (EELV) Delta-IV Heavy and/or Falcon Heavy are provided instead. This document will be updated when launch environment estimates are available for the SLS or another launch vehicle is selected.

2 Environmental Program and Verification Requirements

2.1 General

The environmental design and verification program is intended to demonstrate, through design, test and/or analysis methods, the ability of the Europa Lander flight system to successfully survive and operate within specification over the natural and/or induced ground, launch, cruise, and mission operations environments with sufficient margins.

All requirements in this document include a '**shall** statement' and are demarcated further with a '**Requirement**' heading. Some requirements also include an additional '**Policy**' heading to indicate that the intended requirement is non-technical.

2.2 Environmental Verification Requirements

Requirement-Policy: The flight system, instruments, and subsystems/assemblies **shall** be verified to be compatible with the environmental design and test levels presented in Section 4 of this document.

The environmental verification process outlined in Figure 2.2-1 includes test and analysis methods. Aspects of this flow are described in the sections that follow.

Requirement-Policy: The verification activities (test & analysis) **shall** comply with the project-required margins, levels, and durations as specified in Table 2.2-1, unless explicitly specified in Section 4 of this document.

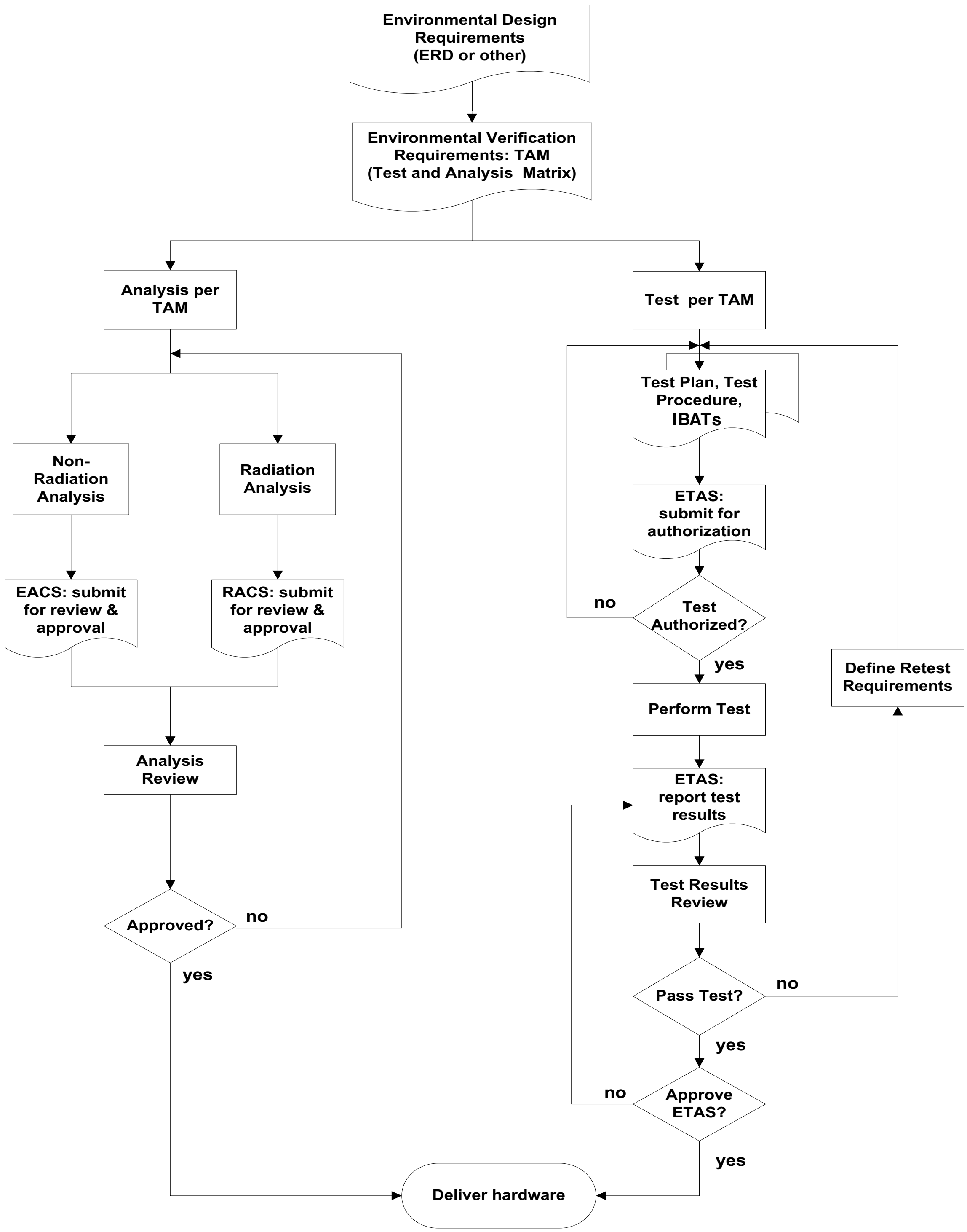


Figure 2.2-1 Europa Lander Project Environmental Program Flow

Table 2.2-1 Europa Lander Project Environmental Design and Test Margin Requirements

	Assembly/Subsystem Level			Flight System Level
Environment	**Design/Qualification (Qual)**	**Protoflight (PF)**	**Flight Acceptance (FA)**	**Protoflight (PF)**
Acoustics: Level (note 1) Duration	Envelope of (MEFL+3dB) and Minimum Workmanship level (note 10) 2 x FA duration (2 min minimum)	Envelope of (MEFL+3dB) and Minimum Workmanship level (note 10) FA duration (1 min minimum)	Envelope of MEFL and Minimum Workmanship level (note 10) 1 min minimum	Envelope of (MEFL+3dB) and Minimum Workmanship level (note 10) 1 min minimum
Random Vibration: Level Duration	FA level + 3dB 2 x FA duration (2 min/axis)	FA level + 3dB FA duration (1 min/axis)	Envelope of MEFL and Minimum Workmanship level (note 11) 1 min/axis	MEFL + 3dB 1 min/axis
Simulated Pyro Shock	1.4 x FA level 2 shocks/axis (note 8)	1.4 x FA level 1 shocks/axis (note 8)	MEFL 1 shock/axis (notes 6 & 8)	N/A (no test required)
Pyro Device Firings	2 firings (note 7)	2 firings (note 7)	N/A (no test required)	2 firings (dominant shock sources), 1 firing (other sources)
Quasi Static Loads:	per NASA-STD-5001B (see note 9)	per NASA-STD-5001B (see note 9)	per NASA-STD-5001B (see note 9)	
Thermal: Test Media (note 2)	• Thermal Vacuum: Flight System, Payload, Assemblies/Subsystems • Atmosphere/GN2: Assemblies, Payloads (on case-by-case basis, upon Project approval only)	• Thermal Vacuum: Flight System, Payload, Assemblies/Subsystems • Atmosphere/GN2: Assemblies, Payloads (on case-by-case basis, upon Project approval only)	• Thermal Vacuum: Flight System, Payload, Assemblies/Subsystems • Atmosphere/GN2: Assemblies, Payloads (on case-by-case basis, upon Project approval only)	• Thermal Vacuum: Flight System Thermal Balance Phase: -Simulate extreme Space Thermal Environments (per Section 4.6.3), along with worst-case power modes and interface boundary conditions. Flight System Thermal Margin Phase: Drive key assemblies to AFT limits for system functional margin demonstration
Temp. Levels (Test Margins) (note 4)	• Electronics: **Cold**: (AFTcold-15ºC) or -35ºC (whichever is colder) **Hot:** (AFThot+20ºC) or +70ºC (whichever is warmer) • Mechanisms without Electronics, Optics, Detectors, and Others: **Cold:** (AFTcold-15ºC) **Hot:** (AFThot+20ºC)	• Electronics: **Cold**: (AFTcold-15ºC) or -35ºC (whichever is colder) **Hot:** (AFThot+20ºC) or +70 ºC (whichever is warmer) • Mechanisms without Electronics, Optics, Detectors, and Others: **Cold:** (AFTcold-15ºC) **Hot:** (AFThot+20ºC)	• Electronics: **Cold**: (AFTcold-5ºC) or -25ºC (whichever is colder) **Hot:** (AFThot+5ºC) or +55ºC (whichever is warmer) • Mechanisms without Electronics, Optics, Detectors, and Others: **Cold:** (AFTcold-5ºC) **Hot:** (AFThot+5ºC)	
Thermal Test Duration (note 3)	• Flight System & Payload Electronics: **Op:** 24 hrs cold/72 hrs hot **Non-Op:** 6 hrs cold/6 hrs hot • Non-Electronics: **Op:** Hot and cold dwell times as needed to run tests for all required functions. **Non-Op:** 6 hrs cold/6 hrs hot	• Flight System & Payload Electronics: **Op:** 24 hrs cold/72 hrs hot **Non-Op:** 6 hrs cold/6 hrs hot • Non-Electronics: **Op:** Hot and cold dwell times as needed to run tests for all required functions. **Non-Op:** 6 hrs cold/6 hrs hot	• Flight System & Payload Electronics: **Op:** 8 hrs cold/60 hrs hot **Non-Op:** 6 hrs cold/ 6 hrs hot • Non-Electronics: **Op:** Hot and cold dwell times as needed to run tests for all required functions. **Non-Op:** 6 hrs cold/6 hrs hot	
Number of Thermal Cycles (note 5)	• 3 cycles minimum to 10 cycles max (cumulative)	• 3 cycles minimum to 10 cycles max (cumulative)	• 3 cycles minimum to 10 cycles max (cumulative)	
Temperature Ramp Rate	• $\|dT/dt\| \le 5^{o}C/min$	• $\|dT/dt\| \le 5^{o}C/min$	• $\|dT/dt\| \le 5^{o}C/min$	
Number of Thermal Startups	• 3 cold/3 hot (**Op**) • 3 cold (**Non-Op**, if self-heat & no heaters)	• 3 cold/3 hot (**Op**) • 3 cold (**Non-Op**, if self-heat & no heaters)	• 3 cold/3 hot (**Op**) • 3 cold (**Non-Op**, if self-heat & no heaters)	

	Assembly/Subsystem Level			Flight System Level
Environment	**Design/Qualification (Qual)**	**Protoflight (PF)**	**Flight Acceptance (FA)**	**Protoflight (PF)**
Planetary Protection		Dry Heat Microbial Reduction: 125ºC [TBR] for up to 1329 hours [TBR], **Non-op** (note 14)	Dry Heat Microbial Reduction: 125ºC [TBR] for up to 1329 hours [TBR], **Non-op** (note 14)	
Depressurization	>1.5 x max dP/dt (note 13)	>1.5 x max dP/dt (note 13)		
EMC (RE, RS, CE, CS)	MEFL +6 dB (susceptibility) MinEFL -6 dB (emissions)	MEFL +6 dB (susceptibility) MinEFL -6 dB (emissions) (see note 6)	N/A (grounding/isolation testing only) CE may be invoked for FM hardware	Launch Sources: Minimum Expected Flight Level -6 dB (emissions) Maximum Expected Flight Level +6dB (susceptibility) Flight System Self - Compatibility at Maximum Expected Flight Level
Internal Electro-Static Discharge (note 12)	IESD Design Factor = 2 [TBR]			
Charge Particle Radiation (TID/DDD)	Radiation Design Factor (RDF) = 2 Spot shielding, RDF = 3			

Notes for Table 2.2-1:

1. MEFL = Maximum Expected Flight Level; MinEFL= Minimum Expected Flight Level.
2. All assemblies will be tested in vacuum ($<10^{-5}$ Torr) unless otherwise exempted.
3. Test duration requirement is cumulative of the test duration employed during thermal cycling.
4. AFT = Allowable Flight Temperature, typically includes both operational and non-operational limits. The number of thermal cycles performed on flight hardware (PF or FA) will be sufficient to detect workmanship defects, mechanical problems, or electrical hysteresis.
5. Typically this is 3 to 10 cycles (except for purely mechanical/structural assemblies). Unless otherwise approved by the Project Environmental Requirements Engineer (ERE), no more than 10 cycles (inclusive of all retest activities) will be performed on flight hardware prior to ATLO (Assembly, Test, and Launch Operations) delivery.
6. For pyrotechnic shock and EMC testing, if there is no EM available for Qualification, then a Protoflight test will be performed on a single PF unit. No test required for remaining flight units.
7. Each pyrotechnic device contained within protoflight or qualification hardware will be fired a minimum of two times in order to characterize the device functionality and the resultant shock responses. Shock levels generated by firings of flight or flight-like pyro devices will not provide a 3 dB Design/Qual/PF level margin and therefore is not a valid PF or Qual Pyroshock test. Shock-sensitive subassemblies within the assembly/subsystem should be assessed for possible Qual/PF level pyroshock testing.
8. For assemblies/subsystems not containing shock-producing devices, shock testing is to be performed at PF test level of 3 dB above FA level with one shock in each of the three orthogonal axes. Qualification tests are the same level as PF tests but with 2 shocks per axis. If performed, FA test levels are at MEFL.
9. JPL takes exception to the NASA-STD 5001B factor of safety for glass. The JPL required ultimate factor of safety for glass will be a minimum of 2.0, and the acceptance proof test factor will be 1.2 or greater.
10. Minimum Workmanship Acoustics Level is 138 dB Overall SPL.
11. Minimum Workmanship Random Vibration Level is 6.8 g_{rms} overall.
12. IESD prone materials will be subject to the Europa Lander Surface Charging/iESD Control Plan (JPL D-97636).
13. Design criteria is conservatively met when V/A<2000 in or 5080 cm. (V=Volume, A=Vent area, P=pressure, t=time.)
14. This is a representative bounding temperature/duration profile. See Europa Lander Planetary Protection Plan (D-97653) for other acceptable temperature/duration profiles. Additional Vapor Hydrogen Peroxide (VHP) processing may also be required.

2.3 Environmental Testing

Environmental testing approaches are categorized for the purpose of hardware quality verification as Qualification (Qual), Protoflight (PF), and Flight Acceptance (FA).

Requirement-Policy: Flight, spare, and qualification hardware design, performance, and workmanship quality **shall** be verified by test using a Protoflight (PF) test program or a Qualification/Flight Acceptance (Q/FA) test program.

Project Environmental Requirements Engineers (EREs) will approve the verification approach for flight hardware through development of the Test and Analysis Matrix (TAM) (JPL D-xxxxx **TBD**) in conjunction with the engineering teams.

Requirement-Policy: Formal environmental tests **shall** be performed on all flight, spare, and qualification hardware at the level of assembly indicated in the environmental Test and Analysis Matrix (TAM) (JPL D-xxxxx **TBD**).

Requirement-Policy: All formal environmental tests shall be authorized prior to testing and summarized subsequent to testing using the Environmental Test Authorization and Summary form (Appendix B: ETAS Form (Environmental Test Authorization Summary), JPL Form 2683).

2.3.1 Qualification Test Approach

Qualification (Qual) testing is performed on a dedicated Qualification (or Engineering) Model of the flight hardware that is not intended to fly, in order to qualify the hardware design for the maximum expected flight environment plus margin, including margin on environment duration or cycles.

If approved by the Project ERE, an engineering model of an assembly may be used as a qualification unit and be subjected to qualification environmental testing.

Requirement-Policy: If used for qualification testing, the engineering model **shall** be flight-like and manufactured using the same assembly techniques and fabrication processes as the flight hardware including: structure, thermal design, shielding, cabling, circuit layout, power consumption, functional modes, and electrical parts with the same signal characteristics.

Requirement-Policy: Hardware that has been used as a qualification unit and is being considered for use as a flight unit or spare **shall** be evaluated upon completion of testing to determine the details of refurbishment and retest, if any, and an assessment of the residual risk to the project for using the hardware.

2.3.2 Protoflight Test Approach

Protoflight (PF) testing is performed on flight hardware and serves to simultaneously fulfill the requirements of design qualification and workmanship demonstration for flight acceptance. Protoflight environmental testing demonstrates design adequacy and flight hardware readiness, including appropriate performance and margin.

2.3.3 Flight Acceptance Test Approach

Flight Acceptance (FA) testing is performed on flight hardware and spares only when a previous qualification test has been performed on an identical item. If, as determined by a Heritage Review, previous qualification test levels of a heritage assembly are adequate for the mission and the heritage design and operation is not modified in a way that negates the previous qualification, then the assembly may be tested to Flight Acceptance levels and durations. Flight Acceptance testing may also be required to verify hardware quality and workmanship following minor modifications, rework, or repairs.

2.3.4 Assembly Level Environmental Tests

Requirement-Policy: Assembly/instrument level testing **shall** be performed prior to delivery for higher-level integration.

Requirement-Policy: Flight hardware with documentation claiming prior Qualification by Heritage or Similarity for the required environmental tests **shall** be evaluated and approved by the Project ERE.

2.4 Environmental Analyses

Environmental analyses are performed to verify hardware design compatibility with ground, transportation, storage, launch, and mission environments that may be impractical to verify by test or that are more cost effectively analyzed than tested (e.g. radiation dosage compatibility, venting, and atomic oxygen susceptibility), or where analysis is a better verification method.

Requirement-Policy: Environmental analyses **shall** be performed against the environmental design criteria in Section 4 of this document.

Requirement-Policy: Each level of assembly, where analyses are indicated, **shall** summarize those analyses using the Environmental Analysis Completion Statement (Appendix C: ECAS Form (Environmental Analysis Completion Statement), JPL Form 2566).

Requirement-Policy: Since analysis results may affect hardware design, all reports for a given hardware item **shall** be submitted to the Project Office prior to the beginning of flight hardware environmental testing.

3 Environmental Test Policies

3.1 General

This section establishes the implementation, control, and reporting policies for environmental testing of Europa Lander flight and qualification hardware, whether performed at JPL or subcontractor facilities.

Requirement-Policy: All flight and qualification hardware **shall** be environmentally tested in accordance with the requirements of this document.

Requirement-Policy: Deviations from the Environmental Program requirements **shall** require approval through one of the following processes prior to start of any environmental testing:

1. A Category A waiver for project-wide deviations from JPL institutional standards, including this standard.
2. A Category B waiver for all deviations that compromise the intent of the Environmental Program as contained in the approved project environmental documentation. This includes deviations for non-technical reasons, such as those resulting from schedule or cost constraints.
3. The Environmental Test Authorization and Summary (ETAS) form documentation of minor test deviations with technical justification with concurrence of the ERE.

3.2 Test Configuration

Requirement-Policy: Flight system level environmental testing **shall** be in a flight-like configuration per the Europa Lander Environmental Test and Analysis Matrix (TAM) (JPL D-xxxxx **TBD**).

Requirement-Policy: All assembly, instrument, and subassembly level environmental testing (including spares) **shall** be in a flight-like configuration per the Europa Lander Environmental Test and Analysis Matrix (TAM) (JPL D-xxxxx **TBD**).

Requirement-Policy: Electrical cabling, connectors, and other flight fittings normally associated with the assembly or the system **shall** be used as part of the test article.

Requirement-Policy: The same configuration **shall** be used for Qualification, Protoflight, and Flight Acceptance environmental testing.

Requirement-Policy: Hardware configurations qualified in previous environmental test programs **shall** be evaluated for consistency with Europa Lander environmental testing.

3.3 Assembly Operation/Functional Test

The hardware will operate in logic and power states that validate the integrity of all electrical circuits and interfaces, including redundant circuitry, and every effort should be made to simulate all operational modes. This includes circuits internal to the assembly and circuits that interface directly with other assemblies of the Europa Lander flight system.

Requirement-Policy: During environmental testing, hardware **shall** be operated in the appropriate functional modes as defined in their functional specifications, demonstrating that the assembly performs to specification when exposed to the test environment.

Requirement-Policy: Functional test procedures **shall** ensure all electrical circuits and interfaces, as defined in their functional specifications, are exercised.

3.4 Test Sequence

For electronic assemblies, dynamic testing should precede thermal testing (in order of flight exposure). For certain composite structures, thermal cycling should precede dynamic testing for more effective workmanship verification.

EMC testing may be conducted when convenient. However, any re-work due to EMC anomalies (e.g., connector back shell rework and gasket installations) should be completed prior to dynamics and thermal testing.

Requirement-Policy: The sequence of environmental tests on a given flight hardware assembly **shall** be established by the responsible engineer and concurred by the Project EREs, based on the flight environment sequence or a review of the hardware design and materials, the sensitivity of the assembly to each environment, and the potential effect of each environment on other environmental characteristics.

Requirement-Policy: Where applicable, the Qualification or Protoflight test article for a given assembly configuration **shall** successfully pass its Qual/PF tests prior to commencing with FA tests on an identical flight article.

3.5 Environmental Test Facilities

Any agency that performs environmental testing will do so in accordance with certain minimum standards, whether these facilities are at JPL, at a subcontractor's facility, or at an independent test laboratory. For testing performed for JPL-developed flight hardware, these minimum standards are defined in Standard Environmental Testing Facilities and Practices document. Test facility conformance to this Standard will be reviewed and evaluated by the JPL Environmental Test Lab (ETL). For hardware developed and delivered by other agencies, test facilities should include provisions to protect flight hardware from facility anomalies (i.e. power failures, temperature excursions, etc.). The applicable test standards for EMC tests are given in MIL-STD-461F/462.

Requirement-Policy: The hardware Cognizant Engineer **shall** be responsible for the handling of a test article in an environmental test facility, including attachment of any test fixture.

3.6 Environmental Test Pans

Requirement-Policy: Environmental Test Plans or Specifications shall be prepared by each supplier to define the environmental test levels and durations for assembly/subsystem and instrument level environmental testing.

Environmental test plans should cover the following topics:

- Description of the test article
- Test objectives
- Test setup, test support equipment, and test facility
- Instrumentation and data
- Test tolerances
- Environmental simulations and test media, if applicable
- Test phases, test cases, and test profiles
- Test parameters (Test levels, margins, and durations), as applicable
- Functional and performance verifications
- Success/failure criteria
- Requirements for ETAS, Problem/Failure Reporting (PFRs)
- Flight hardware and personnel protection
- QA provisions
- Post-test activities and analysis

Requirement-Policy: Environmental Test Plans and their revisions **shall** be submitted to the Project ERE for approval before beginning an environmental test.

Requirement-Policy: Environmental Test Plans **shall** be under revision control, with redlines incorporated prior to environmental testing of any redundant units.

3.7 Environmental Test Procedures

Requirement-Policy: The operation of environmental test equipment and facilities during the performance of environmental tests of flight hardware **shall** be accomplished in accordance with approved test procedures.

3.8 Environmental Test Authorization and Summary

For the purpose of assuring flight hardware readiness for environmental testing and documenting test results, the Environmental Test Authorization and Summary (ETAS) form is the document of record. For test authorization and approval as well as requirement verifications, the appropriate portions of the ETAS form, and its supporting documents, must be submitted to the Project ERE.

Requirement-Policy: For the purpose of assuring flight hardware readiness for environmental testing and documenting test results, the Environmental Test Authorization and Completion Statement (ETAS) Process **shall** be followed.

Requirement-Policy: The Test Authorization portion of the ETAS form **shall** be completed by the Cognizant Engineer for the test article and approved by the Project ERE for each flight hardware serial number prior to commencing environmental testing.

Requirement-Policy: The ETAS **shall** reference the approved test plan and procedures.

Requirement-Policy: The ETAS **shall** describe any deviations from the hardware's flight configuration.

Requirement-Policy: Upon conclusion of the environmental tests, the Test Results portion of the ETAS form **shall** be completed and approved, clearly denoting the pass/fail disposition of the flight hardware.

Requirement-Policy: The ETAS **shall** reference the environmental test reports and include a description of any anomalies recorded during environmental testing and reference the associated Problem/Failure Reports (P/FRs).

Appendix B: ETAS Form (Environmental Test Authorization Summary) contains a sample ETAS form. Additionally, the project will use the on-line ETAS in the Mission Assurance Information System (MAIS) system.

3.9 Test Failure

Requirement-Policy: Any hardware failure or malfunction during an environmental test or any failure or malfunction of an environmental test facility that would affect an environmental test **shall** be cause for the issuance of a P/FR (Problem/Failure Report).

Requirement-Policy: Hardware failure, malfunction, or out-of-specification performance during formal environmental testing **shall** be interpreted as a test failure.

For assembly-level environmental testing, the test may be continued if the Cognizant Engineer and Test Engineer agree that continuation is of diagnostic value and will not damage the flight hardware.

Requirement-Policy: At the system (and Instrument) level, the applicable test plan **shall** designate the responsible representative with the authority to determine whether or not to interrupt the test in the event of a failure or malfunction of the flight hardware.

Requirement-Policy: Failures associated with environments, but accepted by the project, **shall** be handled through the Category B Waiver process.

3.10 Test Reports

Requirement-Policy: After each assembly, subsystem, Instrument or system environmental test is terminated (whether because the test requirements were successfully completed or because a test failure has occurred) the testing agency **shall** prepare a Test Agency Report, that includes or addresses any deviations from the approved test procedure.

Requirement-Policy: For each serial number of each hardware group subjected to formal environmental testing, a report **shall** be prepared by the hardware provider.

Requirement-Policy: The report(s) **shall** be available for Project Reviews and as inputs to the assembly Delivery and Pre-Shipment Review Boards.

3.11 Re-Test Policies

Requirement-Policy: Environmental retests of assemblies **shall** be required under the following circumstances:

1. To complete the protoflight or flight acceptance testing of hardware that has failed during its environmental test program.
2. To re-qualify flight hardware design where design changes, modifications or configuration changes occur after completion of environmental testing.
3. To verify the flight worthiness of refurbished units used as flight spares.
4. To verify the flight acceptability of workmanship performed as part of rework not covered by items 1 to 3.

Requirement-Policy: Re-testing of assemblies to environmental requirements **shall** be coordinated with the Project EREs.

Requirement-Policy: The specific re-test requirements **shall** be determined jointly between the cognizant engineer and the Project EREs (with MAM concurrence).

Requirement-Policy: Flight hardware **shall** not be retested without a re-approval of the updated ETAS or test approval documentation.

4 Environmental Design and Verification Requirements

The environmental design and verification requirements contained within this section are established to assure design compatibility of Europa Lander Project flight and qualification hardware with the specified environments and corresponding mission modes.

4.1 Handling and Ground Operation Environments

The handling and ground operations environmental design requirements include the environments that the flight hardware would encounter during fabrication, integration, calibration, alignment, and pre-launch operations. The ground handling environments also include transportation and storage of the flight hardware in handling fixtures or shipping containers.

4.1.1 Transportation and Handling Dynamics Environments

Requirement: Flight hardware **shall** be designed to survive without degradation the ground transportation and handling vibration, acceleration, and shock environments specified in Table 4.1.1-1

Table 4.1.1-1 Environments for Ground Transportation and Handling Vibration, Acceleration, and Shock. [TBD]

4.1.2 Thermal, Pressure, and Relative Humidity Environment

Flight hardware must survive and operate in nominal ground and transportation environmental conditions. Standard thermal, pressure, and relative humidity ranges are given in Table 4.1.2-1.

Requirement: Flight hardware **shall** be designed to survive without degradation the thermal, pressure, and relative humidity environments specified in Table 4.1.2-1.

Requirement: Flight hardware **shall** be designed to operate in the thermal, pressure, and relative humidity environments specified in Table 4.1.2-1., if they need to operate in those environments.

Requirement: If flight hardware would be damaged by the thermal, pressure, and relative humidity environments in Table 4.1.2-1., then special environmental protective devices **shall** be necessary.

Table 4.1.2-1 Environments for Handling, Transportation, and Storage.

Control Parameter	Low Limit	High Limit
Air Temperature (Storage)	+5°C (1)	+50°C (1)
Air Temperature (Operational)	+5°C	+40°C (2)
Temperature Change Rate	-10°C/hr [TBR]	+10°C/hr [TBR]
Pressure (10,000’ max altitude)	6.9 x 10^4 N/m^2 (520 Torr)	1 x 10^5 N/m^2 (760 Torr)
Relative Humidity	30% (3)	70% (3)

NOTES:
1) Limits could be as wide as -40°C to +70°C during shipping or storage if the environment is not controlled (such as the cargo bay of an aircraft or outside in the direct sun). Provisions should be made to limit the temperature to those in Table 4.1.2-1.
2) If the hardware is operating in an environment that is within 10°C of this limit, the hardware should be monitored to ensure that its Flight Acceptance temperature is not exceeded.
3) Relative humidity could be as low as 0% during shipping or storage or as high as 100% in uncontrolled containers. Provisions should be made to limit the relative humidity to those in Table 4.1.2-1.

4.2 Launch Pressure Change Environments

4.2.1 Venting

Requirement: All flight hardware **shall** be designed to survive without degradation a depressurization rate of -4.4kPa/s (-0.638 psi/s) during launch plus an additional 1.5x margin.

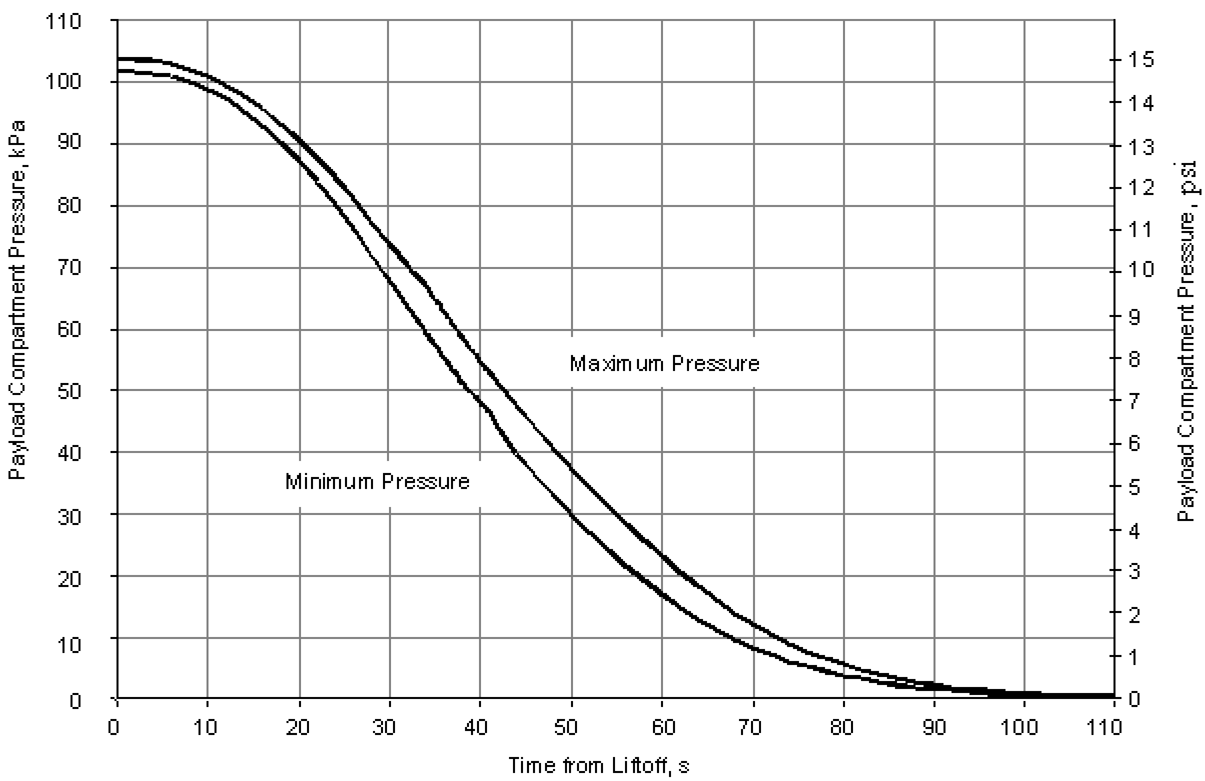


Figure 4.2.1-1 Typical Static Pressure Profiles inside EELV 5-meter fairing.

Requirement: Vent paths **shall** be directed away from sensitive surfaces of the instruments and deployed solar panels.

4.2.2 RF and HV Breakdown During Launch and in Flight

Radio frequency (RF) and high voltage (HV) circuitry in flight hardware is subject to multipacting/arcing damage at critical pressures.

Requirement: Flight hardware that operates during launch or other partial pressure conditions **shall** be designed to prevent corona, or any other forms of high voltage breakdown at pressures between 50 and $5x10^{-4}$ Torr.

Requirement: All Microwave and RF components subjected to high RF power levels (>1 Watt) **shall** demonstrate adequate margins to multipaction and/or RF breakdown via either test (>6dB) or analysis (>10 dB).

4.3 Structural Loads and Dynamics Environments

The structural loads and dynamics environments for the Europa Lander Project are in a preliminary state. As the design for both Europa Lander and launch vehicle hardware matures, these environments will become more clearly defined.

Dynamics environments for the Delta IV Heavy and preliminary environments for the SLS are publically available in each vehicle's respective Users Guides. Final environments for Europa Lander are expected to be in-family with, but not necessarily enveloped by, available Delta IV Heavy or SLS environments.

Dynamics environments and structural loads are induced in the flight system and assemblies during ground handling, transportation, launch, cruise, deceleration, descent, landing, and surface operation phases of the mission.

Requirement-Policy: The flight system, assemblies, subsystems, and instruments **shall** be tested for dynamic environments in their relevant flight-like operational mode and mechanical configuration per the Europa Lander Test and Analysis Matrix (TAM) (JPL D-xxxxx **TBD**).

Table 4.3-1 provides a summary of Europa Lander mission events with associated dynamic environment components. Dynamic environments for assemblies and instruments will be defined as the launch vehicle, flight system, and instrument suite matures.

Table 4.3-1 Dynamics Environments Summary [TBR]

	Event	Flight Hardware Configuration	Environment(s)
1	Launch and Cruise Vehicle Separation	Cruise Vehicle	Acoustics Quasi-static Loads Random Vibration Sinusoidal Vibration Pyro-Shock
4	Deorbit Vehicle Separation	Carrier and Relay Stage + Deorbit Vehicle	Pyro-Shock
5	Deorbit Vehicle Deceleration	Deorbit Vehicle	Quasi-static loads
6	Powered Descent Vehicle Separation	Deorbit Vehicle + Powered Descent Vehicle	Pyro-Shock Quasi-static loads
7	Powered Descent Vehicle Descent + Lander Separation	Powered Descent Vehicle	Quasi-static loads Sinusoidal Vibration Pyro-Shock
8	Lander Touch Down	Lander	Half-Sine Shock Pyro-shock
9	Lander Mechanical Deployments	Lander	Pyro-Shock
10	Lander Operations	Lander	Microphonics

4.3.1 Structural Loads

Quasi-static structural design loads represent the combined quasi-steady accelerations and the low frequency mechanically transmitted dynamic accelerations occurring during launch, deorbit, and descent.

4.3.1.1 Launch Structural Loads

Launch Structural Loads apply to the Cruise Vehicle, which includes all of the Europa Lander constituent stages. Generally, the most conservative and earliest available design launch loads are from the Mass Acceleration Curve (MAC) defined in Figure 4.3.3-1 [**TBR**].

Requirement-Policy: The Mass Acceleration Curve in Figure 4.3.3-1 **shall** be used as preliminary design curve for all appendage structures (including primary structures other than the flight system core), secondary structures, support structures for equipment, and equipment structural attachments and housings.

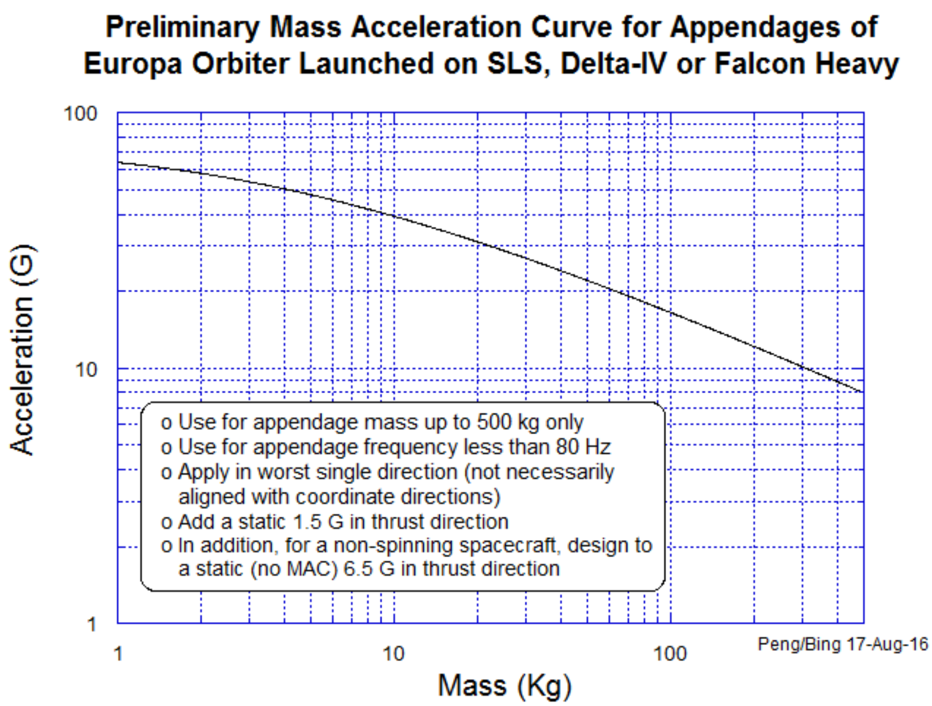


Figure 4.3.3-1 Preliminary Mass Acceleration Curve (MAC) for the Europa Lander Mission (SLS, Delta-IV Heavy, and Falcon Heavy). [TBR]

The MAC represents an upper bound on the dynamic portion of acceleration as a function of mass for physical masses less than 500 kg. This physical mass is the actual mass for single degree of freedom systems. For multi-degree of freedom systems, the physical mass can be approximated as the portion of mass supported by the element being analyzed. The dynamic acceleration is applied in the single direction producing the greatest load component (axial load, bending moment, reaction component, stress level, etc.) being investigated. The load component from the accompanying static acceleration in the launch vehicle thrust directions should be added in addition.

The loads derived from the MAC should be treated as load limits. The MAC loads are considered preliminary until validated by a coupled loads analysis (CLA).

The flight system center of gravity (CG) limit load factors, shown in Table 4.3.1-1, are provided by the launch vehicle organization and are appropriate for sizing the primary structure. Loads are applicable at the flight system CG and should be multiplied by the appropriate safety factors to obtain structural design loads. They are intended to provide a preliminary design envelope and are superseded by Augmented Coupled Loads.

Table 4.3.1-1 Preliminary Flight System CG Limit Load Factors During Launch. [TBR]

Load Condition	Max. Lateral Case	Max. Axial Case
Thrust Axis	**TBD**	**TBD**
Lateral Axes	**TBD**	**TBD**

4.3.1.2 Deorbit, Descent, and Landing Structural Loads

The design of Deorbit, Descent, and Landing (DDL) as well as Deorbit Vehicle (DOV) are in the early stages. Therefore, the predicted structural loads experienced during DDL are yet to be determined. It is possible that the DDL structural loads will exceed those experienced during launch. This document will be updated as designs are completed and limit loads are calculated.

Deorbit structural loads apply to the Deorbit Vehicle (DOV) and all of its constituent stages. These loads result from deceleration by use of a solid rocket motor. Table 4.3.1-2 provides the predicted limit load factors during deorbit.

Table 4.3.1-2 Deorbit Vehicle CG Limit Load Factors During Deorbit [TBD]

Load Condition	Max. Lateral Case	Max. Axial Case
Thrust Axis	**TBD**	**TBD**
Lateral Axes	**TBD**	**TBD**

Descent structural loads apply to the Powered Descent Vehicle (PDV) and its constituent stages. These loads are a result of the descent and thrust vector control engines firing. These motors may be pulsed, which would also induce a sine vibration environment (See Section 4.3.2.1.2). Table 4.3.2-3 provides the predicted limit load factors during deorbit.

Table 4.3.1-3 Powered Descent Vehicle CG Limit Load Factors During Descent [TBD]

Load Condition	Max. Lateral Case	Max. Axial Case
Thrust Axis	**TBD**	**TBD**
Lateral Axes	**TBD**	**TBD**

Landing structural loads apply to the Lander. These loads are a result of the Lander touchdown on Europa's surface. Table 4.3.1-4 provides the predicted limit load factors during deorbit.

Table 4.3.1-4 Lander CG Limit Load Factors During Touchdown [TBD]

Load Condition	Max. Lateral Case	Max. Axial Case
Thrust Axis	**TBD**	**TBD**
Lateral Axes	**TBD**	**TBD**

4.3.2 Random and Sinusoidal Vibration

4.3.2.1 Europa Lander System Level

4.3.2.1.1 Launch Random and Sinusoidal Vibration Environment

The flight system will experience random, periodic, and transient vibration mechanically transmitted from the launch vehicle. These environments are specified as random and sinusoidal vibration test requirements with the acceleration at the flight system adapter base interface defined in Table 4.3.2-1 and Table 4.3.2-2. The objective of the flight system random and sinusoidal vibration test is workmanship verification and qualification of the assembled flight system, interconnections, and electromechanical equipment.

At the time of launch, the flight system is in the Cruise Vehicle configuration.

Requirement: The flight system **shall** be capable of operating within specification after being subjected to the specified random vibration test levels defined in Table 4.3.2-1.

Table 4.3.2-1 Flight System Random Vibration Test Levels. [Preliminary, TBR]

Frequency Hz	FA Acceleration Spectral Density	Qual/PF Acceleration Spectral Density
5– 10	**TBD**	**TBD**
10– 200	**TBD**	**TBD**
Overall	**TBD**	**TBD**

Qual: 2 minutes in each of the three orthogonal axes, one of which is the launch thrust axis.
PF/FA: 1 minute in each of the three orthogonal axes, one of which is the launch thrust axis.

Requirement: The flight system **shall** be capable of operating within specification after being subjected to the specified sine vibration test levels defined in Table 4.3.2-2.

Table 4.3.2-2 Flight System Sinusoidal Vibration Test Levels. [Placeholder, TBD]

1 g = standard acceleration due to gravity = 9.81 m/s^2.
Qual Test sweep rate: 2 octave/minute in each of three orthogonal axes, one of which is the launch thrust axis.
PF/FA Test sweep rate: 4 octave/minute in each of three orthogonal axes, one of which is the launch thrust axis.
D.A. = Double Amplitude.

Requirement-Policy: The flight system random and sinusoidal vibration test **shall** be force-limited to reduce over-testing at hard mounted resonance frequencies.

The upper bound force spectrum in Table 4.3.2-3 and Table 4.3.2-4 may be used to limit the input acceleration to the flight system. Additional notching of the random and sinusoidal vibration input levels at flight system resonances may be required during testing and should be based on the results of the CLA multiplied by 1.2. The force and acceleration limit values may be modified based on information gathered during shaker testing. The random and sinusoidal vibration levels Table 4.3.2-1 and Table 4.3.2-2 are not intended for use in the design of the flight system primary structure, or for the structural integrity of equipment supports.

Table 4.3.2-3 Flight System Random Vibration Test Force Limit Specification.

Frequency, Hz	**Force Spectral Density Level**
$f < f_o$	$S_{FF} = C^2 * M_o^2 * S_{AA}$
$f \geq f_o$	$S_{FF} = C^2 * M_o^2 * S_{AA} * (f_o/f)^2$

[Note: f is frequency, f_o is the last predominant frequency in the axis of testing, S_{FF} is the force spectral density, C is a dimensionless constant which depends on the configuration, M_o is the total mass of the test item, and S_{AA} is the acceleration spectral density (given in Table 4.3-6 for FA and Qual/PF). The value of C^2 will be derived using the methodology of NASA-HDBK-7004C. Contact the project dynamicists for assistance with determining an appropriate value of C. The equations in **Error! Reference source not found.** are to be in consistent units.]

Table 4.3.2-4 Flight System Sinusoidal Vibration Test Force Limit Specification

Frequency, Hz	Force Spectral Density Level
$f < f_o$	$F = C * M_o * A(f)$
$f \geq f_o$	$F = C * M_o * (f_o/f) * A(f)$

Note: f is frequency, f_o is the last predominant frequency in the axis of testing, F is the force limit, C is a dimensionless constant which depends on the configuration, M_o is the total mass of the test item, and A is the input acceleration (given in Table 4.3-7 for FA and Qual/PF). The value of C will be derived using the methodology of NASA-HDBK-7004C. Contact the project dynamicists for assistance with determining an appropriate value of C. The equations in **Error! Reference source not found.** are to be in consistent units.

4.3.2.1.2 Descent Thruster Pulsing Environment [TBR]

The Descent Vehicle may be subjected to vibration loads from thruster pulsing during the descent phase. For this draft of the ERD, the requirements are left in skeleton form information purposes. Future versions of the Europa Lander ERD will be updated as the design matures.

Vibration induced during powered descent is characterized by low-level sinusoids at a [**TBD**] thruster pulse frequency and at higher-order harmonics.

During descent, the flight system is comprised of the Carrier and Relay Stage and the Powered Descent Vehicle. The Powered Descent Vehicle and the Descent Stage post separation would encounter the descent thruster pulsing environment.

Requirement: The flight system **shall** be designed to survive and function as required during and after exposure to the bounding thruster sine levels shown in Table 4.3.2-5

Table 4.3.2-5 Descent Vehicle Sine Vibration Levels [TBD]

Axes	Frequency (Hz)	Dwell Duration (s)	FA (g)	PF (g)
X	**TBD**	**TBD**	**TBD**	**TBD**
Y	**TBD**	**TBD**	**TBD**	**TBD**
Z	**TBD**	**TBD**	**TBD**	**TBD**

4.3.2.2 Assembly Random Vibration [TBD]

Requirement: Flight hardware and spares **shall** be designed to survive without degradation when subjected to the application of the specified random vibration environment defined in the appropriate section of Table 4.3.2-6 [Table will be provided at a later date as the design matures.]

Table 4.3.2-6 Assembly Random Vibration Specifications [TBD].

The General Environmental Verification Specification (GEVS) (GFSC-STD-7000A) provides a generalized assembly-level random vibration specification. (See Table 2.4-3 in GFSC-STC-7000A.) Until further updates on are available, for components with mass >50 kg, the general specifications provided in GEVS are suggested as early guidance only. Final requirements for Europa Lander assemblies will be released in updates to this document.

4.3.2.3 Assembly Quasi- Static Launch Loads

Requirement: In the event that launch loads are not achieved during vibration testing, those loads **shall** be produced by the performance of a sinusoidal burst test, static pull, centrifuge test, or another static loads test.

4.3.3 Acoustics Environment

4.3.3.1 Launch Acoustic Environment

The acoustic environment (Table 4.3.3-1) is the envelope of the acoustic environments for the alternate candidate EELV launch vehicles (Falcon Heavy and Delta-IV Heavy) and the current acoustic prediction for the 5 m fairing on the Space Launch System (SLS) launch vehicle. The bounding case is the envelope of the Delta-IV H with composite PLF (payload fairing) and Delta-IV H with iso grid PLF and the current acoustic prediction for the 5 m SLS fairing. This document will be updated when reliable acoustic levels for the baseline SLS, or other potential alternate launch vehicles, is available. The maximum acoustic environment for the Europa Lander flight system occurs during lift-off and transonic flight. The environment is represented as a diffuse acoustic field with random incidence specified in 1/3-octave bands.

4.3.3.1.1 Flight System Acoustics

At the time of launch, the flight system is in the Cruise Vehicle configuration.

Requirement: The flight system **shall** perform within specification after being subjected to acoustic test levels defined in Table 4.3.3-1

Table 4.3.3-1 Acoustic Qual/Protoflight and Flight Acceptance Test Levels (Placeholder – TBD)
(Duration: Qual: 2 minutes; PF and FA: 1 minute.)

4.3.3.1.2 Assembly Acoustics

Requirement: Flight hardware and spares shall be designed to survive without degradation, when subjected to the application of the specified acoustic test environment in Table 4.3.3-2

Table 4.3.3-2 Assembly Level Acoustic Qual/Protoflight and Flight Acceptance Test Levels [TBD].

Note: Assemblies with a high surface area to mass ratio may see higher levels than the Flight System.

4.3.4 Pyrotechnic Shock

Pyrotechnic shock testing will occur at the Europa Lander flight system and assembly level. Flight system testing is intended to validate the capability of the flight system and verify the expected dominant shock sources for potentially susceptible hardware. Any assemblies that might be susceptible to shock dynamics are tested at a lower-level of assembly.

4.3.4.1 Flight System Shock Environments

The system level separation/deployment pyrotechnic shock tests will be employed to verify the adequacy of the assembly Qual/Protoflight pyrotechnic shock test environments, which will be provided in an update to the ERD. For assembly level shock testing, the MEFL will be multiplied by a factor of 1.4 for Qual/PF.

Requirement: The flight system **shall** be designed to survive without degradation and to function safely when subjected to the induced shock environments during separation event with a Maximum Expected Flight Level (MEFL) shown in Table 4.3.4-1. These levels are predicted at about 6-inch from the source.

For reference, Table 4.3.4-1 contains maximum predicted shock levels for Launch Vehicle Separation at the flight system interface due to payload separation from the PAF for the alternate candidate EELV launch vehicles (Atlas-V and Delta-IV Heavy). The ERD will be updated when reliable payload separation shock levels for the baseline SLS, or other potential alternate launch vehicles, is available.

Table 4.3.4-1 Maximum Expected Flight Level Pyroshock Environment at Major System-Level Events. [Preliminary, TBR]

Event: Launch Vehicle Separation	
Interface: Cruise Vehicle – Launch Vehicle	
Frequency, Hz	**MEFL SRS (Q=10)**
100	100 g
1,100	5050 g
10,000	5050 g
Event: Deorbit Vehicle Separation	
Interface: Carrier and Relay Stage – Deorbit Vehicle	
Frequency, Hz	**MEFL SRS (Q=10)**
100	40 g
1,100	4000 g
10,000	4000 g
Event: Descent Vehicle Separation	
Interface: Deorbit Stage – Descent Vehicle	
Frequency, Hz	**MEFL SRS (Q=10)**
100	40 g
1,100	4000 g
10,000	4000 g
Event: Lander Separation	
Interface: Descent Stage – Lander	
Frequency, Hz	**MEFL SRS (Q=10)**
100	40 g
1,100	4000 g
10,000	4000 g
Event: Lander Touchdown	
Interface: Lander-Europa Surface	
TBD	

[Note: Launch vehicle/flight system interface pyroshock environment updated to envelope the separation system type D1666 and shock environments for the candidate launch vehicles.]

4.3.4.2 Assembly Level Shock Environments

The pyroshock levels defined this section will be based on a reasonable envelope of the possible pyroshock conditions that will be encountered during launch, flight, DDL and Lander deployment events. The requirements are based on test information available to date and may be updated as the spacecraft configuration matures.

Requirement: Flight hardware **shall** be designed to survive without degradation, when subjected to the application of the specified pyrotechnic shock environment defined in Table 4.3.4-2 and Table 4.3.4-3

Designations provided in Table 4.3.4-2 and Table 4.3.4-3 are for demonstration purposes only. Designations and values will be provided in a later update.

Table 4.3.4-2 Assembly Pyrotechnic Shock Zones. [TBD]

Assembly Location/Assembly	Zone	Comments
Carrier and Relay Stage (CRS)		
CRS Assembly 1	a	
CRS Assembly 2	a	
Deorbit Stage (DoS)		
DoS Assembly 1	b	
DoS Assembly 2	b	
Descent Stage (DS)		
DS Assembly 1	c	
DS Assembly 2	c	
Lander		
Instrument 1	d	
Instrument 2	d	

Table 4.3.4-2 Assembly Pyrotechnic Shock Requirements by Zone. [TBD]

Zone	Frequency, Hz	QUAL, PF Peak SRS Response (Q=10)
	100	**TBD** g
a	100 - 1,600	**TBD** dB / Oct.
	1,600 - 10,000	**TBD** g
	100	**TBD** g
b	100 - 1,600	**TBD** dB / Oct.
	1,600 - 10,000	**TBD** g
	100	**TBD** g
c	100 - 1,600	**TBD** dB / Oct.
	1,600 - 10,000	**TBD** g
	100	**TBD** g
d	100 - 1,600	**TBD** dB / Oct.
	1,600 - 10,000	**TBD** g

4.3.4.3 Pyrotechnic Testing Requirements

System/subsystem level tests will consist of two actual firings of each pyrotechnic device that is to be used for the Europa Lander mission separation and deployment events. The system level tests will be employed to verify the adequacy of the assembly Qual/Protoflight pyrotechnic shock environments found in Table 4.3.4-2 and Table 4.3.4-3, less a factor of 1.4 or 3 dB.

Pyrotechnic shock testing is required for assemblies exposed to pyrotechnic shock loading, whether the loading is self-generated or induced by external sources.

Requirement: Assemblies **shall** be subjected to a synthesized shock twice for Qualification and once for Protoflight in each of three orthogonal directions (some pyroshock simulation facilities may be capable of inputting required shock levels in more than one axis at a time).

Flight Acceptance pyroshock testing at the assembly level is not required. However, each assembly will be further subjected to pyrotechnic shock during spacecraft system and/or subsystem level testing.

Requirement: The test article **shall** be mounted to the test fixture at its normal flight interfaces and shall be in its flight configuration at the time of the flight pyro shock event. Test shocks shall be applied at assembly mounting points.

The pyroshock test may be conducted either a) using an electro-dynamic shaker or b) using a shock-generating apparatus. The shaker shock test may be conducted in conjunction with the random vibration test, but there are some operating limitations (e.g., maximum acceleration levels and severe roll-off at frequency above 3000 Hz).

Requirement: Synthesized shock waveforms **shall** meet the following criteria: the time history shall be oscillatory in nature, and the pulse shall decay to less than 10% of its peak value within 20 milliseconds.

Requirement-Policy: A single point open loop control **shall** be utilized using lower level 'spectrum shaping' runs to calibrate the test control.

Requirement-Policy: Control and monitor accelerometers **shall** be mounted on the test fixture near the test article attachment point.

Requirement-Policy: The control shock spectrum at the control accelerometer **shall** be matched to the required spectrum.

Requirement-Policy: Time history data from control and any monitoring accelerometers **shall** be recorded and preserved.

Requirement: Since test margin is unachievable, robustness to self-induced shock environments **shall** be verified through a minimum of (2) actual firings of pyrotechnic devices in a configuration which is representative of flight

The assembly mechanical/pyroshock test levels will be provided at a later date.

Requirement: Flight hardware that performs critical operations during a shock-producing event **shall** be designed to function within specification during the application of the specified pyrotechnic shock environment.

For more information on pyrotechnic shock testing consult NASA-STD-7003A.

4.3.5 Induced Microphonics and Jitter Effects

Low-level dynamic environments would occur during post-separation operations. The principal sources of these environments are flight system deployments, nominal articulation of solar arrays, attitude control maneuvers, nominal main engine burn as well as other mechanical system operations, including reaction wheels, and sensors and instruments with moving masses. These low-level vibrations may induce microphonics or jitter effects in science instruments or flight system hardware. (Note: Microphonics is the inducement of noise in electrical devices and jitter is the smearing of images in optical systems caused by vibration-induced motions).

Lander instruments and flight subsystems will be subject to microphonics and jitter effects from articulation of stabilizers, telecom antennae, instruments subsystems inside and outside the vault, and sampling system. These environments are not yet defined and will be included in updates to this document. The values presented here are representative and may be revised as the Lander design matures.

Requirement: Instruments **shall** be designed to operate within specification during broadband base dynamic input per Table 4.3.5-1.

Requirement: Verification of the Microphonic Environment **shall** be by test in three orthogonal axes, one axis at a time.

Table 4.3.5-1 Broadband Microphonic Environment for Instruments. [TBD]

Frequency (Hz)	Requirement
1-20	[**TBD**] dB/octave
20-300	[**TBD**] g^2/Hz
300-1000	[**TBD**] dB/octave
Overall	[**TBD**] g_{rms}

Requirement: Instruments and subsystems **shall** be designed to operate during sinusoidal base inputs per Table 4.3.5-2.

Table 4.3.5-2 Sinusoidal Microphonic Environment for Instruments. [TBD]

Frequency (Hz)	Requirement
10-1000	[**TBD**] g, 0-to-Peak

1 g = standard acceleration due to gravity = 9.81 m/s^2.
Test sweep rate: 2 octave/minute (upsweep only) in each of three orthogonal directions.

Requirement: Instruments and subsystems that are required to operate during post boost, cruise, tour, DDL, and science segments of the mission **shall** be designed not to propagate low-level vibration environments (resulting in microphonics/jitter) to susceptible operating flight hardware.

Requirement: Instruments **shall** be designed to impart forces to the flight system no greater than those specified in Table 4.3.5-3 below.

Table 4.3.5-3 Maximum Instrument Microphonic/Jitter Emission. [TBD]

Frequency (Hz)	Requirement
1-1000 Hz sinusoidal	[**TBD**] lb, 0-to-Peak
Peak transient	[**TBD**] lb Peak
Broadband	[**TBD**] lb rms

Force measured on rigid mount using appropriate instrumentation.
Instruments exhibiting higher measured forces may be subject to operational constraints.

4.3.6 Flight System Modal Test

The purpose of modal testing or analysis is to characterize the fundamental dynamic characteristics of a structure's:

- Natural frequencies
- Damping
- Mode Shapes

Modal testing removes uncertainties regarding joint stiffness and structure damping. This information will be used to update the finite element model and will result in an improved coupled loads analysis.

4.3.7 Dynamics Test Tolerances

Requirement: The dynamics tests **shall** be controlled to the tolerances in Table 4.3.7-1.

Table 4.3.7-1 Dynamics Test Tolerances.

Static loads (Centrifuge or Shaker)	Within +/- 5% of the specified value.
Random vibration spectral shape	Within +/- 3 dB of the power spectral density test spectra, measured in frequency bands no more than 25 Hz wide.
Random vibration wide-band (RMS) level	Within +/- 1.0 dB (true RMS) of the specified level.
Acoustic test spectral shape	Equal to the sound pressure level tolerances of 4.3.3-1 measured in fixed 1/3-octave bands.
Acoustic test overall level	Within +/- 1.0 dB (true RMS) of the specified level.
Frequency	Within +/- 5% or +/- 1 Hz, whichever is greater.
Time	Within +/- 5%.
Shock response spectrum (SRS) shape	1) SRS measured with a minimum resolution of one-sixth (1/6) octave frequency band. SRS spectrum magnitudes within +/- 6 dB, with at least 50% of the spectrum magnitudes exceeding the nominal test specification. 2) Time history oscillatory in nature and decays to 10% of peak value within 20 milliseconds. 3) Mechanical impact shock-generating apparatus is recommended for pyroshock simulation. Other shock-generating apparatus, such as shakers, ordnance, and drop tables, generally do not meet the tolerance criteria above and their use is discouraged.

4.4 Thermal Environments

4.4.1 Definitions

Terms used herein for thermal design and test are defined as follows:

Operating Allowable Flight Temperature (AFT):
For specified assemblies and subsystems, the Operating AFT range includes the worst case nominal (i.e., non-emergency) hot and cold temperature limits, including allowances for prediction uncertainties. These limits encompass all nominal operating modes, performance within functional specifications, that the Thermal Control Subsystem is designed to accommodate. Temperatures are measured at the thermal control surface (e.g. mounting surface, radiator surface, etc.), as specified by Thermal Engineering.

Non-Operating Allowable Flight Temperature:
For specified assemblies and subsystems, the Non-Operating AFT range includes the worst case powered-off hot and cold mission temperature limits, including allowances for prediction uncertainties. These limits encompass all nominal (i.e., non-emergency) non-operating modes that the Thermal Control Subsystem is designed to accommodate. Temperatures are measured at the thermal control surface (e.g. mounting surface, radiator surface, etc.), as specified by Thermal Engineering.

Qualification/Protoflight (PF) Temperature Limits (Operating and Non-Operating):
Protoflight thermal test magnitude and duration are identical to qualification test magnitude and duration. Operating Protoflight testing implies meeting all functional specifications in the PF operating environments.

Flight Acceptance (FA) Temperature Limits (Operating and Non-Operating):
FA is the temperature range over which flight assemblies whose design has been previously qualified will be tested to verify workmanship and functionality within specification. FA testing serves to demonstrate a moderate degree of margin beyond the AFT range.

Design Temperatures:

Design temperatures are the temperature limits to which assemblies are designed to meet functional and performance specifications. Design temperatures are normally equivalent to or exceed the Qualification/ Protoflight limits.

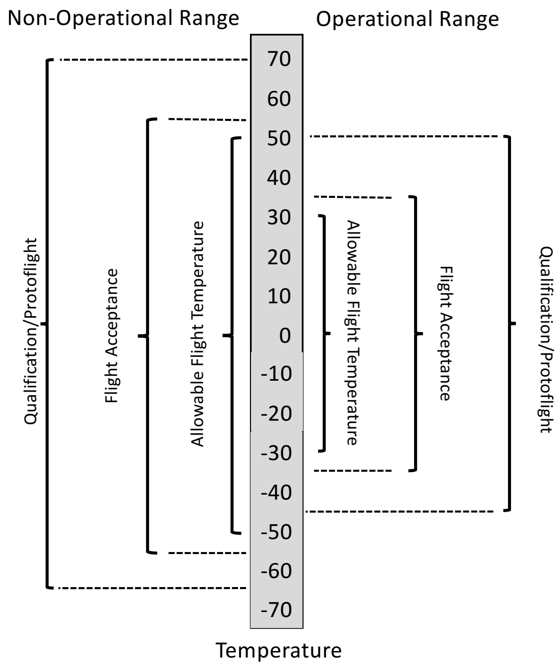


Figure 4.4.1-1 Example Thermal Design Limits for Operational and Non-Operational Ranges. Design limits are normally equivalent to or exceed the Qualification/Protoflight limits.

4.4.2 Launch Thermal Environment

The launch thermal environment is dependent on both the launch vehicle and the flight trajectory. The values presented in this section are preliminary and may change. This document will be updated as the design matures.

Requirement: The thermal control design **shall** maintain assemblies within their respective AFT limits while exposed to the launch induced thermal environments in Section 4.4.2.1 and Section **Error! Reference source not found.**.

4.4.2.1 Payload Fairing Wall Temperature

The inner surfaces of the EELV composite 5-m PLF cone and cylinder have an emittance of 0.9. The peak heat flux radiated by the inner surfaces of the LV cone and cylinder of the 5-m PLF is less than 914 W/m^2 (290 Btu/hr-ft^2), and peak temperatures remain below 93°C (200°F) at the warmest location. SLS fairing thermal conditions are expected to be similar to EELV environments.

4.4.3 Planetary Protection Thermal Environment

The Europa Lander Planetary Protection Plan (JPL D-97653) is the reference document for planetary protection requirements. One of the planetary protection processes available is Dry Heat Microbial Reduction (DHMR) and therefore would likely drive a non-operational thermal environment for flight hardware. Here, a short description of DHMR and notional specifications are given.

DHMR Description: Dry Heat Microbial Reduction (DHMR) is a bake-out process where the hardware is held at an elevated temperature, typically > 125°C for many hours in a controlled humidity (<25% relative humidity) environment, such as partial vacuum or dry nitrogen. Exact temperatures and durations for the hardware would be dependent on accommodation of the hardware on the spacecraft and would be investigated in detail when thermal models become available. Steps should be taken after bioburden reduction processing to prevent recontamination, potentially involving specialized handling procedures, seals, covers, filters and/or other techniques incorporated in the design. Recently, NASA specifications have been changed to allow HMR without humidity control. While simpler to implement, the durations are longer for equivalent microbial lethality.

For surface and encapsulated elements, a 4-order of magnitude bioburden reduction equates to a DHMR bake at slightly higher than 125˚C for 88.6 hours and 442.9 hours, respectively.

4.4.4 Space and Europa Surface Thermal Environment

The space thermal environment includes all phases of the Europa Lander mission following launch and including surface operations. Table 4.4.4-1 contains external heat sources for all mission phases.

Requirement: The thermal control design **shall** maintain assemblies within their respective AFT limits when subjected to the worst-case external mission environments as specified in Table 4.4.4-1

Table 4.4.4-1 External Heat Sources/Sinks and Constants [TBR]

	Minimum	**Maximum**
Solar Flux [W/m^2][1]	46.2 (at 5.43 AU)	1737.4 (at 0.885 AU)
Earth IR [W/m^2]	227	241
Earth Albedo	0.29	0.31
Jupiter IR [W/m^2][2]	13.6	13.6
Jupiter Albedo	0.311	0.375
Europa IR [W/m^2]	1 [**TBR**]	15 [**TBR**]
Europa Geometric Albedo[3]	0.64 [**TBR**]	0.7 [**TBR**]
Europa Surface Temperature [K][4]	70	132
Europa Surface Pressure [bar]	10^{-12}	10^{-12}
Jupiter Eclipse [hours]	9.2	9.2
Space Temperature [K]	2.7	2.7

Note1: Solar Flux is calculated using a solar constant of 1360.8 W/m2 at 1 AU given by *Kopp and Lean*, [2011] doi:10.1029/2010GL045777
Note 2: IR flux is given at the thermal boundary. Jupiter IR flux at Europa is approximately 0.15 W/m^2.
Note 3: Geometric Albedo of Europa is given. Localized albedo values as a function of wavelength may vary. Future updates to this document will include local albedo values.
Note 4: Minimum and maximum Europa surface temperature limits are the extreme macroscopic observed temperatures from Galileo PPR. Localized surface temperatures may be more extreme. Possible extreme local temperature are 29 K and 190 K.

The atmosphere of Europa is extremely tenuous (~10^{-12} bar). Possible heat transfer paths to the Lander will be through radiation and conduction to the surface. The average equatorial Europa surface temperature is 106 K over a diurnal cycle with ~50 K fluctuations (See Figure 4.4.4-2). One Europa diurnal cycle is 85.2 hours. Landing latitude will influence the average diurnal temperature and diurnal variation (see examples from Figure 4.4.4-2). The maximum observed brightness temperature from Galileo PPR is 132 K (equatorial local noon), and the lowest is ~70 K (high-latitude night) (See Figure 4.4.4-1). PPR resolution is much larger than the scale size of the Lander. It is possible that extreme hot and cold temperatures exist at Lander-scale. On the hot side, *Abramov and Spencer* [2008] (doi:10.1016/j.icarus.2007.11.027) suggest that any warm surface (e.g. liquid water at 273 K) would rapidly cool to < 190 K within 10 Earth days. On the cold side, *Paige et al.,* [2010] (10.1126/science.1187726)measured temperatures as low as 29 K in polar Lunar craters. While shaded Lunar craters are not a one-to-one analog, they represent a potential extreme cold that may also exist on Europa in localized areas. Therefore, we specify 29 K and 190 K as the local temperature extremes that could be found on Europa's surface.

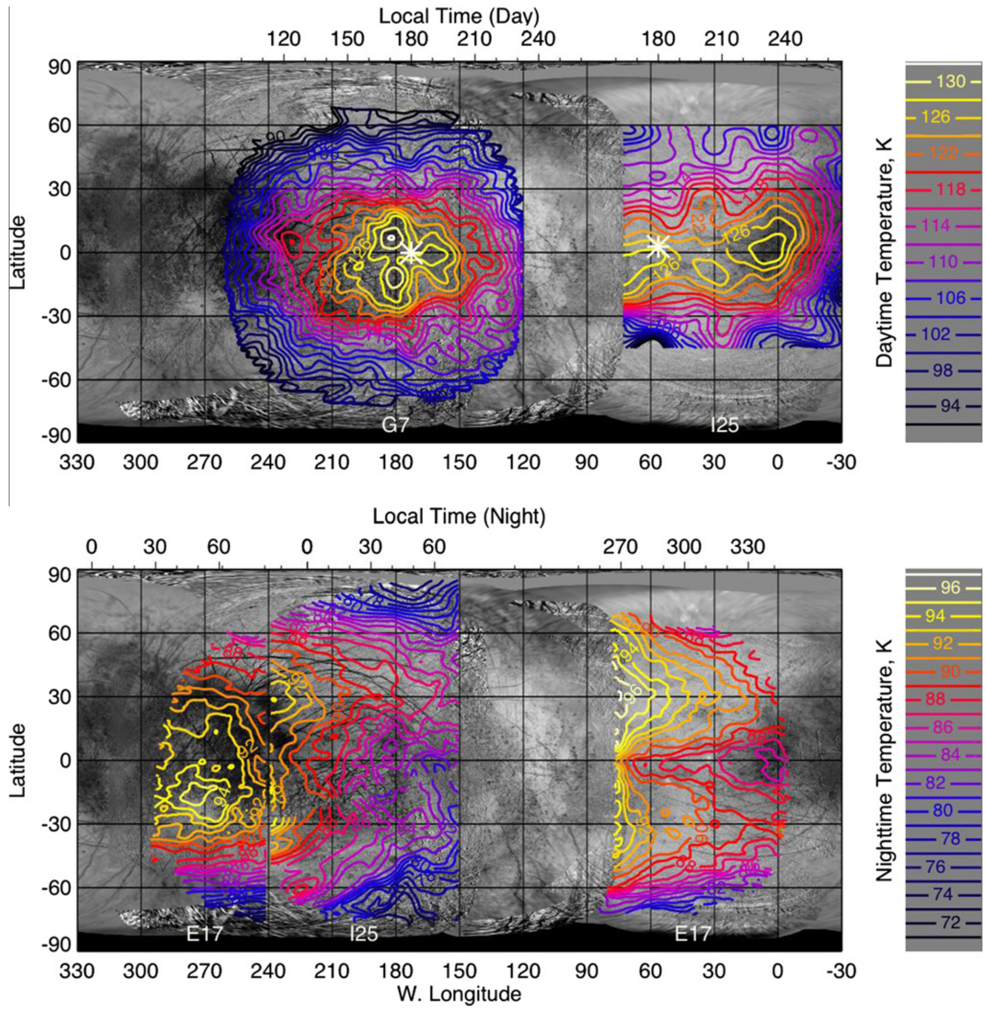


Figure 4.4.4-1 Brightness temperature observations from Galileo PPR. [From *Rathbun, Rodriguez, and Spencer*, 2009.] (doi:10.1016/j.icarus.2010.07.017)

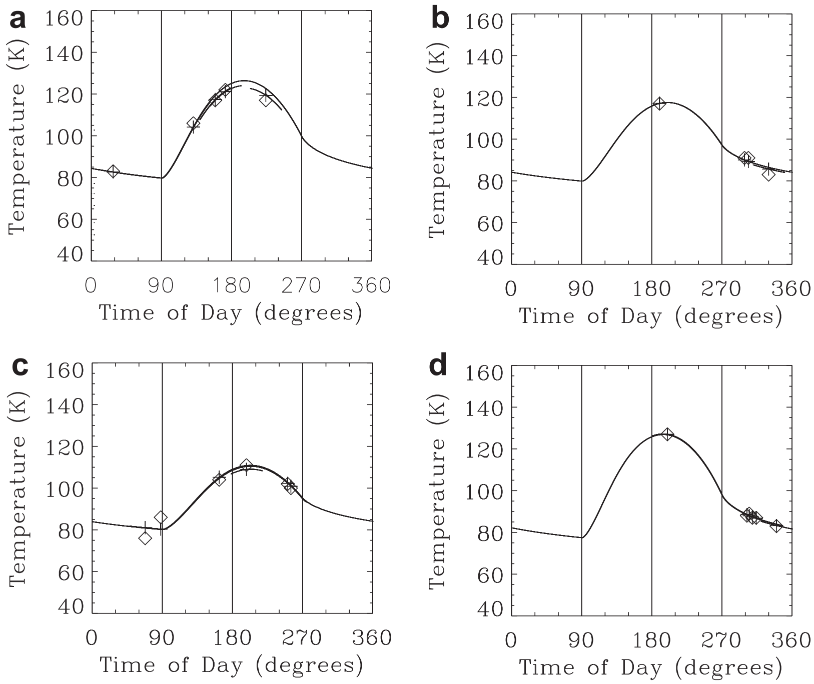


Figure 4.4.4-2 Modeled diurnal surface temperature fluctuations at four different latitudes on Europa's surface located at approximately: a) 25˚, b) 35˚, c) 45˚, and d) 5˚. [From *Rathbun, Rodriguez, and Spencer*, 2009.] (doi:10.1016/j.icarus.2010.07.017)

Europa IR radiation minimum and maximum is calculated using the measured surface minimum (70 K) and maximum (132 K) temperatures and assuming Europa is a nearly perfect blackbody with an average emissivity of 0.9 [*Spencer*, 1987] (http://hdl.handle.net/10150/184098).

4.4.4.1 Internal Heat Sources

The internal heat source for an assembly is defined as the minimum and maximum heat dissipation (thermal Watts) of the assembly for all mission-operating modes.

Requirement: Allowable Flight Temperatures (AFT's) **shall** be maintained under all modes of internal heat-dissipation.

4.4.4.2 Vacuum

Requirement: Assemblies **shall** be designed to operate within specification for a space vacuum condition of $1.0x10^{-14}$ Torr.

(Test vacuum chambers with pressure ≤ 10^{-5} Torr, will satisfy the above requirement if verified through testing.)

Requirement: Assemblies and instruments on the Lander **shall** be designed to operate within specification for Europa surface vacuum condition of $1.0x10^{-9}$ Torr [**TBR**].

(Test vacuum chambers with pressure ≤ 10^{-5} Torr, will satisfy the above requirement if verified through testing.)

4.4.5 Flight System Thermal Vacuum Test

Requirement: The flight system **shall** be designed to function within specification at the Qualification/Protoflight operating temperature extremes specified in the Temperature Requirements Table [JPL D-xxxxx **TBD**].

Requirement: The flight system **shall** be designed to survive without permanent degradation at the Qualification/Protoflight non-operating temperature extremes specified in the Temperature Requirements Table [JPL D-xxxxx **TBD**].

Requirement-Policy: The flight system-level thermal vacuum test **shall** include thermal balance and functional testing phases.

At flight system-level thermal vacuum testing, red limits will set to less than flight acceptance levels.

4.4.6 Assembly/Subsystem Thermal Design and Verification Requirements

Requirement: Flight hardware assemblies **shall** be designed and tested to perform within specification according to the parameters, levels, and margins shown in Table 2.2-1 (including start-up capability), while in vacuum, over their respective thermal test limits per the Temperature Requirements Table (TRT) (JPL D-xxxxx **TBD**).

1. Note: Exceptions are made for instrument sensor performance, for which performance degradation should be fully recoverable after operation at operational thermal test limits specified in the TRT (JPL D-xxxxx **TBD**).
2. Note: The specific assembly test temperature requirements at the assembly thermal control surface are specified in the TRT (JPL D-xxxxx **TBD**). The design and test temperature requirements described in the TRT apply for Qualification, Protoflight, and Flight Acceptance tests in both the operating and non-operating conditions. There is no further operating margin above and beyond the qualification test level for faulted or off-design conditions.

Requirement: Flight hardware **shall** be tested to Protoflight/Qualification limits specified in the TRT (JPL D-xxxxx **TBD**) as per the environmental TAM (JPL D-xxxxx **TBD**).

[The environmental Test and Analysis Matrix, TAM (table number **TBD**) specifies the type of test to be performed (ie. Qual, PF or FA) for each assembly, subsystem, instrument, or the flight system.]

Requirement: Flight hardware assemblies **shall** be designed to survive without permanent degradation after exposure, while in vacuum, to the non-operating thermal test limits specified in the TRT (JPL D-xxxxx **TBD**).

A recommended test profile is shown in Figure 4.4.6-1. Alternate test profiles may be used, as long as all the required test parameters are met.

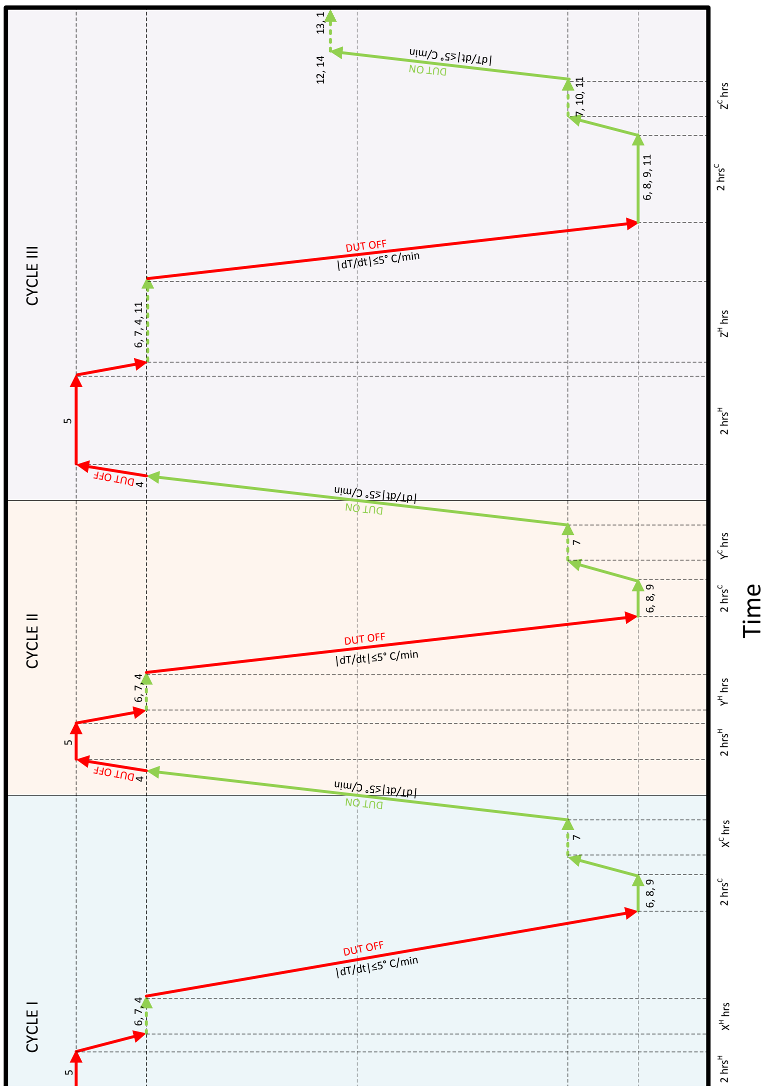


Figure 4.4.6-1 Example of a Recommended Thermal Test Profile for Assembly, Subsystem, and Instrument Testing.

Suggested Steps for Thermal Testing:

1. Full Functional Test @ 1atm
2. Pump down to 1 x 10^{-5} Torr
3. Full Functional Test @ Vacuum
4. Power Down DUT
5. Dwell @ Hot Non-Op
6. Hot/Cold Start
7. Performance Test @ Vacuum and Dwell
8. Heater test and/or cold non-op start-up (if required)

Notes for Thermal Testing

1. $X^H+Y^H+Z^H$ = 72 hours
2. $X^C+Y^C+Z^C$ = 24 hours
3. Cumulative dwell time should equal 6 total hours at hot and 6 total hours at cold non-operating temperatures.

9. Dwell @ Cold Non-Op
10. Full Performance Test @ Vacuum and Dwell
11. Accumulate Remaining Operating/Non-Operating Hours as Required
12. Temperature = Ambient + 5 C
13. Return to Ambient Pressure and Temperature
14. Full Functional Test @ Vacuum

Requirement: Assemblies, which are powered-on during launch, **shall** be operated during chamber evacuation to monitor corona effects.

Requirement: Each temperature plateau **shall** be maintained for a minimum of 2 hours, once temperature stabilization is achieved.

Requirement: Functional testing **shall** be performed a minimum of three times at the hot operational test limits and a minimum of three times at the cold operational test limits.

Ideally, a functional test would be run at each temperature plateau for a total of 6 times, minimum.

Requirement: Assemblies **shall** demonstrate a minimum of three proper start-ups at the hot and three proper start-ups at cold operational test limits and also at the cold non-operational test limit (if required*).

Exception: One-time deployable mechanisms, or launch latches and restraints, with the approval of the Project ERE and the environmental discipline specialists, and documented in the project-approved ERD, may be operated at a minimum of four operations, defined as once at ambient temperature, once at PF hot bound, once at PF cold bound, and once in system test on the flight system. ***Note:** Start-ups from the cold non-op test limit are required if the mission thermal control scheme requires powering on equipment at the cold non-operational Allowable Flight Temperature.

Special thermal design features, such as heater and louver operations, should be demonstrated if applicable. Additionally, any flight temperature sensing devices should be calibrated in this test.

4.4.7 Thermal Test Tolerances and Stabilization Criteria

Requirement: The thermal test tolerance **shall** be as Table 4.4.7-1.

Table 4.4.7-1 Thermal Test Tolerances and Stabilization Criteria.

Temperature	Within +2° C.
Time	Within 0 and +15 minutes.
Pressure	+5% above 1 Torr. At vacuum conditions, tolerances will be such that a pressure of <10^{-5} Torr is maintained.
Thermal Stability	Temperature control systems for thermal vacuum environments will have the capability of holding the control point thermocouple readings to within +2ºC.
Thermal Equilibrium	Change of temperature of largest centrally-located thermal mass is less than 2ºC/hour for three consecutive readings taken 15 minutes apart and dT/dt slope is approaching zero.

4.4.8 Thermal Cycling Design Criteria

All flight hardware will be designed to accommodate the effects of thermal cycling.

Requirement-Policy: Electronic hardware **shall** be capable of surviving three times the planned mission expected number of thermal cycles, each over the allowable flight temperature (AFT) extremes, plus an estimate of the thermal cycles expected in the planned ground operations. **(**This requirement may be verified via analysis or sample coupon test.)

Requirement-Policy: In the absence of a specific mission thermal cycling profile, electronic hardware **shall** be capable of surviving 10,000 cycles, each of a 15 C ΔT excursion.

4.5 Electromagnetic Interference and Compatibility (EMI/EMC)

The EMC design and verification program is intended to produce a Europa Lander flight system that would be electromagnetically compatible with itself and its external environment during all mission phases. This includes performance-compliant flight system operation in space and in the launch environment (launch vehicle and launch site) under all mission operation conditions. This section describes the Electromagnetic Interference / Electromagnetic Compatibility (EMI/EMC) requirements applicable to the Europa Lander flight system and its instruments, subsystems, and assemblies throughout its entire mission duration. This section will be updated in future releases as the flight system and payload designs mature.

Requirement: The flight system hardware shall be designed and verified to be electromagnetically self-compatible per Table 2.2-1.

4.5.1 Launch Vehicle/Launch Site Electromagnetic Environments

Flight system electromagnetic compatibility with the launch vehicle and the launch site (LV/LS) should be considered in two areas:

- Compatibility with the launch vehicle and launch site power system.
- Electromagnetic/RF compatibility (emissions, susceptibility, transmitters, receivers).

4.5.1.1 Flight System Power Subsystem Compatibility with the LV/LS

The flight system will be powered by the umbilical power supply during testing at the launch site facility. It is assumed that the umbilical power subsystem will provide power to the flight system, will not generate noise and transients that exceed the conducted susceptibility (Section 4.5.2.2) and power transient (Section 4.5.3) requirements in order to provide compatibility between the respective power systems. If the respective requirements are not compatible, however, it will be necessary to provide power profiles of the umbilical power supply.

It should be noted that the flight system power supply, being a subsystem, must comply with the requirements presented in this section. Further, since the power supply provides power to the instruments, subsystems, and assemblies, its output power characteristics must be compatible with the specified power requirements. The following requirements are therefore imposed in order to provide clarity in compliance.

4.5.1.1.1 Flight System Power Subsystem Conducted Emissions and Susceptibility Requirements [TBR]

To be provided in a later revision.

4.5.1.1.2 Flight System Power Supply Output Noise Requirements

To be provided in a later revision.

4.5.1.2 Flight System Electromagnetic/RF Compatibility with the LV/LS

The launch vehicle and the launch site electromagnetic environments include radiated emissions and radiated susceptibility requirements. These are listed below as RE102 for the radiated emissions requirements and RS103 for radiated susceptibility requirements. Both include a baseline limit as well as narrower notches for the RF transmitter bands. The Evolved Expendable Launch Vehicle (EELV) transmitters are limited to the S-band telemetry transmitters (for operation with the Tracking/Data Relay Satellite System and GPS Metric Tracking System). The flight system and all its components (instruments, subsystems, and assemblies) must be compatible with these environments as outlined below.

4.5.1.2.1 Flight System Radiated Emissions Requirements (RE102)

The radiated emission requirements are imposed on the flight system in order to ensure that it will not interfere with the operational and RF receiver systems installed at the launch site or on the launch vehicle.

Requirement: The flight system, while installed on the launch vehicle or operating in the launch site, **shall** not generate radiated emissions in excess of the limits shown in Table 4.5.1-1 (and plotted in Figure 4.5.1-1).

(Note: Testing to be in accordance with the RE102 test method outlined in MIL-STD-461F and using the dwell times and measurement bandwidths specified in Section 4.3.10.3 of MIL-STD-461F.)

Table 4.5.1-1 LV/LS baseline radiated emissions limits (RE 102). [TBR]

Type	HW	Frequency Range	RE102 Limit dBμV/m	Note
Baseline Limit	General	10 kHz – 18 GHz	114	6 dB margin added
UHF Command & Destruct	LV/LS	408 MHz – 430 MHz	25	6 dB margin added
GPS Receiver, L2	LV/LS	1127 MHz – 1237 MHz	30	6 dB margin added
GPS Receiver, L1	LV/LS	1565 MHz – 1585 MHz	30	6 dB margin added

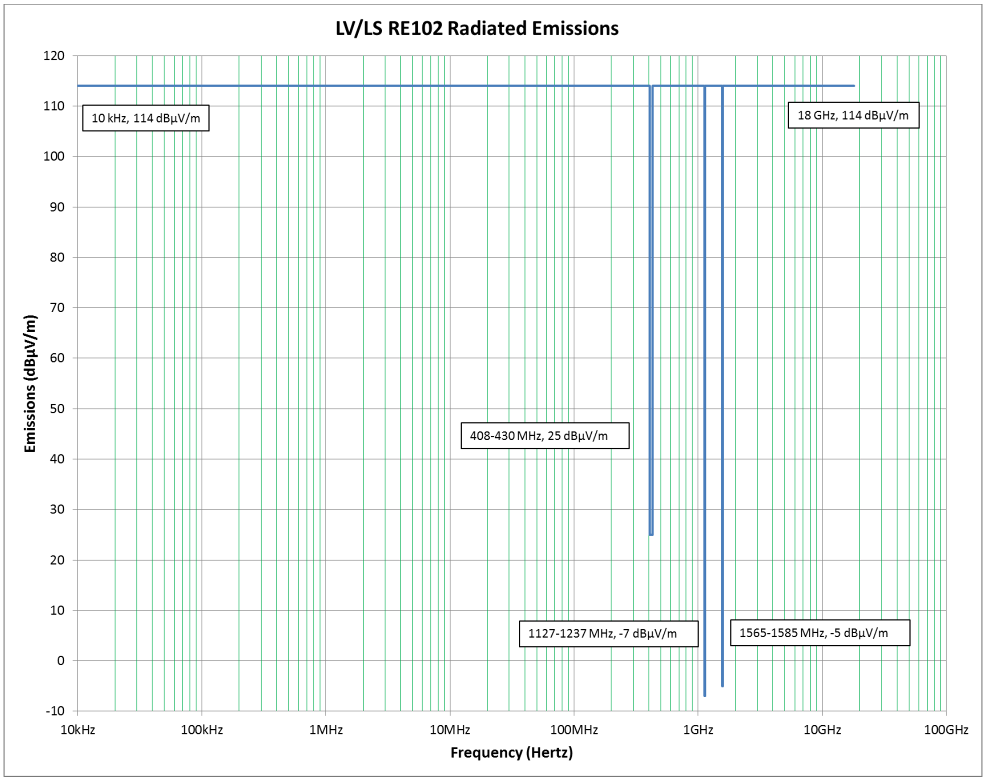


Figure 4.5.1-1 LV/LS baseline radiated emissions limits (RE 102). [TBR]

4.5.1.2.2 Flight System Radiated Susceptibility Requirements (RS103)

This requirement is imposed to ensure the flight system will not be damaged and will operate as required when installed on the launch vehicle or operating at the launch site.

Requirement: The flight system, while installed on the launch vehicle or operating in the launch site, shall meet its performance requirements after exposure to the RS103 environments shown in Table 4.5.1-2 (and plotted Figure 4.5.1-2).

Note: Equipment that is required to operate during launch must meet its performance requirements while exposed to the radiated fields in Table 4.5.1-2.

Note: Testing to be in accordance with the RS103 test method and Section 4.3.10.4 in MIL-STD-461F.

Table 4.5.1-2 LV/LS Baseline Radiated Susceptibility Limits (RS103).* [TBR]

Type	Frequency Range	RS103 Limit V/m	Note
General Baseline	10 kHz – 18 GHz	20	6 dB margin added
S-Band Transmitter	2200 MHz – 2300 MHz	85	6 dB margin added
S-Band Transmitter	2300 MHz – 2400 MHz	20	6 dB margin added

* Note: These limits cover any of the potential launch vehicles as well as the launch site RS103 environments. These are survival requirements for equipment that is off during launch and performance requirements for equipment required to operate during launch.

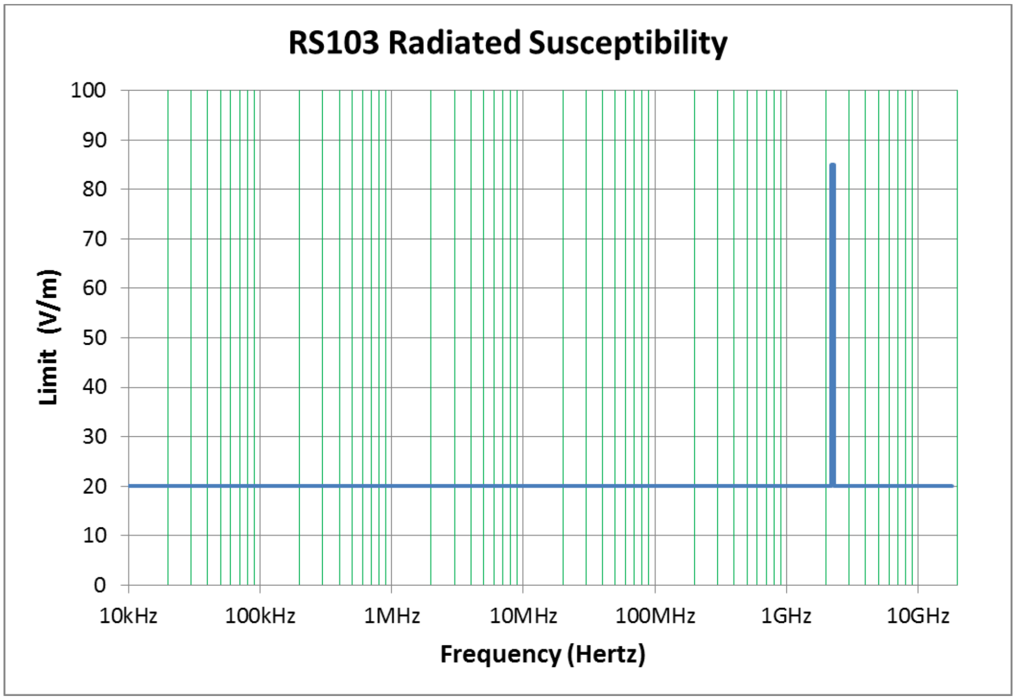


Figure 4.5.1-2 LV/LS Baseline Radiated Susceptibility Limits (RS103). [TBR]

4.5.1.3 LV/LS – Flight System EMC/RF Compatibility Analysis

It is essential that a detailed effort be undertaken to develop accurate descriptions of the EMC/RF environments that the flight system will be exposed to during launch and prior testing at the launch site. This effort will confirm that proper EMC/RF compatibility requirements have been defined and flowed down to the flight system.

Requirement: An EMC analysis **shall** be performed to verify EMC/RF compatibility will exist between the flight system and the launch vehicle, and between the SC/LV and the Eastern and Western Range environments of the launch site, including updates as available.

The EMC testing organization should issue a report based on analyses performed for the LV/LS – flight system EM**C**/RF Compatibility and provide conclusions and recommendations as needed.

4.5.2 Electromagnetic Interference / Compatibility (EMI/EMC) Requirements for Instruments, Subsystems, and Assemblies.

4.5.2.1 Conducted Emissions (CE) Requirements

These requirements apply to the electrical (power and signal) interfaces between instruments, subsystems, and assemblies and the flight system (spacecraft and instruments).

4.5.2.1.1 Conducted Emissions, Power Leads, 30 Hz to 75 MHz (CE101, Tailored)

Requirement: Conducted emissions appearing on instruments, subsystems, and assemblies primary power lines in differential mode **shall** not exceed the limits shown in Figure 4.5.2-1

These limits are for differential measurements between the active and return power leads.

Note: The method to be used is the CE101 test method in MIL-STD-461F, with the measurement bandwidth selected in accordance with Table II, MIL-STD-461F.

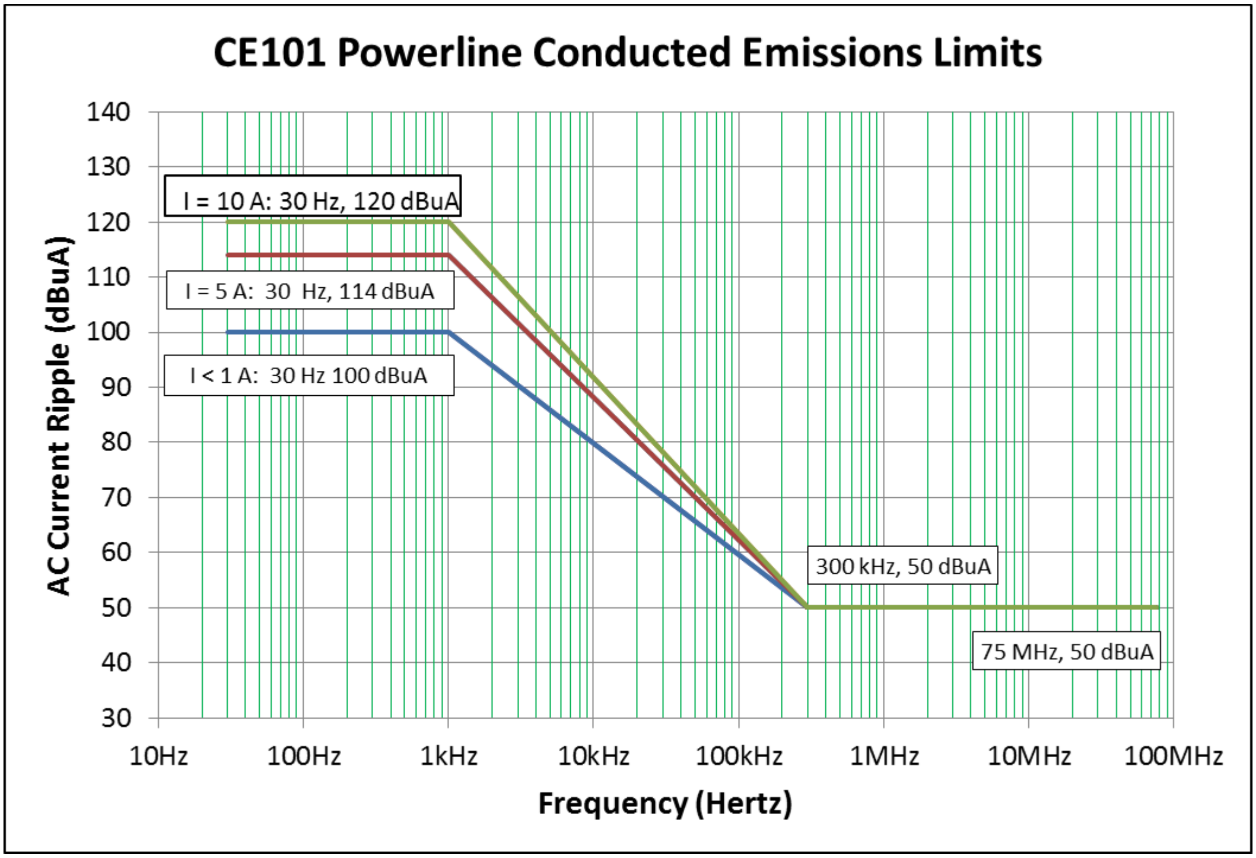


Figure 4.5.2-1 Conducted emission limits, powerlines, differential mode.

Requirement: The CE Limit in Figure 4.5.2-1 **shall** be scaled up by 20*log(I/1A), as illustrated in Figure 4.5.2-1 for I = 5 A and I = 10 A, if the instruments/subsystems/assemblies nominal input DC current, I, exceeds 1 A.

For I< 1A, the limit-line for I = 1 A is applicable. [**TBR**]

4.5.2.1.2 Conducted Emissions, Power and Signal Leads, Common Mode

Requirement: Common Mode conducted current emissions appearing on the flight system's primary power lines **shall** not exceed the limits of Figure 4.5.2-2.

Requirement: Common Mode conducted current emissions appearing on signal lines **shall** not exceed the limits of Figure 4.5.2-2.

Note: The CE101 test methodology in MIL-STD-461F may be used for the common mode test, with the measurement bandwidth set in accordance with MIL-STD-461F/Table II. [**TBR**]

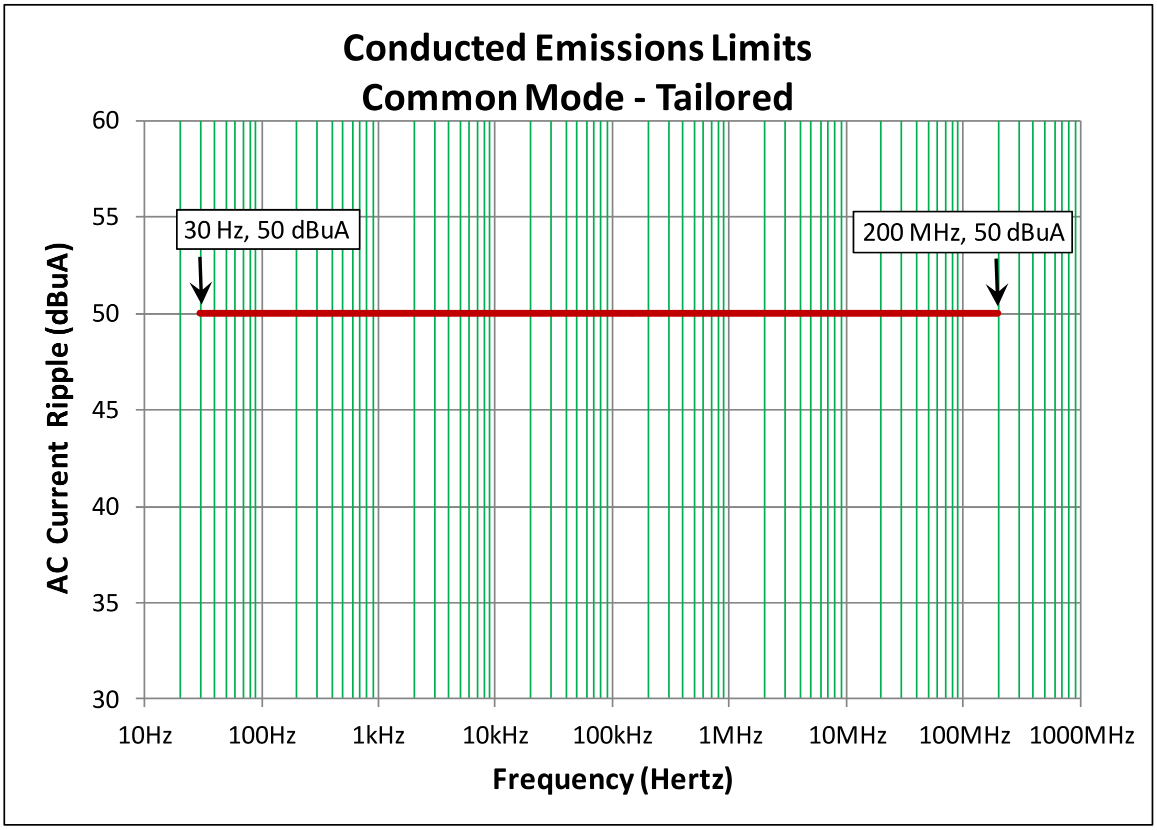


Figure 4.5.2-2 Conducted Emission Limit, Power and Signal Lines, Common Mode. [TBR]

Requirement-Policy: The Differential Mode and Common Mode conducted current test wiring configuration shall be in accordance with Figure 4.5.2-3.

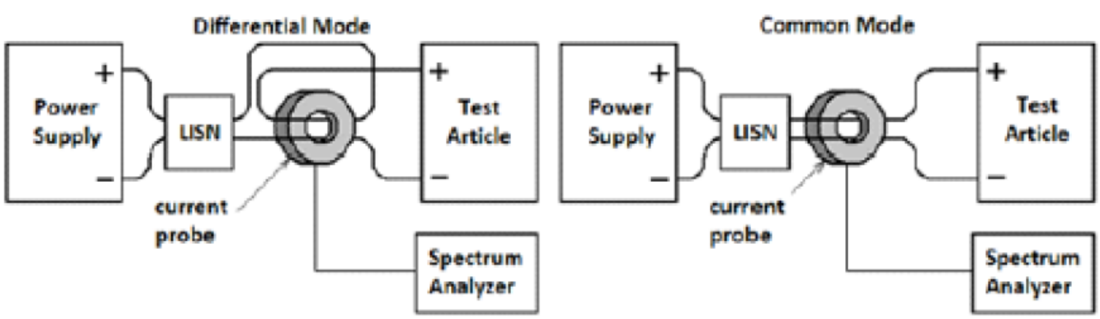


Figure 4.5.2-3 Conducted Emission Differential Mode and Common Mode Test Wiring Configurations.

4.5.2.1.3 Conducted Emissions, Antenna Terminals, 10 kHz to 18 GHz (CE106)

Requirement: Conducted emissions appearing on the antenna-connected terminals of RF transmitters and receivers **shall** not exceed the limits shown below:

a. Receivers: 34 dBμV
b. Transmitters and amplifiers (standby mode): 34 dBμV
c. Transmitters and amplifiers (transmit mode): Harmonics, except the second and third, and all other spurious emissions to be at least 80 dB down from the level at the fundamental. The second and third

harmonics to be suppressed to a level of -20 dBm or 80 dB below the fundamental, whichever requires less suppression.

Note:

- The test method is in accordance with CE106 in MIL-STD-461F.
- The transmit mode portion of this requirement is not applicable within the bandwidth of the transmitter.
- The requirement is not applicable to RF instruments that are designed with antennas permanently mounted to the instrument. (RE103 is then the applicable requirement.)
- The test start frequency may be selected based on the operating frequency of the RF instrument, as shown in Table 4.5.2-1 below:
-

Table 4.5.2-1 CE106 Operating Frequency Ranges.

Operating Frequency Range	Start Frequency for Test
10 kHz – 3 MHz	10 kHz
3 MHz – 300 MHz	100 kHz
300 MHz – 3 GHz	1 MHz
3 GHz – 18 GHz	10 MHz

4.5.2.1.4 Conducted Emissions, Power Transients, Time Domain (CE107)

Requirement: Instruments, subsystems, and assemblies **shall** not produce differential transient voltage spikes on the flight system power bus under any operating mode or transition condition in excess of the limits specified in figure 4.5.2-4. [**TBR**]

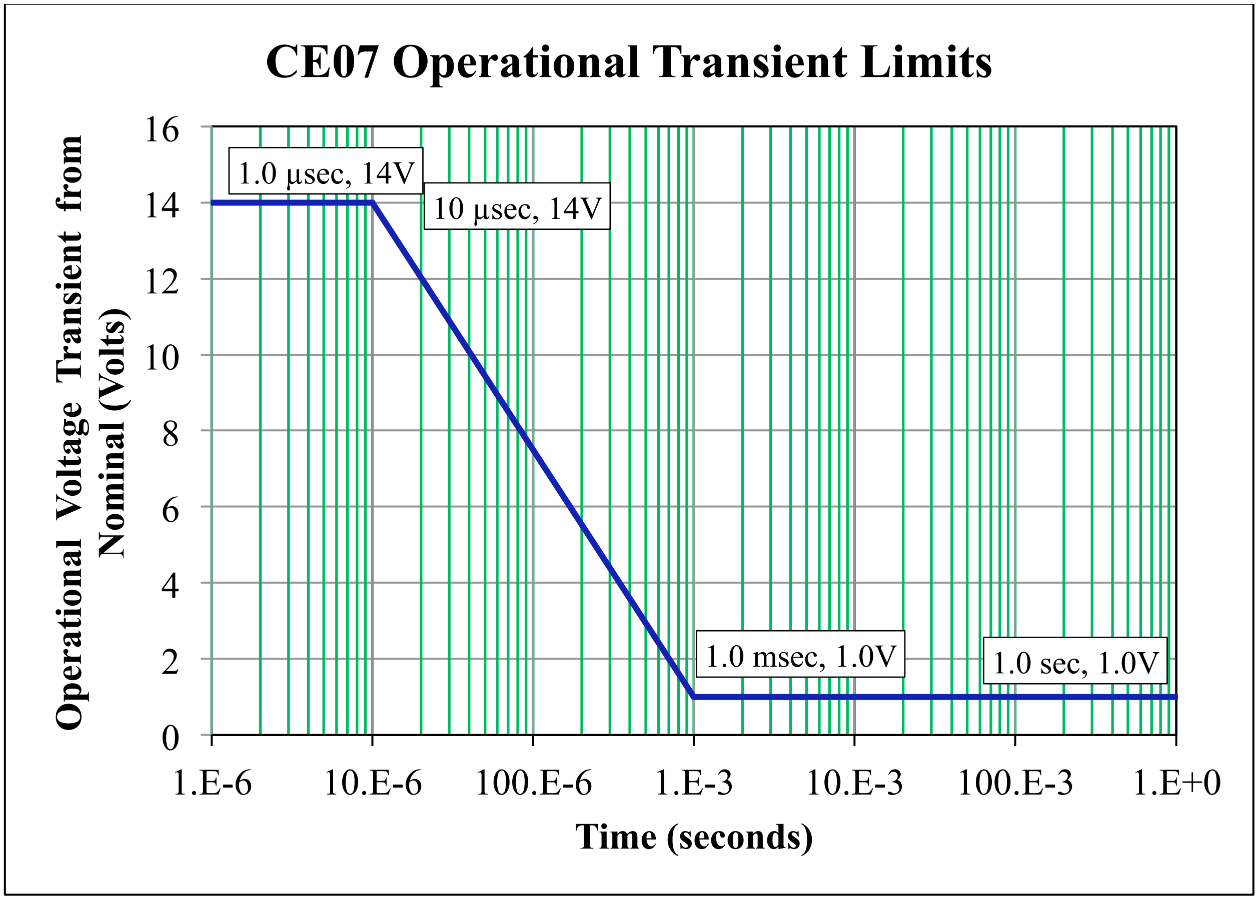


Figure 4.5.2-4 Conducted emission limit, Power lines, time domain. [TBR]

4.5.2.2 Conducted Susceptibility (CS) Requirements

These requirements are applicable to any instruments, subsystems, and assemblies that receive power from the flight system Power Supply (SPS) or have electrical interfaces with the flight system. Since the SPS is also a flight system assembly, these requirements are also applicable to the SPS when it receives power from an external power source.

When tests are used for compliance verification, grounding and cabling (shielding and twisting) must be similar to flight configuration to the extent feasible, except for the configurations imposed by MIL-STD-461F.

4.5.2.2.1 Conducted Susceptibility, Power Leads, 30 Hz to 150 kHz, (CS101)

Requirement: Instruments, subsystems, and assemblies **shall** not exhibit any malfunction, degradation of performance, or deviation from specified indications when their input power line is subjected to injected voltage levels or the power limit (whichever occurs first) shown in Table 4.5.2-2.

Table 4.5.2-2 CS101 Conducted Susceptibility Limits, 30 Hz – 150 kHz.

Frequency Range	Voltage Limit (V_{rms})	Power Limit (Watts)
30 Hz – 5 kHz	1	2
5 kHz – 150 kHz	0.5	0.5

Note: the CS101 test method in MIL-STD-461F may be used for testing. The applicable power limit is the power level that has been calibrated into a 0.5-ohm load using the voltage levels in Table 4.5.2-2 and the procedure in MIL-STD-461F.

4.5.2.2.2 Conducted Susceptibility, Rejection of Undesired Signals (CS104)

This requirement is applicable to RF equipment/subsystems such as communications receivers, RF amplifiers, transceivers, and radar receivers.

Requirement: RF receiver subsystems and assemblies **shall** operate within performance parameters without indications of interference, degradation of performance or malfunction when subjected to the test signal levels shown in Figure 4.5.2-5.

Note: The CS104 test method in MIL-STD-461F is the applicable test method.

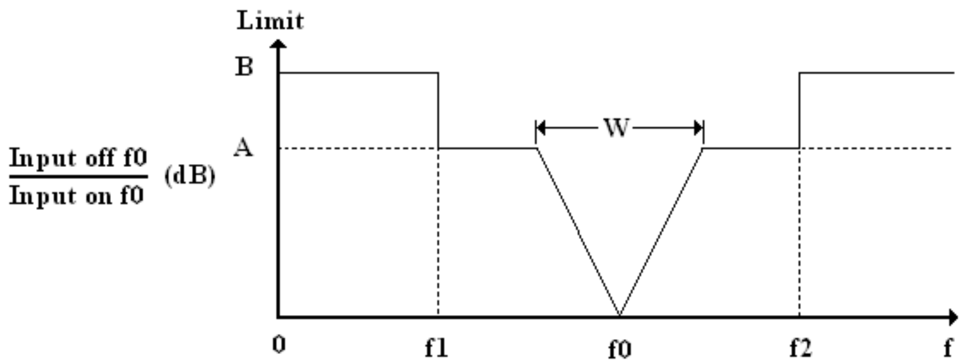


Figure 4.5.2-5 Sample Receiver Terminals Conducted Susceptibility Limit CS104.

Definitions:

1. fo = receiver tuned frequency or band center for amplifiers.
2. f1 = lowest tunable frequency of receiver band in use or the lowest frequency of amplifier passband.
3. f2 = highest tunable frequency of receiver band in use or the highest frequency of amplifier passband.
4. W = bandwidth between the 80 dB points of the receiver selectivity curve as defined in the test article's technical requirements or the control plan.

Limits:

The limit at A is 80 dB above the input level required to produce the standard reference output (this limit not to be used for amplifiers).

The limit at B to be set as follows:

1. Receivers: 0 dBm applied directly to the receiver input terminals.
2. Amplifiers: The limit to be as specified in the test article's technical requirements or control plan. In the absence of such information, the default for limit B is 0 dBm.

4.5.2.2.3 Conducted Susceptibility, Power Line Transients – Spikes (CS106)

Requirement: Instruments, subsystems, and assemblies **shall** not exhibit malfunction or unacceptable degradation of performance when the input power line is subjected to transient voltage levels shown in Figure 4.5.2-6 and as noted below.

1. Use the test methodology as outlined in the CS106 in MIL-STD-461F. The applied voltage profile may be tailored by the responsible EMI/EMC engineer.
2. Both positive and negative pulses must be applied to the +28 V power line (differential mode). However, care must be taken to prevent the line voltage from falling below 0 V during negative transients. A dry run

should be performed prior to testing flight hardware in order to make sure the positive voltage amplitude does not exceed the limit in Figure 4.5.2-6 and a negative voltage condition will not occur during flight hardware testing.

3. Current will be limited to 6 A_{pp} regardless of the applied voltage.
4. Both positive and negative pulses must be applied to both power lines (common mode configuration). However, the respective peak voltage amplitude will be 28/2 = 14 V in Figure 4.5.2-6.
5. The transient voltage pulses must be applied for a duration of 5 minutes at a repetition rate of 60 pps.

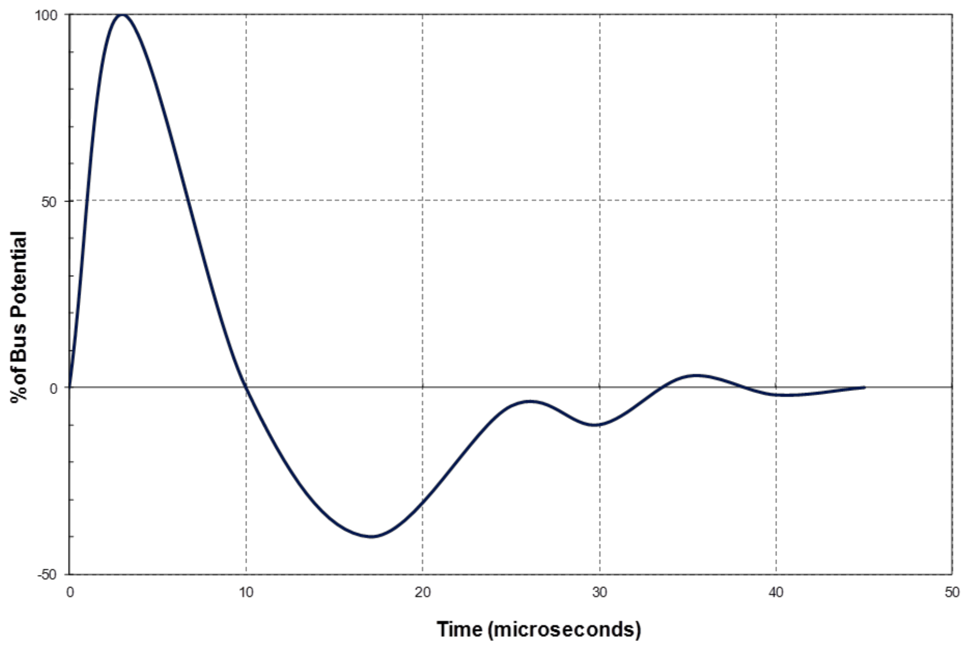


Figure 4.5.2-6 Conducted Susceptibility, CS106, Voltage Transient Profile.

4.5.2.2.4 Conducted Susceptibility, Power Leads, Deferential Mode (CS114)

Requirement: Instruments, subsystems, and assemblies **shall** not exhibit malfunction, unacceptable degradation of performance, or deviation from specified indications when subjected to an injection-probe drive level that has been pre-calibrated to the appropriate current limit as shown in Figure 4.5.2-7 and pulse modulated square wave at 1 kHz rate and 50% duty cycle.

Note: The test method and calibration procedure are in accordance with MIL-STD-461F. This requirement is only applicable to cable interfaces between instruments, subsystems, or assemblies and the flight system.

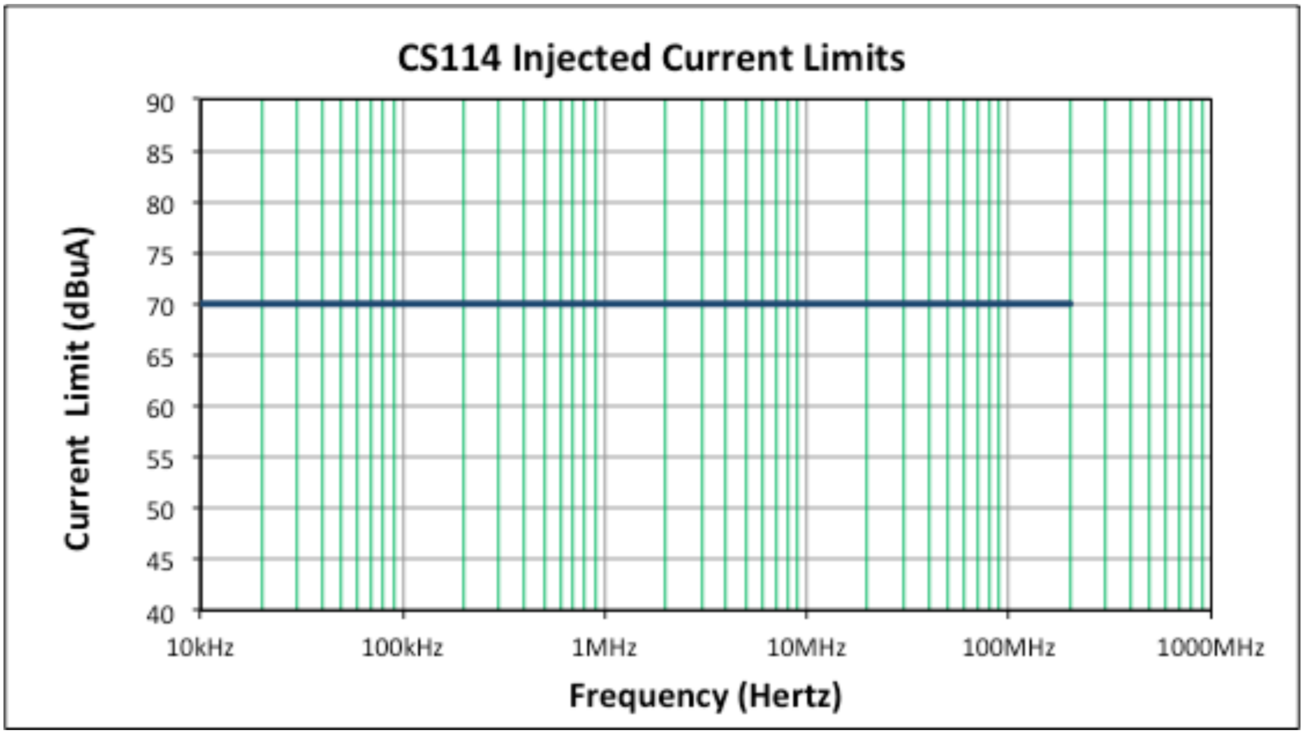


Figure 4.5.2-7 CS114 Calibration Limit.

4.5.2.2.5 Conducted Susceptibility, Power Leads, Differential Mode (CS02)

Requirement: Instruments, subsystems, and assemblies **shall** not exhibit malfunction, unacceptable degradation of performance, or deviation from specified indications when the input power leads (prime power only) are subjected to the requirements of MIL-STD-461C/462, test method CS02, as tailored below:

1. Test frequency range: 150 kHz to 50 MHz
2. Injected signal amplitude: 0.5 Vrms, but injected power not to exceed a calibrated limit of 0.5 W.
3. The applicable power limit is the power level that has been calibrated into a 0.5-ohm load using a 0.5 V_{rms} drive signal.
4. Pulse modulation at 1 kHz, 50% duty cycle.
5. Signal injection to be applied high side to return (differential mode), and high side to chassis.

4.5.2.3 Radiated Emission (RE) Requirements

The radiated emissions requirements, RE101 and RE102, are imposed to safeguard against interference with flight system and RF receivers from noise emitted from instruments, subsystems, or assemblies. The RE103 requirements apply to radar transmitters and place out-of-band spectral limits on antenna radiated spectrum. These limits are intended to prevent interference with other receivers.

Radar transmitters must also comply with the Federal Communications Commission (FCC) and National Telecommunications and Information Administration (NTIA) requirements on broadcasting in free space. These types of requirements, however, are not covered in this document.

4.5.2.3.1 Radiated Emission, AC Magnetic Field, 30 Hz to 100 kHz (RE101) [TBR]

Requirement: Instruments, subsystems, and assemblies **shall** meet the tailored ac magnetic field limits shown in Figure 4.5.2-8 at frequencies above 30 Hz when measured at a distance of 1 meter from the test article, using techniques similar to method RE101 of MIL-STD-462. The applicable limit is determined by the physical distance between the test article and the sensitive unit on the flight system in the installed configuration; all measurements are made at 1 meter distance.

Note: The magnetic field probes typically used for MIL-STD-462 method RE101 measurements are not sufficiently sensitive to measure to the levels shown in Figure 4.5.2-8. Specific test instrumentation and procedures will be documented in the individual equipment EMI test plan.

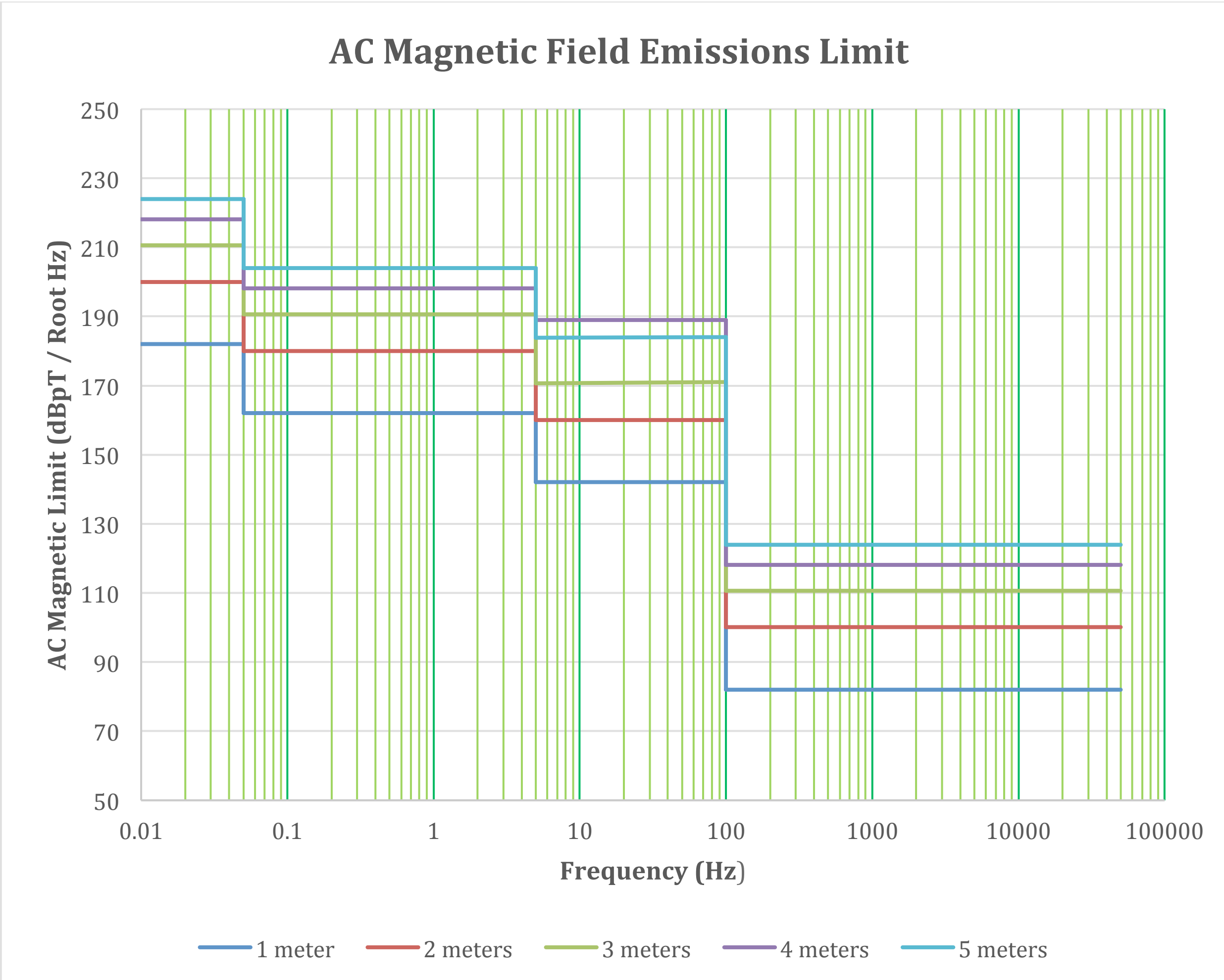


Figure 4.5.2-8 RE101 Magnetic Field Emissions Limit. [TBR]

4.5.2.3.2 Radiated Emissions, Electric Field, 30 Hz to 18 GHz (RE102)

Requirement: Electric field radiated emissions from instruments, subsystems, and assemblies **shall** not exceed the limits listed below:

a) For equipment located within the shielded vault area, the limit is 60 dBµV/m from 30 Hz to 18 GHz.
b) For equipment located outside the vault, the general limit is as shown in Figure 4.5.2-9 and as shown in Table 4.5.2-3 for Receiver notches. Note that the limit with 6dB margin is the applicable limit in 4.5.2-3.

Note: The test method is in accordance with MIL-STD-461F/ RE102 method.

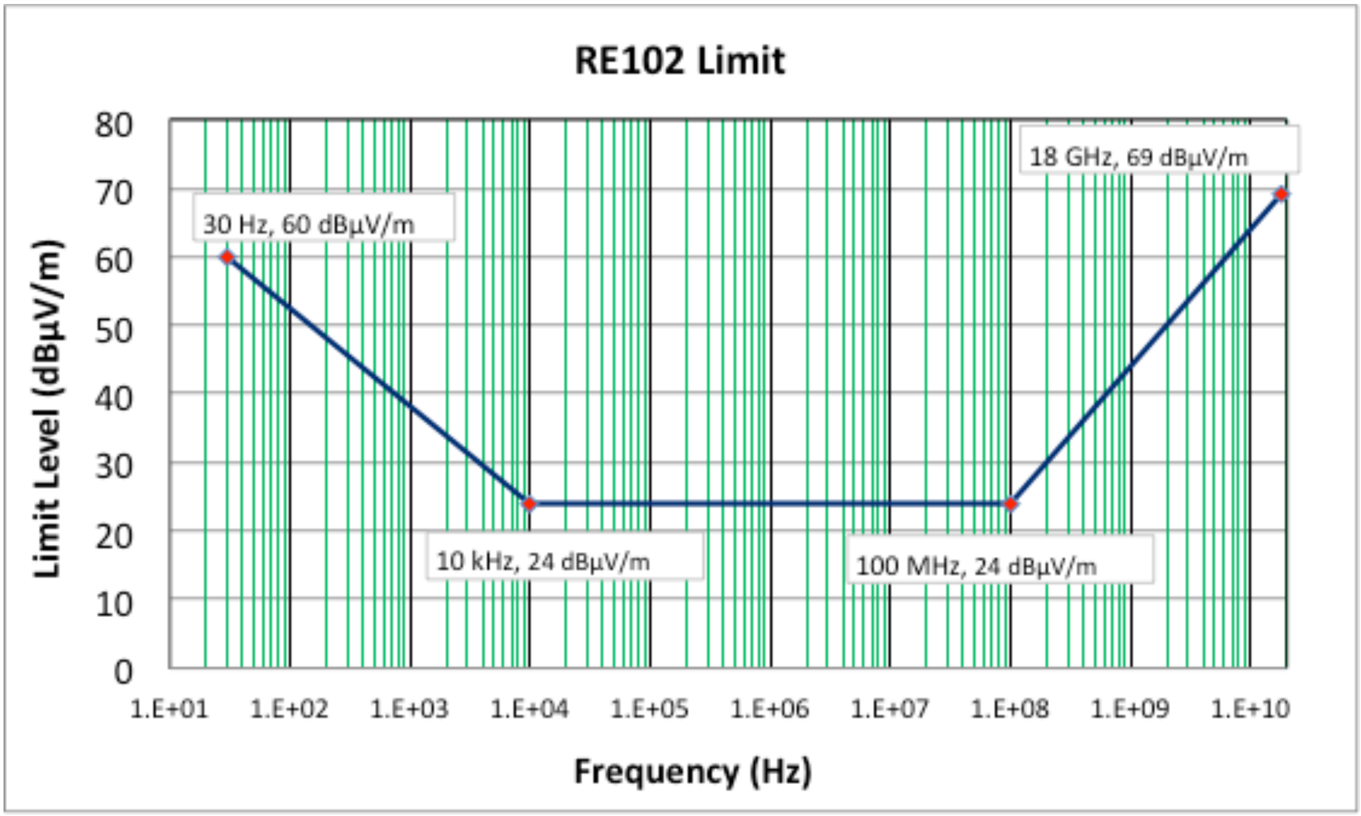


Figure 4.5.2-9 RE102 Electric Field Emissions Limit, Outside the Vault. [TBR]

Table 4.5.2-3 Radiated Emissions Limit in Receiver Notches. [TBR]

Component	Center Frequency (MHz)	Receiver Band (MHz)	Limit (dBµV/m)	Limit with 6 dB Margin (dBµV/m)
		Lander		
X-Band Receiver (from Carrier)	**8400 (TBR)**	**(TBD)**	**6 (TBR)**	**0 (TBR)**
X-Band Receiver (from Clipper)	**8400 (TBR)**	**(TBD)**	**6 (TBR)**	**0 (TBR)**
		Carrier		
X-Band Receiver (from Earth DSN)	**7200 (TBR)**	**7100.0 - 7300.0 (TBR)**	**6 (TBR)**	**0 (TBR)**

4.5.2.3.3 Radiated Emissions, Measurement Bandwidth, and Dwell Times

Requirement: The measurement bandwidths for radiated emissions tests and dwell times **shall** be selected in accordance with Table 4.5.2-4. For receiver notches, custom bandwidths must be used in order to bring the ambient level at least 6 dB below the applicable notch limit line, in accordance with MIL-STD-461F.

Table 4.5.2-4 Radiated Emissions Measurement Bandwidth and Dwell Time.

Frequency Range	6 dB Bandwidth	Dwell Time*	Minimum Measurement Time Analog Measurement Receiver*
30 Hz – 1 kHz	10 Hz	0.15 sec	0.015 sec/Hz
1 kHz – 10 kHz	100 Hz	0.015 sec	0.15 sec/Hz
10 kHz – 150 kHz	1 kHz	0.015 sec	0.015 sec/kHz
150 kHz – 30 MHz	10 kHz	0.015 sec	1.5 sec/MHz
30 MHz – 1 GHz	100 kHz	0.015 sec	0.15 sec/MHz
Above 1 GHz	1 MHz	0.015 sec	15 sec/GHz

* Dwell time must be long enough to ensure all emissions at the test frequency are captured.

4.5.2.3.4 Antenna Spurious and Harmonic Outputs (RE103)

Requirement: Spurious and harmonics levels in the Antenna radiated spectrum **shall** be at least 80 dB down from the level at the fundamental, except for the second and third harmonics. The second and third harmonics levels may be limited to -20 dBm or 80 dB below the fundamental, whichever requires less suppression.

Note: This requirement applies to the radiated spectrum from a transmitter antenna, with measurement to be in accordance with MIL-STD-461F.

4.5.2.3.5 DC Magnetic Field Emissions

Europa Lander Instruments, Subsystems, and Assemblies may be magnetically sensitive. As the design matures, the DC magnetic requirements needed will be determined.

Requirement: The DC magnetic field emissions from instruments, subsystems, and assemblies **shall** not exceed **[TBD]** nT under all operating conditions, when measured at one meter from its center and at a point where magnetic emissions are at maximum.

4.5.2.4 Radiated Susceptibility (RS) Requirements

Radiated susceptibility (RS) requirements are imposed in order to minimize or eliminate risks associated with the proper operation of instruments, subsystems, or assemblies and the flight system while exposed to the electromagnetic environments present at launch and during mission operations. These elements are typically subjected to radiated susceptibility tests to demonstrate compliance with the corresponding requirements. In general, test levels will include a 6 dB safety margin in order to safeguard against unpredicted susceptibility at the flight system level.

4.5.2.4.1 Radiated Susceptibility, AC Magnetic Field, 30 Hz to 100 kHz (RS101) [TBR]

Requirement: Instruments, subsystems, and assemblies **shall** not exhibit malfunction, unacceptable degradation of performance, or deviation from specified indications while exposed to the AC magnetic fields shown in Figure 4.5.2-10.

Note: The test method is in accordance with the RS101 test method in MIL-STD-461F.

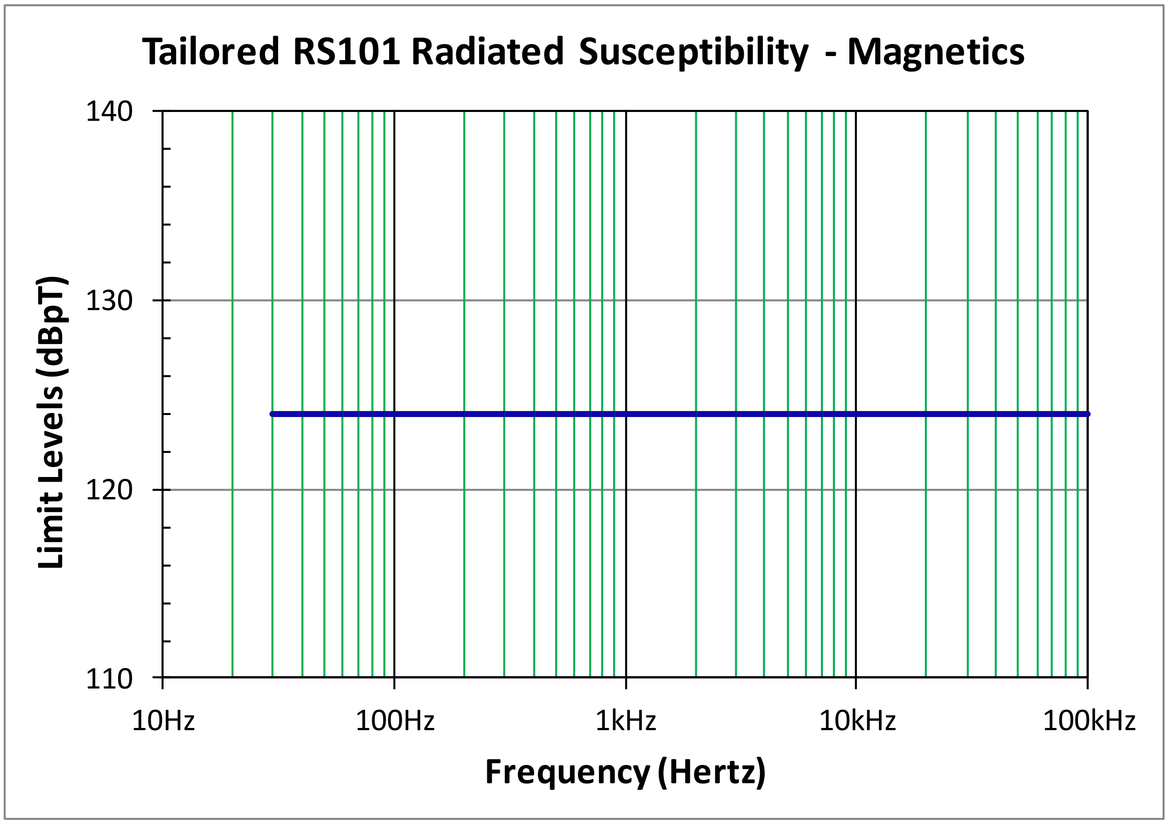


Figure 4.5.2-10 RS101 Magnetic Field Limit. [TBR]

4.5.2.4.2 Radiated Susceptibility, Electric Field, 10 kHz to 18 GHz (RS103)

Requirement: Instruments, subsystems, and assemblies **shall** not exhibit malfunction, unacceptable degradation of performance, or deviation from specified indications when exposed to the radiated E-field levels shown below:

1. Survival E-Field Levels: 20 V/m from 10 kHz to 18 GHz and Transmitters fields per Table 4.5.1-2 for instruments that are powered off during Launch.
2. Operational E-Field Levels: 20 V/m from 10 kHz to 18 GHz and Transmitters fields per Table 4.5.1-2 for instruments that are operating during Launch.
3. 2 V/m from 10 kHz to 18 GHz, On-Orbit/Surface Operational Field Level. The On-Orbit/Surface E-field environment also includes flight system transmitters as listed below in Table 4.5.2-5.

Note: The test method is in accordance with the RS103 test method in MIL-STD-461F using pulse modulation of a 1 kHz square wave and 50% duty cycle as outlined in MIL-STD-461F.

Note: The survival requirement applies to instruments with instrument in power off configuration, and Instrument must meet its performance requirement after exposure to the radiated fields.

Note: The operational requirement applies while an instrument in its normal operational mode. An instrument must meet its performance requirement during exposure to the radiated fields.

Note: Up to 30 MHz, only vertically polarized E-fields are required for RS103 tests. Above 30 MHz, the requirement applies for both horizontally and vertically polarized E-fields. Circular polarized fields are not acceptable.

Table 4.5.2-5 Flight System Transmitters On-Orbit/Surface E-Field Levels. [TBR]

Transmitter	Frequency Range (MHz)	Power (dBm)	Radiated Field (V/m)

X-Band (Carrier)	8400 +/- **(TBD)**	**(TBD)**	**(TBD)**
X-Band (Lander)	7200 +/- **(TBD)**	**(TBD)**	**(TBD)**

4.5.2.4.3 Radiated Susceptibility, Measurement Scan Rates

Requirement: The measurement scan rates or step size for RS103 radiated susceptibility tests **shall** be selected in accordance with Table 4.5.2-6.

Tabel 4.5.2-6 RS103 Scan Rates/Step Size.

Frequency Range	Analog Scans Maximum Scan Rates	Stepped Scans Maximum Step Size
30 Hz – 1 MHz	0.0333 f_0/sec	0.05 f_0
1 MHz – 30 MHz	0.000667 f_0/sec	0.01 f_0
30 MHz – 1 GHz	0.00333 f_0/sec	0.005 f_0
1 GHz – 40 GHz	0.00167 f_0/sec	0.0025 f_0

4.5.2.4.4 Radiated Susceptibility, Measurement Dwell Times

Requirement: When using step scans, dwell time at each tuned frequency **shall** be 0.15 seconds up to 1 kHz and 0.015 seconds above 1 kHz.

Note: The scan rates noted above must be slow enough, or the dwell time long enough, to allow for the observation and recording of the test article response to the radiated E-fields under worst-case operating conditions. Use the guidelines in MIL-STD-461F.

4.5.2.4.5 Radiated Susceptibility, Test Configuration

Requirement-Policy: For Radiated Susceptibility testing, to the extent feasible, the test article grounding, cabling, and RF shielding **shall** be similar to flight configuration.

4.5.2.4.6 DC Magnetic Susceptibility

All flight hardware will be required to operate in the natural background magnetic field. On Earth and in Low-Earth Orbit, the DC magnetic field strength is approximately 0.61 Gauss. Near Europa, Jupiter's intrinsic field is approximately two orders of magnitude smaller. Some subsystems (e.g. motors and telecom systems) are also possible sources of DC magnetic field. As the design matures, the total expected DC field will be determined.

Requirement: Instruments, subsystems, and assemblies **shall** operate without damage or performance degradation when exposed to a DC magnetic field of **[TBD]** Gauss.

4.5.3 Power Transients

The power transient requirements are imposed to ensure each instrument, subsystem, or assembly will be compatible with the transient characteristics and requirements of the flight system power bus.

4.5.3.1 Turn-On and Off Transient Characteristics

4.5.3.1.1 Turn-on In-Rush Current

Requirement: Current Limit. Instrument, subsystem, or assembly inrush current at turn-on **shall** not exceed the limits shown Figure 4.5.3-1, or 50 A [**TBR**], whichever is smaller.

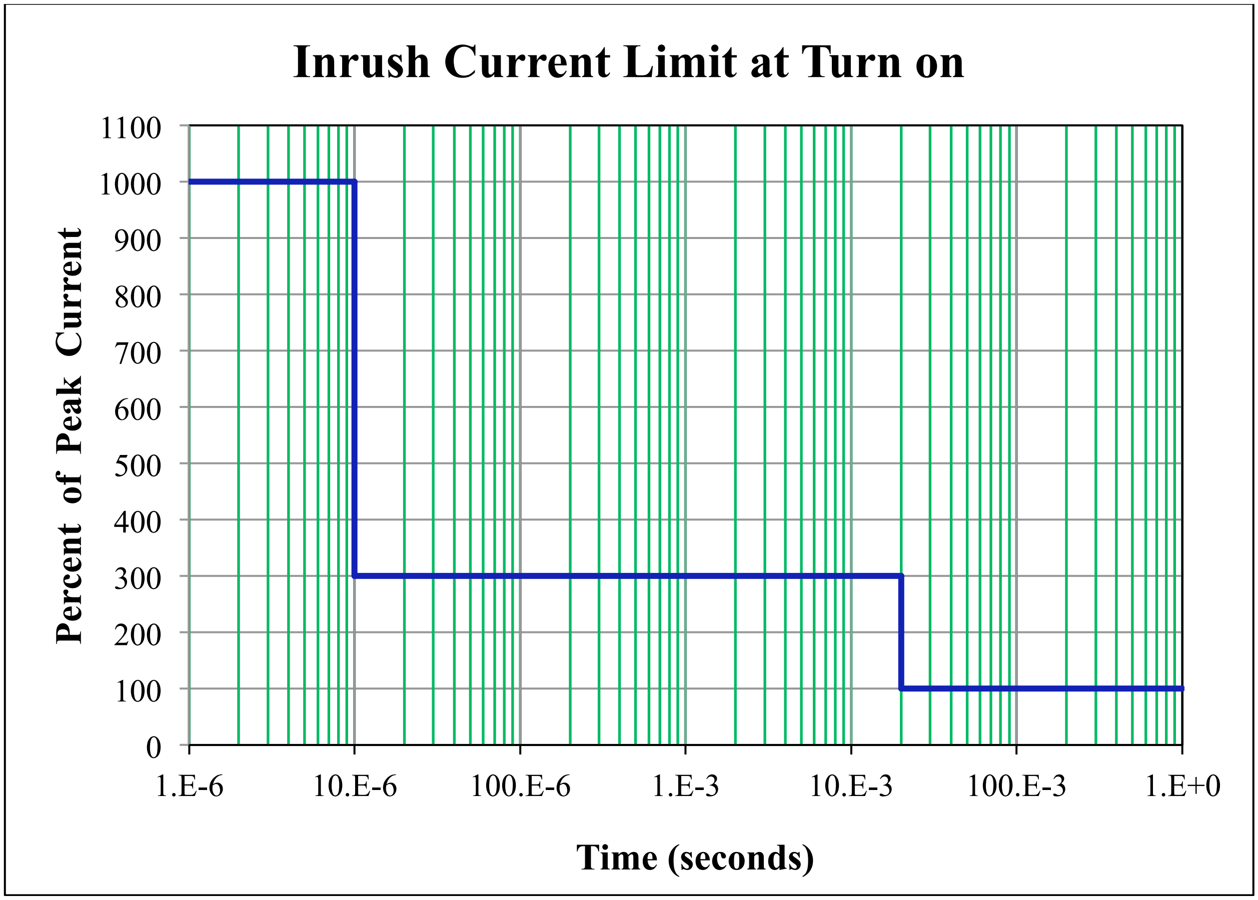


Figure 4.5.3-1 Inrush current limit at turn-on. [TBR]

4.5.3.2 Operational Transients. [TBR]

4.5.3.2.1 Normal Transients

Requirement: Instruments, subsystems, or assemblies **shall** withstand voltage transients on input power lines within the power envelope as shown in Figure 4.5.3-2.

Note: The envelope includes two separate limits: a) voltage increase from nominal 28 VDC to 50 VDC, followed by voltage ramp down from 50 V down to 29 V; and b) voltage ramp down from nominal 28 VDC to 18 VDC, followed voltage ramp up from 18 V to 22 V, as shown in Figure 4.5.3-2.

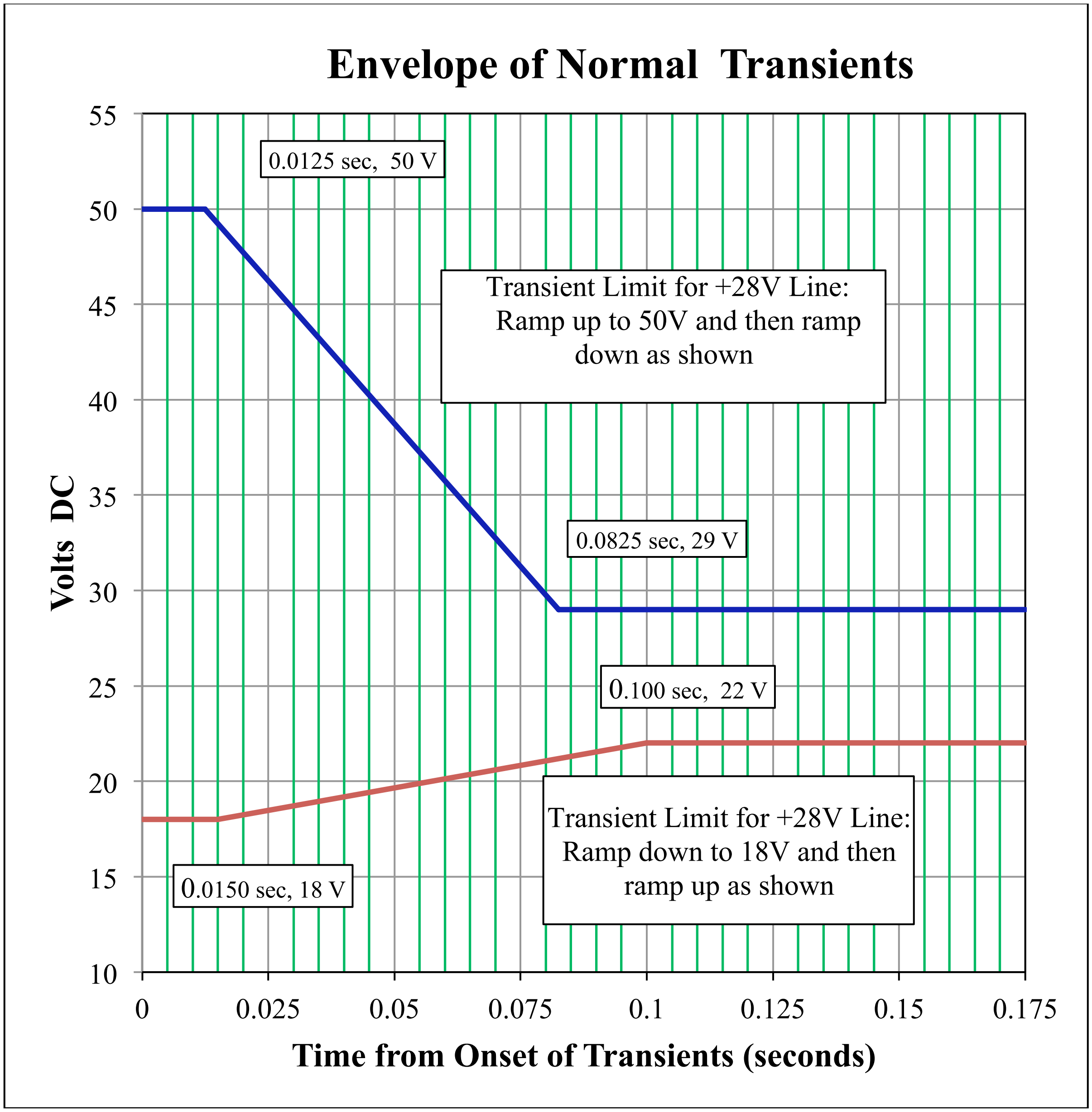


Figure 4.5.3-2 Envelope of Normal Voltage Transients For 28 Volts DC System.

4.5.3.2.2 Abnormal Transients

Requirement: Instruments, subsystems, or assemblies **shall** withstand voltage transients on input power lines up to 58 V for 10 microseconds. The transient may occur at power off or while operating at 28 VDC.

4.5.3.2.3 Power Interrupt Transients

Requirement: Instruments, subsystems, or assemblies **shall** return to normal operation, without experiencing any damage or performance degradation, after encountering an input power interrupt lasting 50 milliseconds. ([**TBR**]: update per MIL-STD-704F.)

Note: A power interrupt is defined as the line voltage dropping to zero volt from nominal operational voltage and then returning to the nominal level.

4.5.3.2.4 Ground Fault

Requirement: Instruments, subsystems, or assemblies **shall** return to normal operation, without experiencing any damage or performance degradation, after being subjected to a Ground Fault test.

The Ground Fault test configuration is shown in Figure 4.5.3-3.

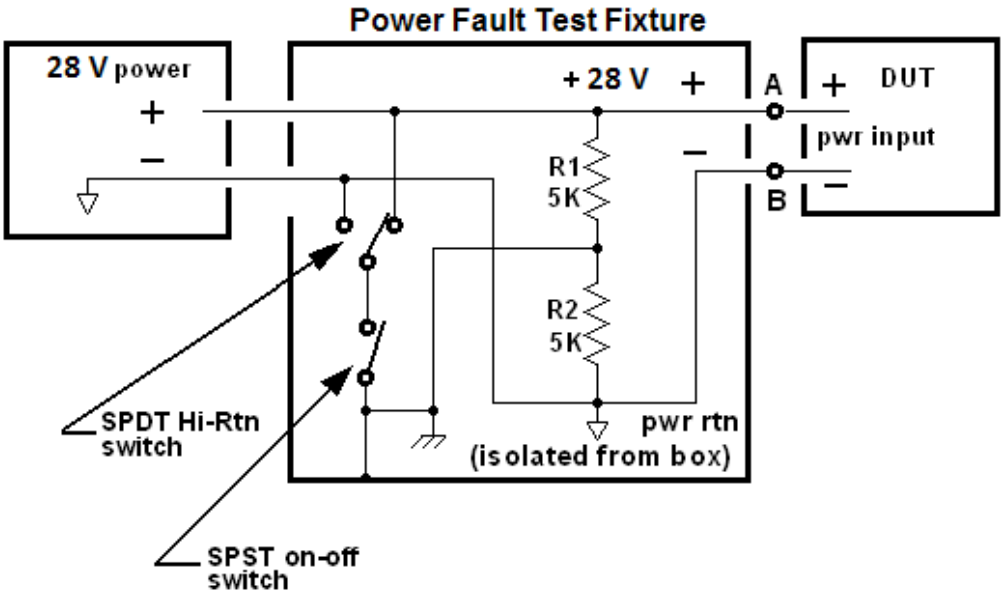


Figure 4.5.3-3 Power Systems Fault Test Configuration. [Preliminary]

4.5.4 Lightning Protection

Requirement: Any interface circuit connected to the T-0 umbilical **shall** be able to survive lighting-induced voltage transients of 200 volts peak and 10 microseconds in duration which has been calibrated against a 5-Ohm resistor.

([TBR]: Need to consider T-0 umbilical length inductive effects which can produce longer transients with higher energy.)

4.5.5 Grounding, EMI Bonding, and Isolation

The requirements of this section are generally consistent with the recommendations of NASA HDBK 4001, Electrical Grounding Architecture for Unmanned Spacecraft, Feb 17, 1998.

The grounding requirements are intended to establish a grounding architecture based on a single-point ground system for instruments, subsystems, or assemblies and to implement a zero-volt ground reference/chassis to which all flight system hardware would be referenced. Figure 4.5.5-1illustrates a Distributed Single Point Grounding Scheme.

EMI bonding requirements are imposed in order to provide compliance with two important design objectives:

1. EMI/RF grounding of electronic equipment to system Ground Reference.
2. EMI/RF shielding for compliance with applicable radiated emissions, radiated susceptibility, and RF compatibility requirements.

4.5.5.1 Grounding and EMI Bonding Requirements

Requirement: Instruments, subsystems, or assemblies **shall** provide a means for grounding to the Flight System Chassis Ground in a scheme similar to that shown in Figure 4.5.5-1

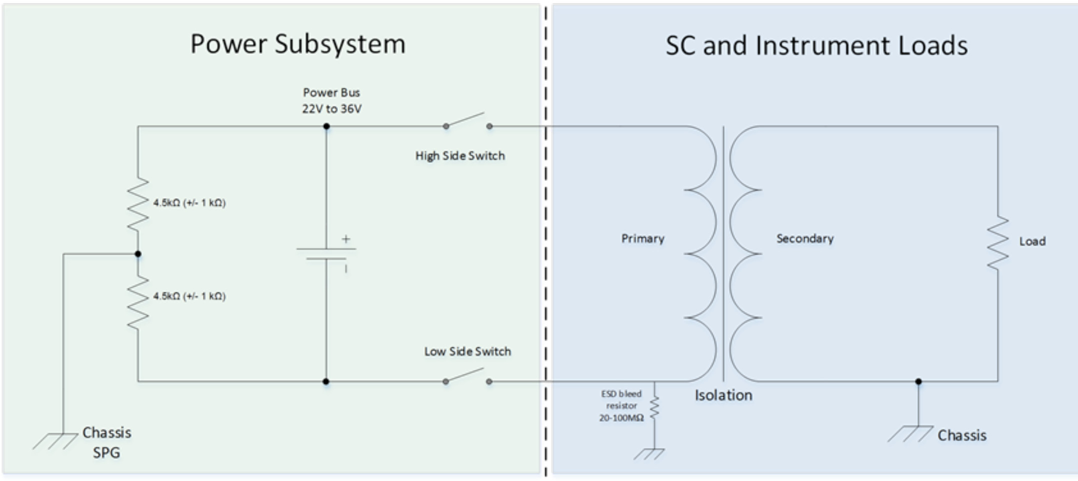


Figure 4.5.5-1 Illustration of Distributed Single Point Grounding Scheme. [TBR]

Requirement: Each electrical equipment chassis (case) **shall** be bonded to the structure or Ground Reference Rails (GRR) by means of direct metal-to-metal contact or a grounding strap, with bonding resistance not to exceed 5 mΩ per joint, as verified by direct measurement.

Note: Direct attachment/bonding of the equipment chassis to the GRR is the preferred approach. Where direct bonding is not possible and/or where moveable metal-to-metal joints are present, grounding straps may be used.

Requirement: For RF type electronic equipment, the bond strap employed for grounding to the Ground Reference **shall** have an inductance of < 100 nH and a length to width ratio of 5 to 1 maximum and 3 to 1 preferred.

Deviations from these requirements are granted subject to approval by the responsible EMI/EMC engineer.

Note: RF type equipment is electronic equipment that contains RF or digital electronics operating at frequencies/clocks of 100 kHz or higher.

Requirement: When ground straps are used to ground equipment to the Ground Reference Rails (GRR), at least two ground straps **shall** be used to provide for redundancy**.**

Requirement: Metallic parts of each electrical equipment chassis (case) **shall** be mutually bonded together by direct metal-to-metal contact (preferred method) across the entire bonding interface.

Note: This requirement is applicable to metallic parts that are used for grounding purposes or for EMI shielding purposes.

Requirement: Where an enclosure is used to provide EMI/RF shielding to internal electronics, the bonding resistance across any metal-to-metal seam or joint of the enclosure **shall** not exceed 2.5 mΩ, as verified by direct measurement across the seam.

Requirement: Each connector installed on electronic equipment **shall** be bonded to its respective panel by direct metal-to-metal contact with bonding resistance not to exceed 5 mΩ, as verified by direct measurement.

Requirement: The DC resistance between a connector on a cable and the respective connecting panel **shall** not exceed 20 milliohm.

Requirement: Preparation and surface finish of metal-to-metal surfaces for electrical bonding purposes **shall** be made in accordance with NASA-STD-4003, paragraph 5.2, "Surface cleaning and finishing."

4.5.5.2 Isolation [TBR]

4.5.5.2.1 Power

Requirement: Any power lead that may be disconnected **shall** be isolated from chassis ground, but terminated from line to chassis with a 20 MΩ resistor for ESD bleed.

Requirement: End-circuits receiving flight system power **shall** be isolated from chassis by at least 20 MΩ but not more than 100 MΩ.

Requirement: Power converters **shall** be isolated from chassis by permitting no lumped capacitance to chassis greater than 0.1 uF.

Requirement: Flight and ground support equipment **shall** maintain end-circuit isolation during both power-on and power-off conditions.

Requirement: Each secondary power form **shall** have a dedicated power return.

Requirement: Secondary power outputs at each instrument, subsystem, or assembly's power supply **shall** be isolated from chassis ground with resistance of 100 kΩ or greater for loads with RF circuits.

Requirement: For non-RF equipment, secondary power return **shall** be grounded to chassis at the power supply, but not grounded to chassis at the load.

Requirement: Chassis ground **shall** not be used as power return.

Requirement: Cable shields **shall** not be used as a power conductor.

Requirement: Signal return **shall** not be used as a power conductor.

4.5.5.2.2 Signal and Data

Requirement: For every signal, data, and command end-circuit pair forming an electrical interface, one end and only one end **shall** be isolated from chassis by > 20 MΩ.

Requirement: All pyro firing circuitry **shall** be isolated from chassis by >5 kΩ.

Requirement: Signal circuits **shall** be isolated from each end-circuit terminal circuit common by < 400 pF when measured from that interface pin to chassis.

Requirement: For standard differential interfaces such as RS-422, MIL-STD-1553, LVDS isolation **shall** be confirmed by resistance measurement using a DC ohmmeter (see NASA HDBK- 4001, par 4.2.5). RS-422 will not meet this isolation limit (see RS-422 differential circuits below).

Requirement: RS-422 differential circuits, which do not have "true" high-impedance isolation, **shall** be considered to have virtual high-impedance because of the differential nature of the circuits current flow (when both ends are powered-on). They are acceptable for general use, and only need to verify their standard level of isolation.

Requirement: Multiple signal end-circuits. The minimum net isolation resistance and capacitance (between circuit common and local chassis) required of two or more signal end-circuits sharing a common return **shall** be, respectively, 1 MΩ divided by the number of sharing end-circuits, and 400 pF multiplied by the number of sharing end-circuits.

Requirement: Except for bleed resistors, no capacitor, diode, filter, or other shunting device **shall** be connected between chassis and the input terminal of an isolated signal end-circuit. Shunting component connected from the input terminal of an isolated signal end-circuit, to chassis or to circuit common, may create a deleterious ground loop. Further, the use of discrete coupling capacitors between circuit elements referenced to different ground trees limits AC isolation and allows inter-ground tree transient coupling.

Requirement: Digital relay acquisitions and relay commands **shall** be completely electrically isolated with dedicated return.

Note: An ESD bleed requirement of 20 MΩ to 100 MΩ, though, still exists.

Requirement: Coaxial Cable End-Circuit Isolation. For coaxial circuits that connect flight equipment to GSE, T-0 umbilical circuits, or direct access circuits referenced to different ground trees, high pass filtering (DC blocking) ≥1.0 MΩ **shall** be provided in the center conductor and/or shield to maintain isolation where necessary to prevent the formation of a DC ground loop.

Requirement: Umbilical and Direct Access End-Circuit Isolation. The flight system end-circuits for all umbilical and direct access circuits **shall** provide the reference for the circuit except for circuits using temperature transducers, switch contacts, or relay coils as isolated flight end-circuits.

Requirement: Isolation tests **shall** be performed on all circuits required to be isolated from circuit common or chassis ground during EMC testing.

4.5.6 Wiring/Cable Shielding [TBR]

4.5.6.1 Standard Cables

Requirement: Interface cables (bundles) that join equipment housings **shall** use grounded over-braids or other approved wire/cable shielding method.

Note: The over-braids together with the equipment housings effectively enclose the entire system in a single electrically continuous Faraday chamber. Single layer wrap with copper tape or equivalent may be used if approved by the EMC engineer.

Requirement: The cable/wire over-braids **shall** be connected to equipment housings (housings are structure grounded) using connector EMI back-shells.

Note: The EMI back-shells serve to bond the cable braid to the housing-mounted connector through an ideal 360-degree termination.

Requirement: Signal or wiring shields **shall** not be used as an intentional power or signal return conductor (except coaxial cables, where shield is used as signal return).

Requirement: Cable wiring **shall** be configured as follows:
1. Wire cables will consist of individually shielded twisted pairs, triples, or quads.
2. Braid shields will cover the twisted pair or twisted group rather than individual wires.

3. Basic wire groups of system electrical interfaces should consist of two wires per circuit or a single wire for each of several circuits that share a circuit return wire.
4. The wires should be routed to minimize the area enclosed by the wires and twisted together.
5. Each cable bundle should contain "out" and "return" wiring such that maximum electromagnetic field cancellation is achieved. All wires providing return current for interfaces within a bundle should, themselves, reside within that bundle. In no case should there be a secondary or partial current path in a different bundle.

4.5.6.2 Ribbon and Flex Cables

Requirement: To minimize magnetic field generation and also to protect against shorting, primary and secondary power and return lines **shall** be located on nearby pins within a connector with a "guard" pin in between, or on adjacent layers of ribbon cabling.

Requirement: For differential circuits, each signal pair **shall** be assigned to conductors on immediately adjacent layers.

Requirement: For single-ended circuits sharing a common return, the signal runs **shall** be clustered into a group on one layer, with the shared return spanning the cluster on the immediate adjacent layer.

Requirement: For shielded conductors (singles, pairs, or multiple conductors), the shield **shall** be implemented using continuous conductors immediately above and below to sufficiently cover the signal line(s).

4.5.6.3 Pyro-Circuit Cable Shielding

Requirement: Pyro circuit pairs **shall** be made from twisted shielded pair (TSP) using braided shielding terminated in the connector back-shells.

Requirement: Individual braid shields **shall** be terminated using halo rings or equivalent and structure-terminated (grounded) to the back-shells.

Requirement: Pigtails, if required, **shall** not exceed 1 inch (2.54 cm) and totally contained within the outer overall shield to the back-shell on either end of a cable.

Requirement: Pigtails **shall** not terminate (grounded) through a connector pin.

Requirement: Pyro control, monitoring, and firing circuits **shall** meet the requirements of the Eastern and Western Range (EWR) 127-1 Range Safety Requirements document, Section 3.13.

4.5.7 EMI/EMC Requirements Verification

4.5.7.1 Test Method

This section describes the test methods and related requirements for verification of the applicable EMI/EMC requirements. Test methods apply at the instrument, subsystem, or assembly level or at the flight system level. Table 4.5.7-1 shows the applicable tests.

Table 4.5.7-1 EMC/EMI Test Matrix.

Test Method	Test Description	Assembly/ Subsystem/ Instrument	Flight System
CE101 (Tailored 461F)*	Conducted Emissions, Power Leads, 30 Hz to 50 MHz	Test	---
CECM	Conducted Emissions, Common Mode, 30 Hz to 200 MHz	Test	---
CE106 (Tailored 461F)	Conducted Emissions, Antenna Terminals	Test	---
CE07 (Tailored 461C)*	Conducted Emissions, Power Transients, Time Domain	Test	---
CS101 (Tailored 461F)	Conducted Susceptibility, Power Leads, 30 Hz to 150 kHz	Test	---
CS104 (Tailored 461F)	Conducted Susceptibility, Rejection of Undesired Signals	Test	---
CS106 (Tailored 461F)	Conducted Susceptibility, Power Line Transients - Spikes (CS106)	Test	---
CS114 (461F)	Conducted Susceptibility, Bulk Cable Injection, 10 kHz to 200 MHz	Test	---
CS114 (461F)	Conducted Susceptibility, Power Leads, Differential Mode, 10 kHz to 200 MHz	Test	---
RE101 (461F)	Radiated Emissions, AC Magnetic Field, 30 Hz to 100 kHz	Test	---
RE102 (Tailored 461F)	Radiated Emissions, AC Electric Field, 30 Hz to 18 GHz	Test	Test
RS101 (461F)	Radiated Susceptibility, AC Magnetic Field, 30 Hz to 100 kHz	Test	---
RS103 (461F)	Radiated Susceptibility, Electric Field, 14 kHz to 18 GHz	Test	Test
Magnetics	DC Magnetic Field Emissions	Test	---
Magnetics	DC Magnetic Field susceptibility	Test	---
Power Transients	In-Rush Current	Test	---
Operational Transients	Normal Transients	Test	---
Operational Transients	Abnormal Transients	Test	---
Power Transients	Power Interrupt	Test	---
Ground Fault	Power Ground Fault Test	Test	---
Lightning	Lightning Transients	Test/ Analyze	---
Bonding and Grounding	EMI Bonding Resistance Measurement	Test/ Inspection	Test/ Inspection
Isolation	Power and Signal Isolation	Test	---
Shielding	Enclosure and Cable Shielding	Test/ Inspection	Test/ Inspection
Plugs out	Powered by Spacecraft Batteries (no external power)	---	Test
Self Compatibility	RE and RS Flight System Components Self Compatibility	---	Test

*MIL-STD-461F or MIL-STD-461C, as applicable.

Early in the development phase, the EMC organization should develop a detailed test plan outlining tests to be performed at instrument, subsystem, or assembly level and at flight system level. The instrument, subsystem, or assembly level tests should also include any tests that should be performed at the component level. Some of these tests such as bonding and isolation must be implemented during assembly and build in order to make sure any potential non-compliance problem is fixed prior to releasing the instrument, subsystem, or assembly for final test.

The test plan should also state test locations for each instrument, subsystem, or assembly and for the flight system.

Requirement: Verification of the EMI/EMC requirements outlined above **shall** be by Test in accordance with MIL-STD-461F, or MIL-STD-461/462, as applicable.

Requirement: The test setup **shall** use either the 5 micro-Henry Line Impedance Stabilization Network (LISN) described in MIL-STD-461F or a custom device described below.

Requirement: The custom LISN **shall** be as shown in Figure 4.5.7-1 and Figure 4.5.7-2.

(Note: Impedance level in the low-frequency region of plot assumes power-source impedance; the plot shown assumes a 1 ohm source).

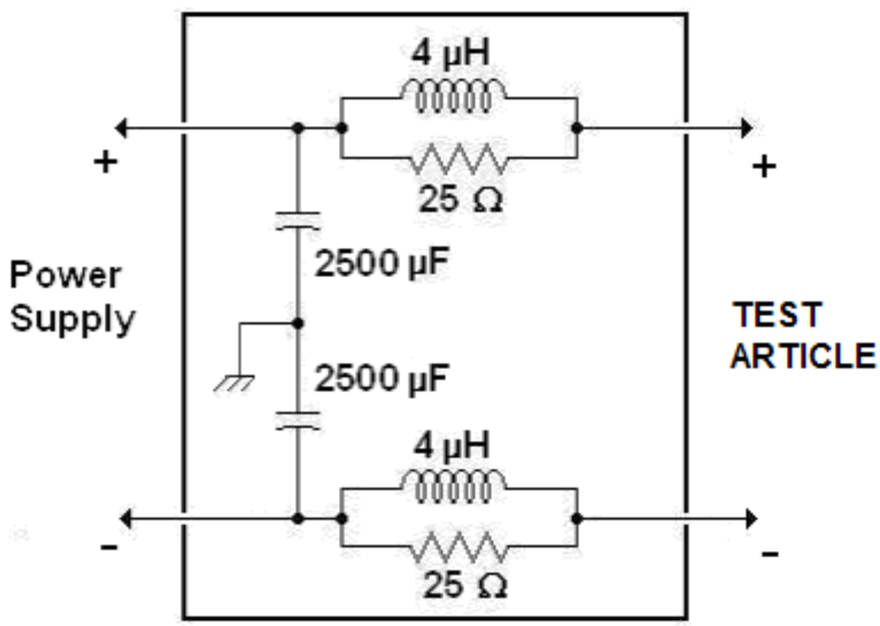


Figure4.5.7-1 Sample LISN Circuit Diagram.

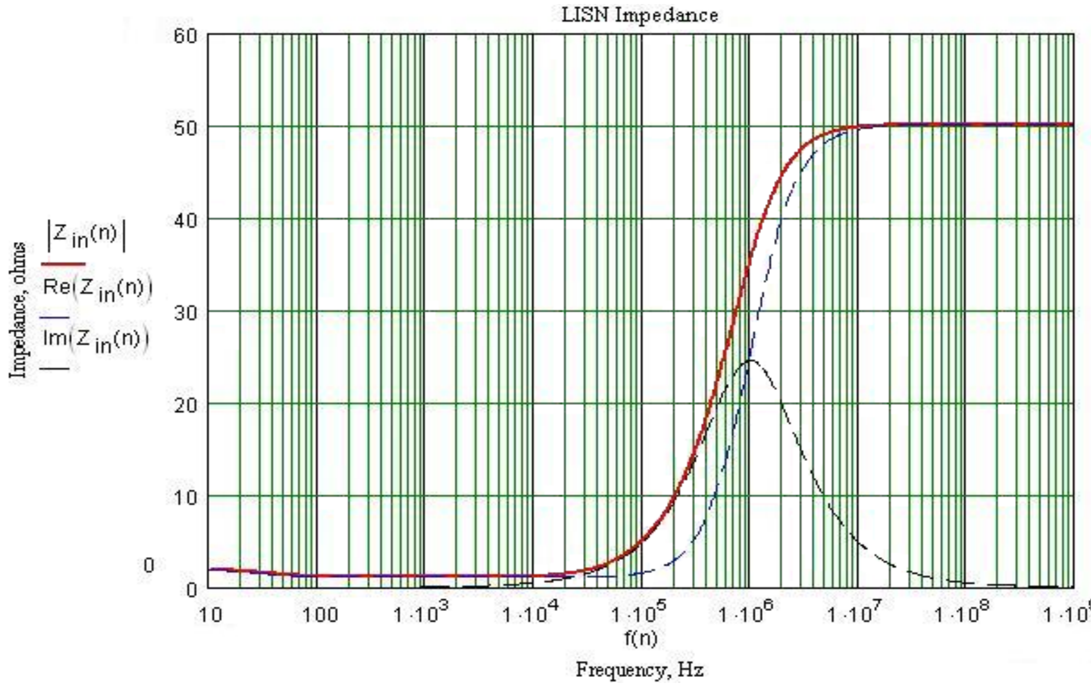


Figure 4.5.7-2 Sample LISN Terminal Impedance (input to test article).

Requirement: Emissions tests **shall** use the bandwidth and dwell time requirements of MIL-STD-461F, Section 4.3.10.3.

Requirement: The RE102 resolution bandwidth in notches **shall** be selected appropriately to bring the ambient level at least 6 dB below the RE102 Limit in the notch.

Note: It should be noted that the dwell times in 461F/Table II are minimum dwell times. Care must be taken to ensure the selected dwell time is long enough to ensure capturing the Payload emissions at each test frequency.

Requirement: The susceptibility scan rates and dwell time **shall** comply with MIL-STD-461F, Section 4.3.10.4.

Requirement: Instrument, subsystem, or assembly operational characteristics **shall** be reviewed carefully to make sure the dwell times are selected long enough to ensure the test article is tested in its worst-case operational mode.

Dwell times for conducted and radiated susceptibility tests may indeed be different.

Requirement: Appropriate Pass/Fail requirements and monitoring procedures **shall** be developed by the responsible EMI/EMC engineer for all tests.

Requirement: Instrument, subsystem, or assembly Ground Supply Equipment (GSE) **shall** have appropriate capabilities for susceptibility monitoring and non-compliance flagging.

Requirement: To the extent feasible, test cables **shall** simulate the configuration used on flight system, except for some CE and CS tests, where special test cables for Power lines may be required in accordance with the requirements in MIL-STD-461F.

Requirement: Isolation tests **shall** be performed on all circuits required to be isolated from circuit common or chassis ground during EMC testing.

Requirement: All isolation tests **shall** be performed with the test article unpowered and disconnected from the support equipment.

Requirement: The test article chassis **shall** be properly bonded to the EMC copper-topped test bench with resistance not to exceed 2.5 milli-ohms per joint.

Requirement: The test article Cognizant Engineer **shall** provide access to the necessary pins and connectors either through a non-flight connector saver or cable, or through a break-out box.

Requirement: All EMI/EMC testing **shall** be performed per an approved EMC test procedure.

Requirement: Test uncertainties **shall** not exceed 3 dB, which is the allowed uncertainty limit in MIL-STD-461F.

4.5.7.2 Test Tolerances

The measurement tolerances when performing EMI/EMC or ESD tests are defined below.

Requirement: The EMI/EMC and ESD tests **shall** be controlled to the measurement tolerances shown in Table 4.5.7-2.

Table 4.5.7-2 EMI/EMC and ESD Test Tolerances.

Line Voltage (DC or AC)	Within 10% of the target value
Line Current (DC or AC)	Within 10% of the target value
RF Radiated Amplitude	Within 3 dB of the target value
RF Conducted Amplitude	Within 3 dB of the target value
Frequency	Within 1% of the target value
Resistance	Within 10% of the target value
Distance	Within 5 cm or 10% of specified distance, whichever is greater
Magnetic Field Intensity	Within 3 dB of the target value

4.5.7.3 Over-Testing and Dry Runs

In general, when testing flight hardware, care must be taken to prevent subjecting the hardware to unsafe conditions, or exposing the hardware to excessive risk. If such situations should occur, it will become necessary to initiate potentially extensive and costly investigations to determine if any over-stress condition has occurred. The requirements outlined below are aimed at preventing such conditions and minimizing test-related risks.

Requirement: Over-testing of instruments, subsystems, or assemblies during susceptibility testing **shall** be avoided to the greatest extent feasible.

Requirement: The uncertainty limit in MIL-STD-461F is 3 dB. This limit **shall** be used as the maximum test level allowed above the requirement level in RS and CS tests.

Requirement: The EMI/EMC testing organization **shall** implement safety and appropriate test procedures that minimize the risk of over-testing.

Requirement: Dry runs **shall** be used for all susceptibility tests to verify the test method is working correctly and at minimal over-test risk before applying the test to flight hardware.

Requirement: Flight hardware susceptibility tests **shall** be allowed only after successful completion of the respective dry runs.

4.5.8 Flight System Charging and Electrostatic Discharge Requirements

The Europa Lander flight system will encounter the Jovian radiation and plasma environment and therefore is subject to both surface and internal charging and discharge events. Additionally, the Lander may experience an electrostatic discharge (ESD) event upon touchdown.

The Europa Lander Surface Charging/iESD Control Plan (JPL D-97636) will contain project policies and design guidance on how to handle the Europa charging environment.

4.5.8.1 Surface Charging and ESD [TBR]

Every flight system configuration is susceptible to the surface charging and electrostatic discharge. Only the Lander stage and Descent stage are potentially susceptible to a touchdown-induced discharge.

Requirement: The flight system **shall** be designed to survive and operate within specifications under the surface charging environment in Figure 4.5.8-1, Figure 4.5.8-2, Figure 4.5.8-3, and Figure 4.5.8-4 with and without sunlight.

Note: The touchdown-induced discharge environment will be presented in an update to this document.

The magnetosphere is the primary controlling factor for local plasma environments. The magnetosphere of Jupiter is dominated by three factors: the magnetic field tilt (11°) relative to its spin axis, its rapid rotation, and the Jovian moon Io at 5.9 Rj. Io generates a vast torus of gas and ions. The more rapid rotation of Jupiter's magnetic field forces the cold plasma associated with this torus to accelerate and expand by centrifugal force into a giant disc. The magnetic field tilt and rotation rate cause the plasma disc to wave up and down so that at a given location plasma parameters vary radically during a 10 h period. Jupiter's environment can be roughly divided into three populations: the cold plasma associated with the Io torus and the plasma disc (< 1 keV), the intermediate "warm" plasma (1 keV – 100 keV), and the trapped radiation environment (>100 keV). The trapped radiation environment is discussed in detailed in Section 4.9. Surface charging described in this subsection is governed primarily by the cold and warm plasma environments near Jupiter.

Jovian plasma environment can be expressed as the sum of Maxwell-Boltzmann distribution(s) and Kappa distribution(s).

The Maxwell-Boltzmann distribution:

$$f_i(v) = \frac{N_i}{\pi^{3/2} v_0^3} \exp(-v^2 / v_0^2) \qquad \text{(Eq. 1)}$$

where:

$v_0 = (2kT/m)^{1/2}$
v = velocity relative to observation point; the convection velocity v_{cnvc} can be incorporated here
N_i = number density of species e^-, H^+, O^+, O^{++}, S^+, S^{++}, S^{+++}, Na^+ (i=0,1,2,..,7)
or e^-(warm) and H^+(warm)
m = mass of species
k = Boltzmann constant
T = temperature of species

The Kappa distribution:

$$f_\kappa(E) = N_\kappa (m_\kappa / 2\pi E_0)^{3/2} \kappa^{-3/2} \frac{\Gamma(\kappa+1)}{\Gamma(\kappa-1/2)} \frac{1}{(1+E/\kappa E_0)^{\kappa+1}} \quad \text{(Eq. 2)}$$

where:

$E = ½mv^2$
N_κ = Kappa number density (cm^{-3}) of species (e- and H+)
m_κ = Kappa mass (g) of species (e- and H+)
κ = Kappa value
E_0 = Kappa temperature or characteristic energy

The plasma parameters from the new Divine-Garrett (DG2) plasma model at 9.5 Rj, 110° West Longitude, and 0° Latitude are:

Electrons:

$N_e = 46.05\ cm^{-3}$,	$T = 100.8$ eV		cold electron
$N_e = 3.278\ cm^{-3}$,	$T = 3720.$ eV		warm electron
$N_\kappa = 0.5666\ cm^{-3}$,	$E_0 = 949.0$ eV,	$\kappa = 1.617$	Kappa electron

Protons:

$N_p = 6.557\ cm^{-3}$,	$T = 143.1$ eV		cold proton
$N_p = 3.278\ cm^{-3}$,	$T = 30000$ eV		warm proton.
$N_\kappa = 4.167\ cm^{-3}$,	$E_0 = 32420$ eV,	$\kappa = 3.368$	Kappa proton

Ions:

$N_T = 46.06\ cm^{-3}$,	$T = 143.1$ eV,	v_{cnvc}= 112.6 km/s	total ions
$N_{O+} = 3.224\ cm^{-3}$, $N_{O2+} = 2.763\ cm^{-3}$,		$N_{Na} = 2.303\ cm^{-3}$	O^+, O^{2+}, Na^+
$N_{S+} = 2.763\ cm^{-3}$, $N_{S2+} = 11.97\ cm^{-3}$,		$N_{S3+} = 2.763\ cm^{-3}$	S^+, S^{2+}, S^{3+}

The plasma distribution functions for each of the components have been converted to Differential Fluxes and are plotted in Figure 4.5.8-1, Figure 4.5.8-2, and Figure 4.5.8-3. The specific components are as follows for the cold (Boltzmann) electron, proton, and heavy ions and for the warm (Boltzmann and Kappa) electron and proton distribution functions versus energy (note: for the cold proton and ion components, one can include the effects of the convection velocity, v_{cnvc} – see Eq. 1).

The legend labels are defined as follows:

DFE1 = Differential Flux for cold electrons
DFE2 = Differential Flux for warm electrons
DFE3 = Differential Flux for high energy electrons (E> 100 keV)
DFE4 = Differential Flux for Kappa electrons

DFH1 = Differential Flux for cold protons
DFH2 = Differential Flux for warm protons
DFH3 = Differential Flux for high energy protons (E> 0.6 MeV)
DFH4 = Differential Flux for Kappa protons

DFS1 = Differential Flux for S^+
DFS2 = Differential Flux for S^{++}
DFS3 = Differential Flux for S^{+++}
DFO1 = Differential Flux for O^+
DFO2 = Differential Flux for O^{++}
DFNA = Differential Flux for Na^+
EPD = Galileo Energetic Particle Detector measurements

The individual Differential Flux spectra in units of $(cm^2\ s\ sr\ eV)^{-1}$ are given in Figure 4.5.8-1, Figure 4.5.8-2, and Figure 4.5.8-3.

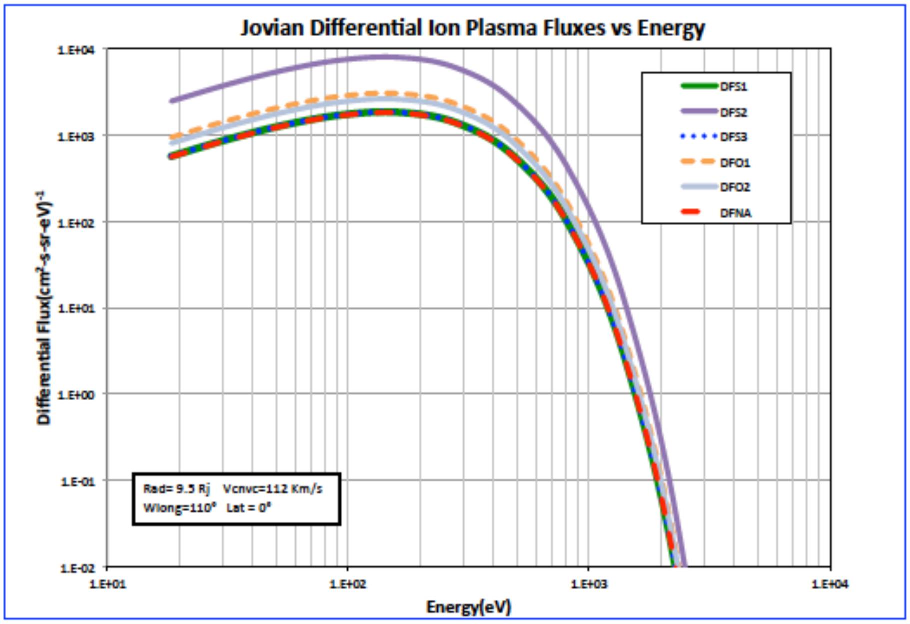


Figure 4.5.8-1 Heavy Ion Fluxes versus Energy at 9.5 Rj. (Note: S+, S2+, and Na+ overlay each other.)

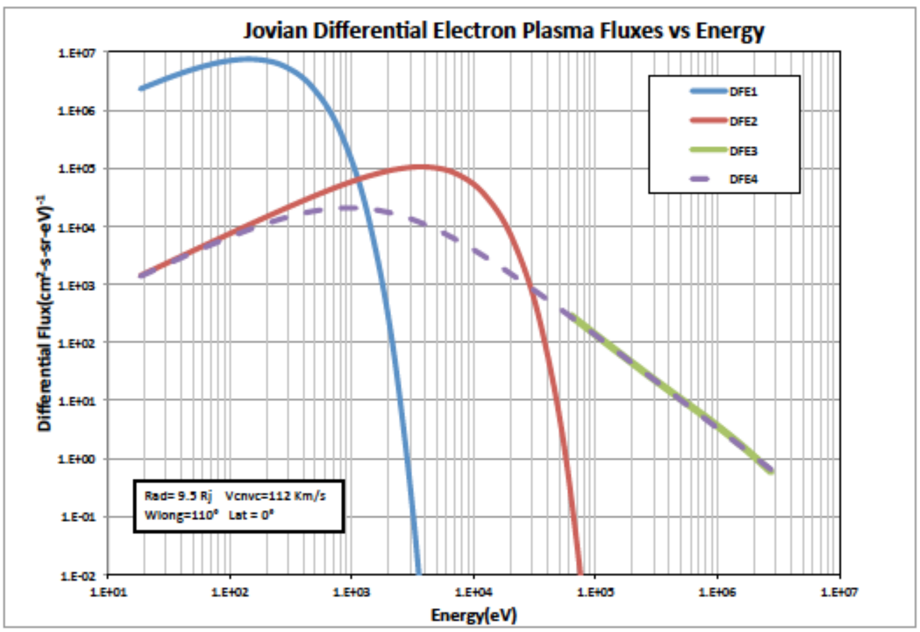


Figure 4.5.8-2 Electron Fluxes at 9.5 Rj. (Note that the Kappa distribution (DFE4) can be used in lieu of the Boltzmann distribution (DFE2) for the warm electrons.)

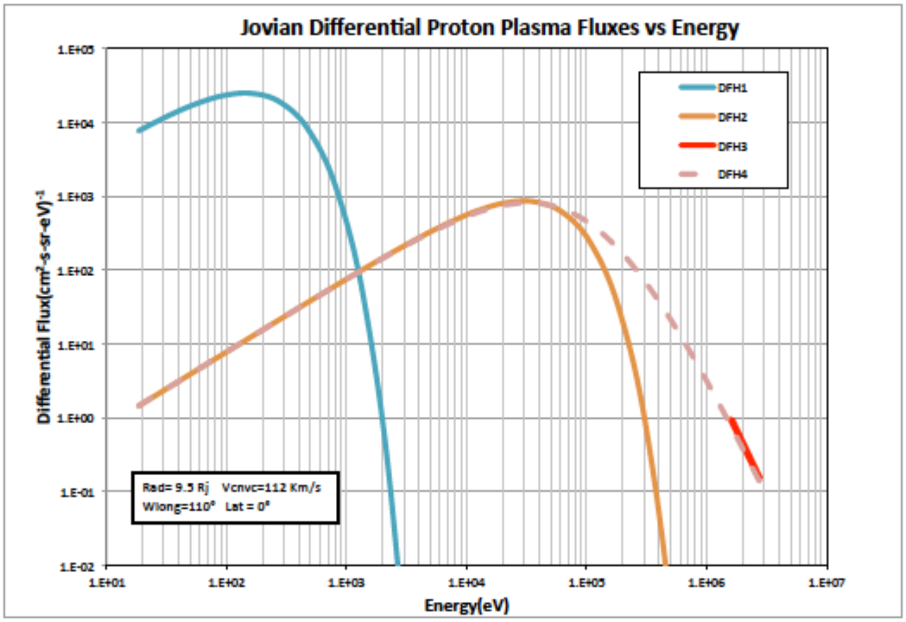
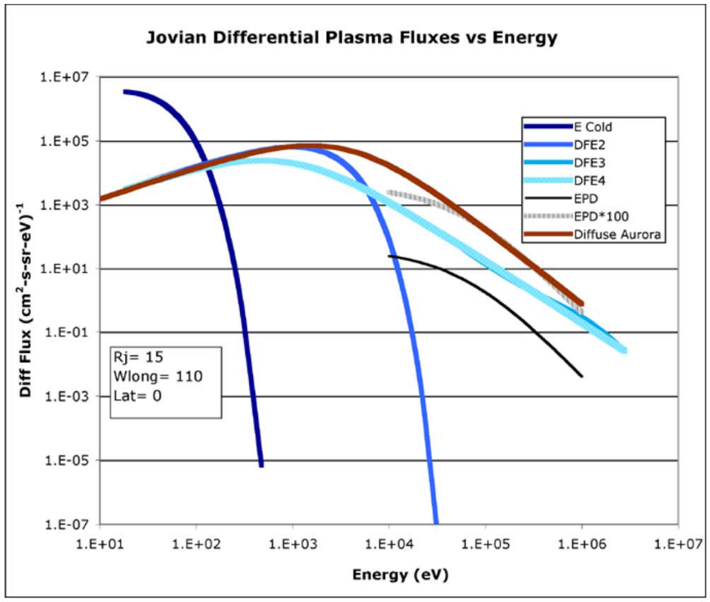
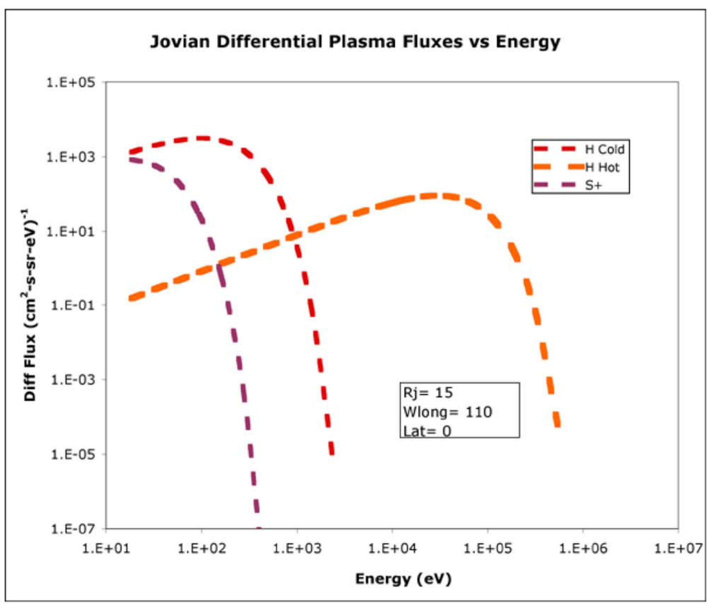


Figure 4.5.8-3 Proton Fluxes at 9.5 Rj. (Note that the Kappa distribution (DFH4) can be used in lieu of the Boltzmann distribution (DFH2) for the warm protons. The cold proton component (DFH1) is the least well defined of all the ion components and can be disregarded if desired.)

The auroral contributions to the Jovian plasma environment are prevalent outside the orbit of Europa, with worst-case conditions expected at 15 Rj (near Ganymede orbit). The energy characteristics of the important plasma components in the equatorial plane at 15 Rj are plotted in Figure 4.5.8-4 below.



A.



B.

Figure 4.5.8-4 Plasma differential fluxes at 15 Rj, 110° W, and 0° (illustrating the various plasma components and their relative flux values in the jovian plasma sheet). E Cold, H Cold, and H Hot are DFE1, DFH1, and DFH2, respectively. 100 times of EPD (normal) flux is used for high-energy electron flux estimation during Auroral events.

The plasma parameters at Rj = 15 are:

Electrons:

N_e = 6.91 cm^{-3},	T = 15 eV		cold electron
N_e = 1.024 cm^{-3},	T = 1000 eV		warm electron
N_κ = 0.4234cm^{-3},	E_0 = 475 eV,	κ = 1.95	Kappa electron
N_κ = 2.00 cm^{-3},	E_0 = 1500 eV,	κ = 2.4	Diffuse Aurora

Protons:

N_p = 0.683 cm^{-3},	T = 15 eV		cold proton

	N_p = 0.341 cm^{-3},	T = 30000 eV		warm proton
Ions:				
	N_T = 6.91 cm^{-3},	T = 15 eV ,	v_{cnvc}= 158 km/s	total ions
	N_{O+} = 0.48 cm^{-3}, N_{O2+} = 0.41 cm^{-3},		N_{Na} = 0.35 cm^{-3}	O^+, O^{2+}, Na^+
	N_{S+} = 0.41 cm^{-3}, N_{S2+} = 1.80 cm^{-3},		N_{S3+} = 0.41 cm^{-3}	S^+, S^{2+}, S^{3+}

Implication of plasma environment for worst-case surface potentials on shadowed surfaces is up to -5 kV. [Jovian plasma environmental models are under review with updates expected for further revisions].

Details of relevant surface charging guidelines will be described in Europa Lander Surface Charging/iESD Control Plan [JPL D-97636].

NASA-HDBK-4002A also contains design guidelines for this environment and are only to be used as references and do not constitute any design requirements.

4.5.8.2 Internal Charging and Electrostatic Discharge (ESD) [TBR]

Internal Electrostatic Discharge (IESD) is caused when energetic particles penetrate into a material, stop, and deposit their charge. Net charge can build up on floating conductors and within dielectrics. If the net charge induces a potential that surpasses the local breakdown voltage, there will be a discharge. These discharges occur when the current from the energetic particle environment is greater than the bleed-off current of a floating conductor or dielectric material.

IESD mitigation methods include shielding of sensitive components, eliminating floating conductors, providing sufficient bleed paths from sensitive components, hardening electronics to the effects of discharges, and use of leaky dielectric materials.

The Europa Lander Flight System will be exposed to the most extreme IESD environment of any spacecraft to date. It is anticipated that multiple mitigation methods and a concerted design effort will be required to ensure mission success.

Requirement: The flight system **shall** be designed to survive and operate within specifications under the IESD environment derived from the energetic particle fluence in Table 4.7.1-1, Table 4.7.1-2, and Table 4.7.1-3 (depending on flight element).

Details of relevant internal charging requirements will be described in Europa Lander Surface Charging/IESD Control Plan [JPL D-97636].

NASA-HDBK-4002A also contains design guidelines for this environment and are only to be used as references and do not constitute any design requirements.

4.5.8.3 v x B Effects on Charging

The co-rotating plasma near Europa reaches a velocity of ~120 km/s, while the local magnetic field is approximately 400 nT (4 mGauss). These conditions lead to ~0.05V/m due to **v x B** effects. The charging effect due to **v x B** is expected to be insignificant.

4.6 Solid Particle Environments

4.6.1 Meteoroid Environment

Requirement: The flight system **shall** be designed to have at least a 95% probability of surviving the meteoroid environment summarized in Table 4.6.1-1 and Table 4.6.1-2 to achieve minimum mission success criteria. [**TBR**]

The meteoroid environment presented was generated using an EVEEGA trajectory with an 8-year cruise originally studied for the Europa Clipper mission (JPL IOM 5132-14-054A). This estimate gives an upper bound of the meteoroid environment due to longer mission duration. This environment will be updated as mission design scenarios mature. The fluence versus mass and speed distribution is shown in Table 4.6.1-1 and Table 4.6.1-2. The micrometeoroid fluence distrubustions were calculated using the MSFC's Meteoroid Environment Model (MEM) inside of 2.3 AU and JPL's METeoroid Environment Model (METEM) outside of 2.3 AU. Dust and other solid particles within Jupiter's rings are located within the orbit of Io (~5.9R_J) and excluded from this estimate.

Table 4.6.1-1 Mission fluence for EVEEGA trajectory from Launch to 2.3 AU.

	Fluence (m^{-2}) above the given Particle Mass, within a given 1km/s Velocity bin. Particle material density is assumed to be 1 g/cm^3.					
Velocity [bin avg.] (m/s)	**1.00E-06g**	**1.00E-05g**	**1.00E-04g**	**1.00E-03g**	**1.00E-02g**	**1.00E-01g**
500	1.14E-01	1.13E-02	7.96E-04	4.59E-05	2.36E-06	1.15E-07
1500	1.14E-01	1.13E-02	7.96E-04	4.59E-05	2.36E-06	1.15E-07
2500	1.14E-01	1.13E-02	7.96E-04	4.59E-05	2.36E-06	1.15E-07
3500	1.14E-01	1.13E-02	7.96E-04	4.59E-05	2.36E-06	1.15E-07
4500	1.14E-01	1.13E-02	7.96E-04	4.59E-05	2.36E-06	1.15E-07
5500	5.88E-01	5.79E-02	4.09E-03	2.36E-04	1.21E-05	5.89E-07
6500	5.88E-01	5.79E-02	4.09E-03	2.36E-04	1.21E-05	5.89E-07
7500	5.88E-01	5.79E-02	4.09E-03	2.36E-04	1.21E-05	5.89E-07
8500	5.88E-01	5.79E-02	4.09E-03	2.36E-04	1.21E-05	5.89E-07
9500	5.88E-01	5.79E-02	4.09E-03	2.36E-04	1.21E-05	5.89E-07
10500	8.15E-01	8.02E-02	5.67E-03	3.27E-04	1.68E-05	8.17E-07
11500	8.15E-01	8.02E-02	5.67E-03	3.27E-04	1.68E-05	8.17E-07
12500	8.15E-01	8.02E-02	5.67E-03	3.27E-04	1.68E-05	8.17E-07
13500	8.15E-01	8.02E-02	5.67E-03	3.27E-04	1.68E-05	8.17E-07
14500	8.15E-01	8.02E-02	5.67E-03	3.27E-04	1.68E-05	8.17E-07
15500	7.90E-01	7.78E-02	5.50E-03	3.17E-04	1.63E-05	7.92E-07
16500	7.90E-01	7.78E-02	5.50E-03	3.17E-04	1.63E-05	7.92E-07
17500	7.90E-01	7.78E-02	5.50E-03	3.17E-04	1.63E-05	7.92E-07
18500	7.90E-01	7.78E-02	5.50E-03	3.17E-04	1.63E-05	7.92E-07
19500	7.90E-01	7.78E-02	5.50E-03	3.17E-04	1.63E-05	7.92E-07
20500	7.22E-01	7.11E-02	5.02E-03	2.90E-04	1.49E-05	7.24E-07
21500	7.22E-01	7.11E-02	5.02E-03	2.90E-04	1.49E-05	7.24E-07
22500	7.22E-01	7.11E-02	5.02E-03	2.90E-04	1.49E-05	7.24E-07
23500	7.22E-01	7.11E-02	5.02E-03	2.90E-04	1.49E-05	7.24E-07
24500	7.22E-01	7.11E-02	5.02E-03	2.90E-04	1.49E-05	7.24E-07
25500	6.16E-01	6.07E-02	4.29E-03	2.47E-04	1.27E-05	6.18E-07
26500	6.16E-01	6.07E-02	4.29E-03	2.47E-04	1.27E-05	6.18E-07
27500	6.16E-01	6.07E-02	4.29E-03	2.47E-04	1.27E-05	6.18E-07
28500	6.16E-01	6.07E-02	4.29E-03	2.47E-04	1.27E-05	6.18E-07
29500	6.16E-01	6.07E-02	4.29E-03	2.47E-04	1.27E-05	6.18E-07
30500	4.60E-01	4.53E-02	3.20E-03	1.85E-04	9.50E-06	4.61E-07
31500	4.60E-01	4.53E-02	3.20E-03	1.85E-04	9.50E-06	4.61E-07
32500	4.60E-01	4.53E-02	3.20E-03	1.85E-04	9.50E-06	4.61E-07
33500	4.60E-01	4.53E-02	3.20E-03	1.85E-04	9.50E-06	4.61E-07
34500	4.60E-01	4.53E-02	3.20E-03	1.85E-04	9.50E-06	4.61E-07
35500	2.95E-01	2.90E-02	2.05E-03	1.18E-04	6.09E-06	2.95E-07
36500	2.95E-01	2.90E-02	2.05E-03	1.18E-04	6.09E-06	2.95E-07
37500	2.95E-01	2.90E-02	2.05E-03	1.18E-04	6.09E-06	2.95E-07
38500	2.95E-01	2.90E-02	2.05E-03	1.18E-04	6.09E-06	2.95E-07
39500	2.95E-01	2.90E-02	2.05E-03	1.18E-04	6.09E-06	2.95E-07
40500	1.59E-01	1.56E-02	1.10E-03	6.36E-05	3.28E-06	1.59E-07
41500	1.59E-01	1.56E-02	1.10E-03	6.36E-05	3.28E-06	1.59E-07
42500	1.59E-01	1.56E-02	1.10E-03	6.36E-05	3.28E-06	1.59E-07
43500	1.59E-01	1.56E-02	1.10E-03	6.36E-05	3.28E-06	1.59E-07
44500	1.59E-01	1.56E-02	1.10E-03	6.36E-05	3.28E-06	1.59E-07
45500	7.44E-02	7.33E-03	5.18E-04	2.99E-05	1.54E-06	7.46E-08
46500	7.44E-02	7.33E-03	5.18E-04	2.99E-05	1.54E-06	7.46E-08
47500	7.44E-02	7.33E-03	5.18E-04	2.99E-05	1.54E-06	7.46E-08
48500	7.44E-02	7.33E-03	5.18E-04	2.99E-05	1.54E-06	7.46E-08
49500	7.44E-02	7.33E-03	5.18E-04	2.99E-05	1.54E-06	7.46E-08

50500	3.40E-02	3.35E-03	2.37E-04	1.36E-05	7.03E-07	3.41E-08
51500	3.40E-02	3.35E-03	2.37E-04	1.36E-05	7.03E-07	3.41E-08
52500	3.40E-02	3.35E-03	2.37E-04	1.36E-05	7.03E-07	3.41E-08
53500	3.40E-02	3.35E-03	2.37E-04	1.36E-05	7.03E-07	3.41E-08
54500	3.40E-02	3.35E-03	2.37E-04	1.36E-05	7.03E-07	3.41E-08
55500	1.90E-02	1.87E-03	1.32E-04	7.63E-06	3.93E-07	1.91E-08
56500	1.90E-02	1.87E-03	1.32E-04	7.63E-06	3.93E-07	1.91E-08
57500	1.90E-02	1.87E-03	1.32E-04	7.63E-06	3.93E-07	1.91E-08
58500	1.90E-02	1.87E-03	1.32E-04	7.63E-06	3.93E-07	1.91E-08
59500	1.90E-02	1.87E-03	1.32E-04	7.63E-06	3.93E-07	1.91E-08
60500	1.28E-02	1.26E-03	8.93E-05	5.15E-06	2.65E-07	1.29E-08
61500	1.28E-02	1.26E-03	8.93E-05	5.15E-06	2.65E-07	1.29E-08
62500	1.28E-02	1.26E-03	8.93E-05	5.15E-06	2.65E-07	1.29E-08
63500	1.28E-02	1.26E-03	8.93E-05	5.15E-06	2.65E-07	1.29E-08
64500	1.28E-02	1.26E-03	8.93E-05	5.15E-06	2.65E-07	1.29E-08
65500	8.00E-03	7.88E-04	5.57E-05	3.21E-06	1.65E-07	8.02E-09
66500	8.00E-03	7.88E-04	5.57E-05	3.21E-06	1.65E-07	8.02E-09
67500	8.00E-03	7.88E-04	5.57E-05	3.21E-06	1.65E-07	8.02E-09
68500	8.00E-03	7.88E-04	5.57E-05	3.21E-06	1.65E-07	8.02E-09
69500	8.00E-03	7.88E-04	5.57E-05	3.21E-06	1.65E-07	8.02E-09
70500	4.42E-03	4.35E-04	3.07E-05	1.77E-06	9.13E-08	4.43E-09
71500	4.42E-03	4.35E-04	3.07E-05	1.77E-06	9.13E-08	4.43E-09
72500	4.42E-03	4.35E-04	3.07E-05	1.77E-06	9.13E-08	4.43E-09
73500	4.42E-03	4.35E-04	3.07E-05	1.77E-06	9.13E-08	4.43E-09
74500	4.42E-03	4.35E-04	3.07E-05	1.77E-06	9.13E-08	4.43E-09
75500	1.66E-03	1.63E-04	1.15E-05	6.65E-07	3.43E-08	1.66E-09
76500	1.66E-03	1.63E-04	1.15E-05	6.65E-07	3.43E-08	1.66E-09
77500	1.66E-03	1.63E-04	1.15E-05	6.65E-07	3.43E-08	1.66E-09
78500	1.66E-03	1.63E-04	1.15E-05	6.65E-07	3.43E-08	1.66E-09
79500	1.66E-03	1.63E-04	1.15E-05	6.65E-07	3.43E-08	1.66E-09
Total	**2.36E+01**	**2.32E+00**	**1.64E-01**	**9.45E-03**	**4.87E-04**	**2.36E-05**

Table 4.6.1-2 Mission fluence for EVEEGA trajectory past 2.3 AU to End of Mission.

	Fluence (m^{-2}) above the given Particle Mass, within a given 1km/s Velocity bin. Particle material density is assumed to be 2.5 g/cm^3.					
Velocity [bin avg.] (m/s)	**1.00E-06g**	**1.00E-05g**	**1.00E-04g**	**1.00E-03g**	**1.00E-02g**	**1.00E-01g**
500	2.00E-03	4.04E-04	7.36E-05	6.52E-06	3.71E-07	1.77E-08
1500	8.72E-03	2.52E-03	5.40E-04	4.92E-05	2.82E-06	1.35E-07
2500	2.83E-02	1.19E-02	2.80E-03	2.59E-04	1.49E-05	7.12E-07
3500	7.35E-02	3.51E-02	8.47E-03	7.86E-04	4.52E-05	2.16E-06
4500	1.33E-01	6.50E-02	1.58E-02	1.47E-03	8.42E-05	4.03E-06
5500	2.15E-01	1.08E-01	2.64E-02	2.45E-03	1.41E-04	6.74E-06
6500	2.91E-01	1.47E-01	3.60E-02	3.34E-03	1.92E-04	9.20E-06
7500	3.57E-01	1.79E-01	4.37E-02	4.06E-03	2.33E-04	1.12E-05
8500	4.15E-01	2.07E-01	5.05E-02	4.70E-03	2.70E-04	1.29E-05
9500	4.74E-01	2.37E-01	5.80E-02	5.39E-03	3.10E-04	1.48E-05
10500	5.32E-01	2.65E-01	6.50E-02	6.05E-03	3.48E-04	1.66E-05
11500	5.95E-01	2.92E-01	7.14E-02	6.64E-03	3.82E-04	1.83E-05
12500	6.48E-01	3.11E-01	7.59E-02	7.05E-03	4.05E-04	1.94E-05
13500	6.46E-01	3.08E-01	7.53E-02	7.00E-03	4.02E-04	1.93E-05
14500	6.01E-01	2.84E-01	6.92E-02	6.44E-03	3.70E-04	1.77E-05
15500	5.63E-01	2.58E-01	6.28E-02	5.84E-03	3.35E-04	1.61E-05
16500	5.35E-01	2.34E-01	5.68E-02	5.27E-03	3.03E-04	1.45E-05
17500	5.12E-01	2.14E-01	5.16E-02	4.79E-03	2.75E-04	1.32E-05
18500	4.70E-01	1.85E-01	4.44E-02	4.12E-03	2.37E-04	1.13E-05
19500	4.31E-01	1.63E-01	3.90E-02	3.62E-03	2.08E-04	9.96E-06
20500	3.93E-01	1.40E-01	3.34E-02	3.10E-03	1.78E-04	8.51E-06
21500	3.54E-01	1.24E-01	2.95E-02	2.73E-03	1.57E-04	7.51E-06
22500	3.19E-01	1.09E-01	2.60E-02	2.41E-03	1.38E-04	6.62E-06
23500	2.86E-01	9.72E-02	2.32E-02	2.15E-03	1.23E-04	5.91E-06
24500	2.63E-01	8.81E-02	2.10E-02	1.95E-03	1.12E-04	5.35E-06
25500	2.46E-01	8.12E-02	1.93E-02	1.79E-03	1.03E-04	4.93E-06
26500	2.25E-01	7.42E-02	1.77E-02	1.64E-03	9.43E-05	4.51E-06
27500	2.07E-01	6.79E-02	1.62E-02	1.51E-03	8.64E-05	4.14E-06
28500	1.91E-01	6.24E-02	1.49E-02	1.38E-03	7.93E-05	3.80E-06
29500	1.76E-01	5.69E-02	1.35E-02	1.26E-03	7.22E-05	3.46E-06
30500	1.59E-01	5.15E-02	1.23E-02	1.14E-03	6.53E-05	3.13E-06
31500	1.39E-01	4.61E-02	1.10E-02	1.02E-03	5.85E-05	2.80E-06
32500	1.19E-01	4.04E-02	9.63E-03	8.93E-04	5.13E-05	2.46E-06
33500	1.01E-01	3.50E-02	8.33E-03	7.72E-04	4.44E-05	2.12E-06
34500	8.24E-02	2.86E-02	6.80E-03	6.31E-04	3.62E-05	1.73E-06
35500	6.24E-02	2.05E-02	4.83E-03	4.48E-04	2.57E-05	1.23E-06
36500	3.53E-02	8.47E-03	1.92E-03	1.77E-04	1.01E-05	4.85E-07
37500	1.96E-02	2.78E-03	5.75E-04	5.25E-05	3.01E-06	1.44E-07
38500	1.26E-02	8.06E-04	1.25E-04	1.10E-05	6.23E-07	2.97E-08
39500	9.32E-03	3.14E-04	2.35E-05	1.75E-06	9.54E-08	4.48E-09
40500	7.58E-03	2.03E-04	7.10E-06	3.24E-07	1.46E-08	6.28E-10
41500	6.33E-03	1.60E-04	4.17E-06	1.15E-07	3.31E-09	1.01E-10
42500	5.25E-03	1.32E-04	3.32E-06	8.34E-08	2.10E-09	5.29E-11
43500	4.41E-03	1.11E-04	2.78E-06	6.99E-08	1.76E-09	4.41E-11
44500	3.83E-03	9.63E-05	2.42E-06	6.08E-08	1.53E-09	3.83E-11
45500	3.34E-03	8.39E-05	2.11E-06	5.29E-08	1.33E-09	3.34E-11
46500	2.64E-03	6.62E-05	1.66E-06	4.18E-08	1.05E-09	2.64E-11
47500	1.70E-03	4.27E-05	1.07E-06	2.69E-08	6.76E-10	1.70E-11
48500	8.35E-04	2.10E-05	5.27E-07	1.32E-08	3.33E-10	8.35E-12

49500	3.03E-04	7.60E-06	1.91E-07	4.79E-09	1.20E-10	3.03E-12
50500	8.03E-05	2.02E-06	5.07E-08	1.27E-09	3.20E-11	8.03E-13
51500	9.46E-06	2.38E-07	5.97E-09	1.50E-10	3.77E-12	9.46E-14
52500	1.76E-07	4.43E-09	1.11E-10	2.79E-12	7.02E-14	1.76E-15
53500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
54500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
55500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
56500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
57500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
58500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
59500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
60500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
61500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
62500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
63500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
64500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
65500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
66500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
67500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
68500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
69500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
70500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
71500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
72500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
73500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
74500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
75500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
76500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
77500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
78500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
79500	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00	0.00E+00
Total	**1.10E+01**	**4.64E+00**	**1.12E+00**	**1.04E-01**	**6.00E-03**	**2.87E-04**

4.6.2 Europa Surface Solid Particle Environment

There are likely solid particles on the surface of Europa that can be energized by landing engines on the Descent Stage in such a way to disturb hardware that is external to the Lander stage body. A similar environment was observed during the Mars Science Laboratory touchdown and rocky particles were observed on the rover deck as shown in Figure 4.6.2-1.

Figure 4.6.2-1 Particulates on the Mars Science Laboratory rover deck after touchdown.

Requirement: External components of the Lander **shall** be designed to survive the solid particle particulate environment created during touchdown as described Table 4.6.2-1.

Table 4.6.2-1 Europa Lander solid particle environment [placeholder]. [TBD]

4.7 High-Energy Radiation Environments

4.7.1 Energetic Particle Fluence

For the Europa Lander Mission, the energetic particle fluence is the driving environment for all radiation effects. The following tables and figures give the Jupiter trapped particle fluences of a representative mission trajectory (12L04_50km) derived using the JPL Jupiter radiation models described in "Updating the Jovian Plasma and Radiation Environments: The Latest Results for 2015" by *Garrett, Kim, and Evans* [2016] (doi: 10.2514/1.A33510). Specifically, we use the Grid version of GIRE2p. Particle peak flux and fluence will change for different mission designs and could increase or decrease relative to the representative values found here.

Proton and electron mission integral fluences are listed Table 4.7.1-1, Table 4.7.1-2, and Table 4.7.1-3 for the Carrier and Relay Stage, Deorbit Vehicle, and the Lander, respectively. Figure 4.7.1-1 shows the mission integral fluences for the Carrier and Relay Stage, Deorbit Vehicle, and the Lander. Integral fluence is in units particles/cm^2 for energy equal to and greater than the energy listed. The highest energy electron fluences modeled (up to 150 MeV) are based upon power-law extrapolations of electrons with energies <30 MeV.

Unless otherwise stated, all tables and graphs within this section represent environments external to the Flight System, and do not contain a design factor (i.e., RDF=1). The Radiation Design Factor (RDF) is defined as:

$$\text{RDF} = \frac{\text{Radiation-tolerance level of a part or component in a given application}}{\text{Radiation environment present at the location of the part or component}}$$

The Cruise Vehicle (CV) is comprised of the Carrier and Relay Stage (CRS) and the Deorbit Vehicle (DOV). During interplanetary cruise, through Jupiter orbital insertion (JOI), all the way until DOV separation, the flight system is in a single, CV configuration and all system elements are exposed to the same radiation environment. Following DOV separation, the COS remains in orbit around Europa, but the orbital distance is assumed to be sufficiently large that Europa, itself, provides no radiation shielding. The DOV is also assumed to have no shielding from Europa. Once the Lander is on the surface, the fluence is assumed to be one half of the value at Europa's orbital distance due to shielding by Europa itself. After the surface mission is over, the CRS will be placed into a stable orbit for planetary protection purposes. These assumptions provide an upper bound on the energetic particle fluence and maintains the entire surface of Europa as an acceptable landing zone from a total dose perspective.

Requirement: The flight system **shall** be designed to survive and operate within specifications under the high-energy particle environment in Table 4.7.1-1, Table 4.7.1-2, and Table 4.7.1-3 (depending on the flight element) and, Table 4.7.1-4 (all flight elements).

Table 4.7.1-1 Carrier and Relay Stage Integral Electron and Proton Fluence [12L04_50km; RDF = 1].

Energy (MeV)	Electron Fluence (cm^{-2})	Proton Fluence (cm^{-2})
0.1	2.19E+15	3.18E+14
0.2	1.22E+15	2.43E+14
0.3	8.56E+14	1.96E+14
0.5	5.36E+14	1.43E+14
1	2.67E+14	8.31E+13
2	1.19E+14	4.04E+13
3	7.05E+13	2.29E+13
5	3.44E+13	8.77E+12
10	1.05E+13	1.47E+12
20	2.17E+12	1.82E+11
30	7.81E+11	5.22E+10
50	1.93E+11	1.08E+10
100	3.21E+10	1.32E+09

Table 4.7.1-2 Deorbit Vehicle integral electron and proton fluence [12L04_50km; RDF = 1].

Energy (MeV)	Electron Fluence (cm^{-2})	Proton Fluence (cm^{-2})
0.1	1.21E+15	1.54E+14
0.2	6.22E+14	1.09E+14
0.3	4.21E+14	8.44E+13
0.5	2.52E+14	5.76E+13
1	1.18E+14	3.10E+13
2	4.94E+13	1.42E+13
3	2.80E+13	7.87E+12
5	1.28E+13	2.99E+12
10	3.67E+12	5.19E+11
20	7.52E+11	6.91E+10
30	2.70E+11	2.07E+10
50	6.60E+10	4.59E+09
100	1.10E+10	6.11E+08

Table 4.7.1-3 Lander integral electron and proton fluence [12L04_50km; RDF = 1].

Energy (MeV)	Electron Fluence (cm^{-2})	Proton Fluence (cm^{-2})
0.1	1.39E+15	1.86E+14
0.2	7.31E+14	1.39E+14
0.3	5.03E+14	1.10E+14
0.5	3.08E+14	7.88E+13
1	1.50E+14	4.52E+13
2	6.56E+13	2.17E+13
3	3.85E+13	1.23E+13
5	1.86E+13	4.70E+12
10	5.63E+12	7.89E+11
20	1.16E+12	9.92E+10
30	4.15E+11	2.87E+10
50	1.02E+11	6.05E+09
100	1.70E+10	7.57E+08

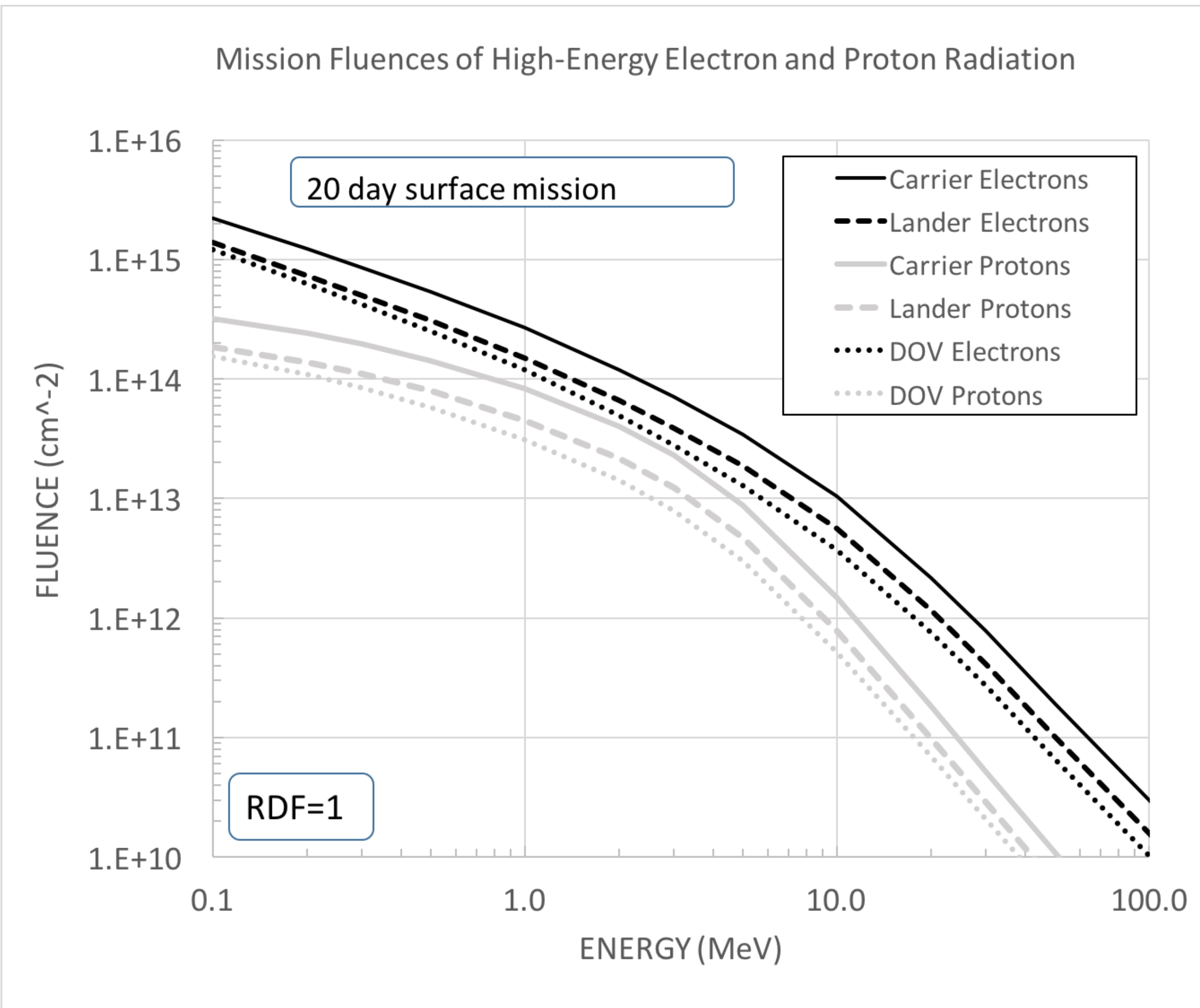


Figure 4.7.1-1 GIRE2p Integral proton and electron fluence [trajectory 12L4_50km; RDF =1].

An estimated Europa Lander Mission solar proton environment is based on the JPL-SPE model and shown in Table 4.7.1-4 and Figure 4.7.1-2.

Table 4.7.1-4 Solar proton fluence for interplanetary cruise from Earth to Jupiter applicable to the entire Cruise Vehicle. [RDF = 1]

Energy (MeV)	FLUENCE (protons/cm2) scaled 95%
1	1.90E+11
4	6.37E+10
10	2.46E+10
30	6.74E+09
60	3.03E+09
100	1.68E+09
150	1.05E+09
200	7.54E+08
300	4.72E+08
500	2.62E+08
1000	1.17E+08
2000	5.28E+07

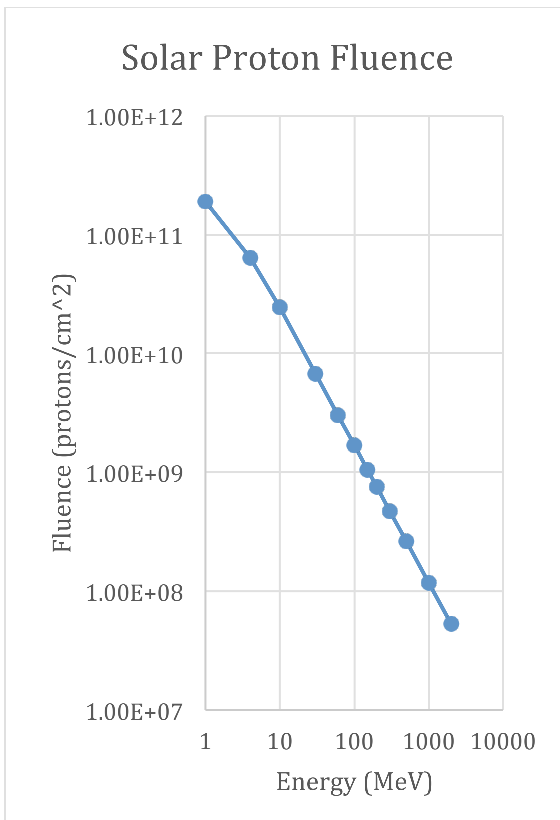


Figure 4.7.1-2 Solar proton integral fluence for interplanetary cruise from Earth to Jupiter applicable to the entire Cruise Vehicle. [RDF = 1]

4.7.2 Ionizing Radiation

The ionizing radiation exposure of Europa Lander flight hardware will come primarily from the Jovian radiation belt environment, and also from solar protons. Solar heavy ions and galactic cosmic rays, and Jovian heavy ions do not contribute significantly to the ionizing radiation exposure and can be ignored when assessing the of effects of ionizing radiation effects on parts and materials. The contribution from high-energy electrons and protons in Jovian radiation belts is expected to dominate for all hardware.

Requirement: Flight System, Subsystem, Assembly, and Instrument components and devices **shall** be selected such that they operate within performance specification during and after the exposure to the TID and DDD radiation environment documented herein at a radiation design factor (RDF) of 2 times the level present at the location of the device.

Requirement: Devices that require spot shielding **shall** be assessed at an RDF of 3 times the TID level present at the location of the device.

Note: The definition of spot shielding will be described in the Europa Lander Radiation Control Plan [JPL D-97634].

4.7.2.1 Total Ionizing Dose

Table 4.7.2-1, Table 4.7.2-2, and Table 4.7.2-3 list the total ionizing dose using NOVICE (Thomas Jordan 1993, NOVICE, A Radiation Transport Shielding Code, Experimental and Mathematical Physics Consultants, Jan. 2006 Version) for the mission fluence of each flight system element specified in Section 4.7.1.

Figure 4.7.2-1, Figure 4.7.2-1, and Figure 4.7.2-3-show the total dose for aluminum and tantalum spherical shell shield thicknesses for each flight system stage, with the relative dose contributions from electrons, secondary bremsstrahlung photons, trapped protons, and solar protons based on NOVICE transport calculations for the mission fluences for trajectory 12L04_50km. Dose is in rad(Si), and shielding thickness is in mil.
The flux of galactic cosmic rays (GCR) contributes approximately 0.02 rad(Si)/day to the mission dose, independent of shielding. The Jovian magnetic field provides some shielding from GCRs, with increasing protection closer to Jupiter. For the Europa Lander Mission, GCR contribution to TID will be neglected.

Table 4.7.2-1 Total ionizing dose-depth table as a function of spherical shell aluminum shielding thickness for the Carrier and Relay Stage for trajectory 12L04_50km. RDF = 1. [TBR]

	Total Ionizing Dose (TID), rad-Si			
mils Aluminum	bremsstrahlung	electron	proton	Total
0.10	9.64E+01	6.69E+07	7.97E+08	8.64E+08
1	2.63E+03	9.37E+07	1.31E+08	2.25E+08
10	9.90E+03	2.11E+07	7.42E+06	2.85E+07
20	1.07E+04	1.21E+07	2.01E+06	1.42E+07
30	1.08E+04	8.67E+06	9.61E+05	9.64E+06
40	1.07E+04	6.75E+06	5.21E+05	7.28E+06
50	1.06E+04	5.52E+06	3.10E+05	5.84E+06
60	1.08E+04	4.66E+06	2.21E+05	4.90E+06
70	1.07E+04	4.01E+06	1.57E+05	4.17E+06
80	1.07E+04	3.53E+06	1.17E+05	3.65E+06
90	1.09E+04	3.12E+06	9.14E+04	3.22E+06
100	1.08E+04	2.80E+06	7.09E+04	2.88E+06
120	1.09E+04	2.31E+06	4.85E+04	2.37E+06
140	1.10E+04	1.95E+06	3.56E+04	1.99E+06
160	1.11E+04	1.67E+06	2.64E+04	1.71E+06
180	1.12E+04	1.45E+06	2.02E+04	1.48E+06
200	1.14E+04	1.27E+06	1.60E+04	1.30E+06
220	1.15E+04	1.13E+06	1.28E+04	1.16E+06
240	1.17E+04	1.01E+06	1.08E+04	1.03E+06
260	1.19E+04	9.12E+05	8.87E+03	9.33E+05
280	1.19E+04	8.26E+05	7.55E+03	8.46E+05
300	1.21E+04	7.50E+05	6.52E+03	7.69E+05
320	1.22E+04	6.83E+05	5.53E+03	7.01E+05
400	1.26E+04	4.86E+05	3.48E+03	5.02E+05
500	1.29E+04	3.40E+05	2.11E+03	3.55E+05
600	1.30E+04	2.47E+05	1.51E+03	2.62E+05
700	1.30E+04	1.81E+05	1.11E+03	1.95E+05
800	1.29E+04	1.37E+05	8.29E+02	1.51E+05
900	1.28E+04	1.05E+05	6.60E+02	1.19E+05
1000	1.27E+04	8.32E+04	5.40E+02	9.64E+04
1100	1.25E+04	6.67E+04	4.55E+02	7.97E+04
1200	1.29E+04	5.39E+04	4.00E+02	6.71E+04
1300	1.29E+04	4.40E+04	3.60E+02	5.73E+04
1400	1.27E+04	3.62E+04	3.70E+02	4.93E+04
1500	1.25E+04	3.01E+04	3.41E+02	4.29E+04
1600	1.22E+04	2.51E+04	2.13E+02	3.75E+04
1700	1.18E+04	2.07E+04	1.33E+02	3.27E+04
1800	1.15E+04	1.74E+04	9.14E+01	2.90E+04
1900	1.13E+04	1.44E+04	6.03E+01	2.58E+04
2000	1.10E+04	1.20E+04	3.64E+01	2.30E+04
3000	8.56E+03	1.26E+03	2.78E+00	9.83E+03
4000	6.87E+03	5.34E+01	3.78E-01	6.92E+03
5000	5.41E+03	1.99E+00	8.22E-02	5.41E+03
6000	4.34E+03	1.09E-01	2.23E-02	4.34E+03
8000	2.91E+03	3.74E-04	2.84E-03	2.91E+03
10000	2.12E+03	5.56E-07	0.00E+00	2.12E+03

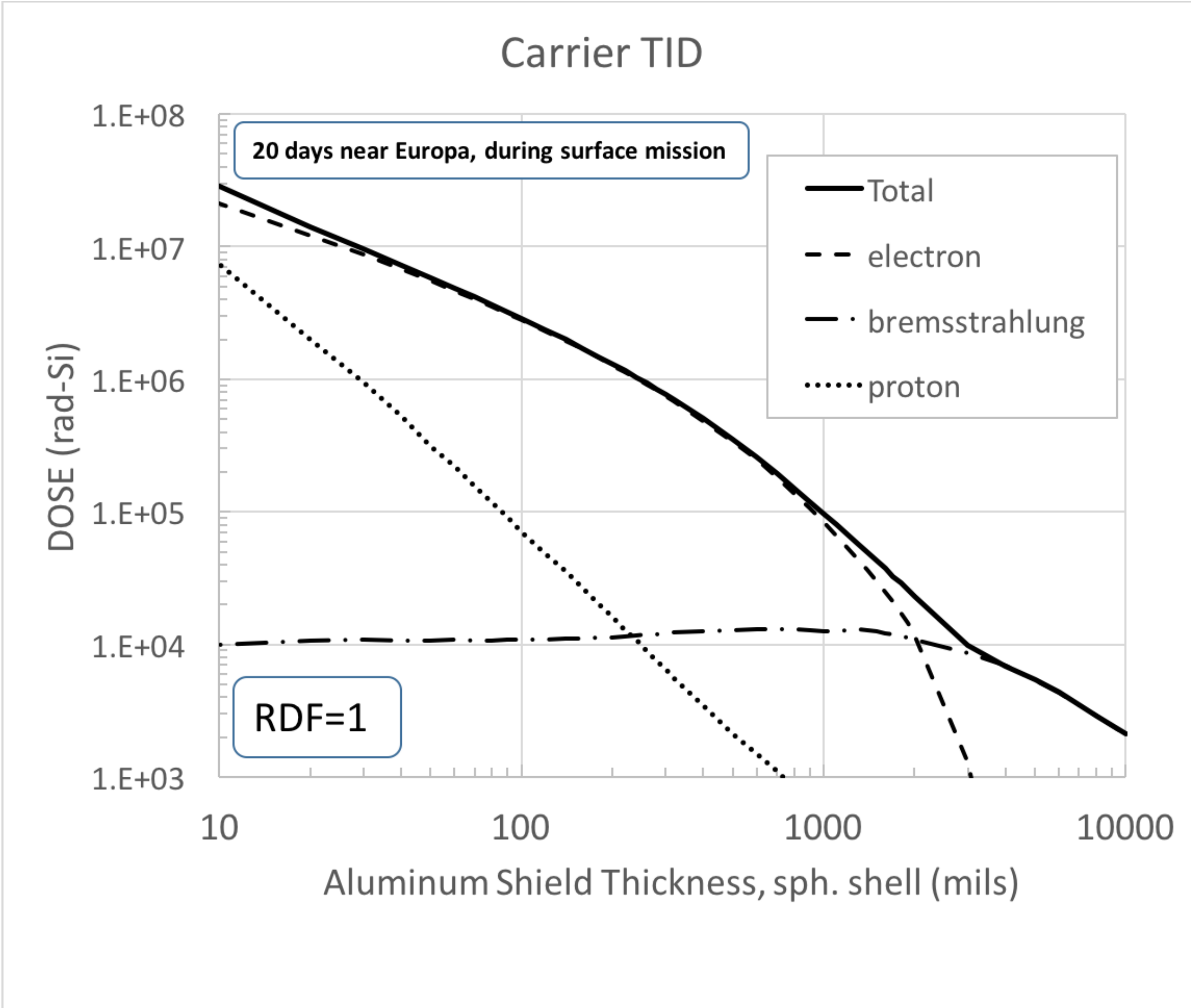


Figure 4.7.2-1 Ionizing dose-depth curves as a function of spherical shell aluminum shielding thickness for the Carrier and Relay Stage for trajectory 12L04_50km. RDF = 1 [TBR]

Table 4.7.2-2 Total ionizing dose-depth table as a function of spherical shell aluminum shielding thickness for the Descent Stage and Deorbit Stage for trajectory 12L04_50km. RDF = 1. [TBR]

	Total Ionizing Dose (TID), rad-Si			
mils Aluminum	bremsstrahlung	electron	proton	Total
0.10	5.78E+01	3.80E+07	3.69E+08	4.07E+08
1	1.53E+03	5.41E+07	4.88E+07	1.03E+08
10	5.16E+03	1.03E+07	2.53E+06	1.29E+07
20	5.31E+03	5.67E+06	6.99E+05	6.37E+06
30	5.18E+03	3.93E+06	3.38E+05	4.28E+06
40	5.02E+03	3.00E+06	1.86E+05	3.19E+06
50	4.88E+03	2.41E+06	1.12E+05	2.53E+06
60	4.87E+03	2.01E+06	8.08E+04	2.09E+06
70	4.77E+03	1.70E+06	5.79E+04	1.77E+06
80	4.71E+03	1.48E+06	4.35E+04	1.53E+06
90	4.73E+03	1.30E+06	3.41E+04	1.34E+06
100	4.67E+03	1.15E+06	2.66E+04	1.18E+06
120	4.61E+03	9.35E+05	1.85E+04	9.58E+05
140	4.59E+03	7.76E+05	1.37E+04	7.94E+05
160	4.59E+03	6.58E+05	1.02E+04	6.73E+05
180	4.58E+03	5.63E+05	7.90E+03	5.75E+05
200	4.61E+03	4.89E+05	6.30E+03	5.00E+05
220	4.65E+03	4.30E+05	5.07E+03	4.40E+05
240	4.67E+03	3.81E+05	4.28E+03	3.90E+05
260	4.72E+03	3.41E+05	3.56E+03	3.49E+05
280	4.73E+03	3.06E+05	3.04E+03	3.14E+05
300	4.78E+03	2.76E+05	2.64E+03	2.84E+05
320	4.80E+03	2.50E+05	2.25E+03	2.57E+05
400	4.88E+03	1.75E+05	1.44E+03	1.81E+05
500	4.93E+03	1.20E+05	8.89E+02	1.26E+05
600	4.93E+03	8.65E+04	6.41E+02	9.21E+04
700	4.91E+03	6.30E+04	4.79E+02	6.84E+04
800	4.84E+03	4.77E+04	3.61E+02	5.29E+04
900	4.78E+03	3.66E+04	2.91E+02	4.16E+04
1000	4.70E+03	2.89E+04	2.40E+02	3.38E+04
1100	4.65E+03	2.31E+04	2.04E+02	2.80E+04
1200	4.81E+03	1.86E+04	1.81E+02	2.36E+04
1300	4.85E+03	1.52E+04	1.64E+02	2.02E+04
1400	4.76E+03	1.25E+04	1.70E+02	1.74E+04
1500	4.64E+03	1.04E+04	1.58E+02	1.52E+04
1600	4.51E+03	8.67E+03	9.90E+01	1.33E+04
1700	4.38E+03	7.17E+03	6.20E+01	1.16E+04
1800	4.25E+03	6.01E+03	4.27E+01	1.03E+04
1900	4.15E+03	4.98E+03	2.82E+01	9.17E+03
2000	4.04E+03	4.14E+03	1.71E+01	8.19E+03
3000	3.11E+03	4.39E+02	1.33E+00	3.55E+03
4000	2.48E+03	1.92E+01	1.84E-01	2.50E+03
5000	1.95E+03	7.55E-01	4.04E-02	1.95E+03
6000	1.56E+03	4.30E-02	1.11E-02	1.56E+03
8000	1.04E+03	1.53E-04	1.42E-03	1.04E+03
10000	7.51E+02	2.30E-07	0.00E+00	7.51E+02

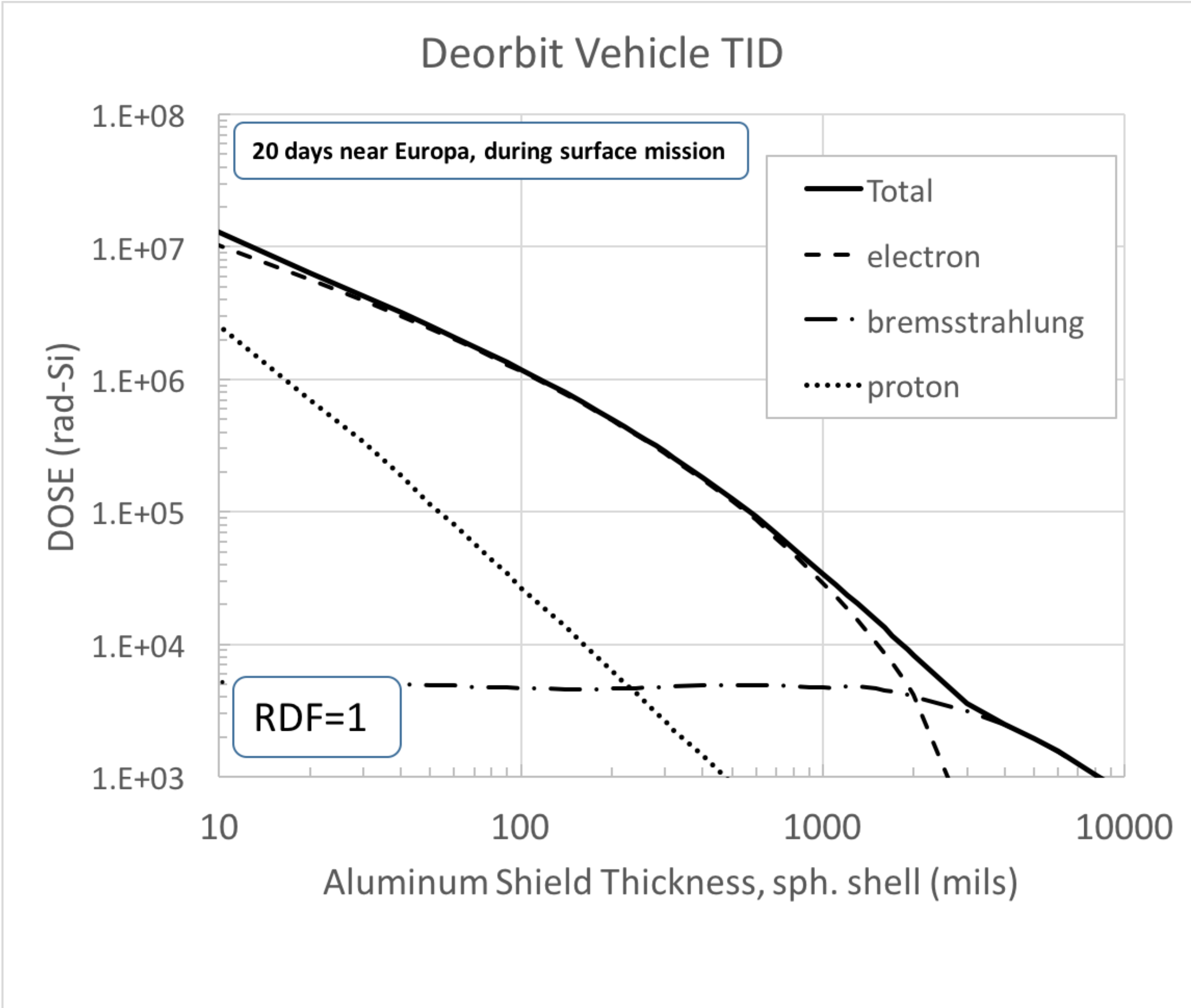


Figure 4.7.2-2 Ionizing dose-depth curves as a function of spherical shell aluminum shielding thickness for the Descent Stage and Deorbit Stage for trajectory 12L4_50km. RDF = 1. [TBR]

Table 4.7.2-3 Total ionizing dose-depth table as a function of spherical shell aluminum shielding thickness for the Lander for the Design Reference Mission. RDF = 1. [TBR]

	Total Ionizing Dose (TID), rad-Si			
mils Aluminum	bremsstrahlung	electron	proton	Total
0.10	6.41E+01	4.30E+07	4.57E+08	5.00E+08
1	1.71E+03	6.08E+07	7.11E+07	1.32E+08
10	6.02E+03	1.24E+07	3.98E+06	1.63E+07
20	6.33E+03	6.95E+06	1.08E+06	8.03E+06
30	6.29E+03	4.90E+06	5.17E+05	5.43E+06
40	6.18E+03	3.79E+06	2.81E+05	4.07E+06
50	6.07E+03	3.08E+06	1.67E+05	3.26E+06
60	6.14E+03	2.59E+06	1.20E+05	2.72E+06
70	6.06E+03	2.22E+06	8.51E+04	2.31E+06
80	6.03E+03	1.95E+06	6.35E+04	2.02E+06
90	6.11E+03	1.72E+06	4.95E+04	1.77E+06
100	6.07E+03	1.54E+06	3.85E+04	1.58E+06
120	6.06E+03	1.27E+06	2.64E+04	1.30E+06
140	6.10E+03	1.06E+06	1.94E+04	1.09E+06
160	6.15E+03	9.12E+05	1.44E+04	9.32E+05
180	6.19E+03	7.88E+05	1.11E+04	8.05E+05
200	6.27E+03	6.91E+05	8.78E+03	7.06E+05
220	6.35E+03	6.13E+05	7.02E+03	6.27E+05
240	6.42E+03	5.47E+05	5.91E+03	5.59E+05
260	6.52E+03	4.93E+05	4.88E+03	5.05E+05
280	6.55E+03	4.46E+05	4.16E+03	4.57E+05
300	6.65E+03	4.05E+05	3.60E+03	4.15E+05
320	6.70E+03	3.69E+05	3.05E+03	3.78E+05
400	6.90E+03	2.62E+05	1.93E+03	2.70E+05
500	7.02E+03	1.83E+05	1.18E+03	1.91E+05
600	7.08E+03	1.32E+05	8.41E+02	1.40E+05
700	7.08E+03	9.68E+04	6.23E+02	1.04E+05
800	7.01E+03	7.34E+04	4.66E+02	8.08E+04
900	6.93E+03	5.63E+04	3.73E+02	6.36E+04
1000	6.84E+03	4.44E+04	3.06E+02	5.15E+04
1100	6.78E+03	3.56E+04	2.59E+02	4.26E+04
1200	6.97E+03	2.87E+04	2.28E+02	3.59E+04
1300	7.01E+03	2.34E+04	2.05E+02	3.06E+04
1400	6.90E+03	1.93E+04	2.12E+02	2.64E+04
1500	6.74E+03	1.60E+04	1.96E+02	2.29E+04
1600	6.57E+03	1.33E+04	1.23E+02	2.00E+04
1700	6.40E+03	1.10E+04	7.66E+01	1.75E+04
1800	6.21E+03	9.25E+03	5.26E+01	1.55E+04
1900	6.08E+03	7.66E+03	3.47E+01	1.38E+04
2000	5.91E+03	6.36E+03	2.10E+01	1.23E+04
3000	4.61E+03	6.68E+02	1.61E+00	5.28E+03
4000	3.69E+03	2.81E+01	2.20E-01	3.72E+03
5000	2.90E+03	1.05E+00	4.81E-02	2.90E+03
6000	2.33E+03	5.72E-02	1.31E-02	2.33E+03
8000	1.56E+03	1.95E-04	1.67E-03	1.56E+03
10000	1.13E+03	2.89E-07	0.00E+00	1.13E+03

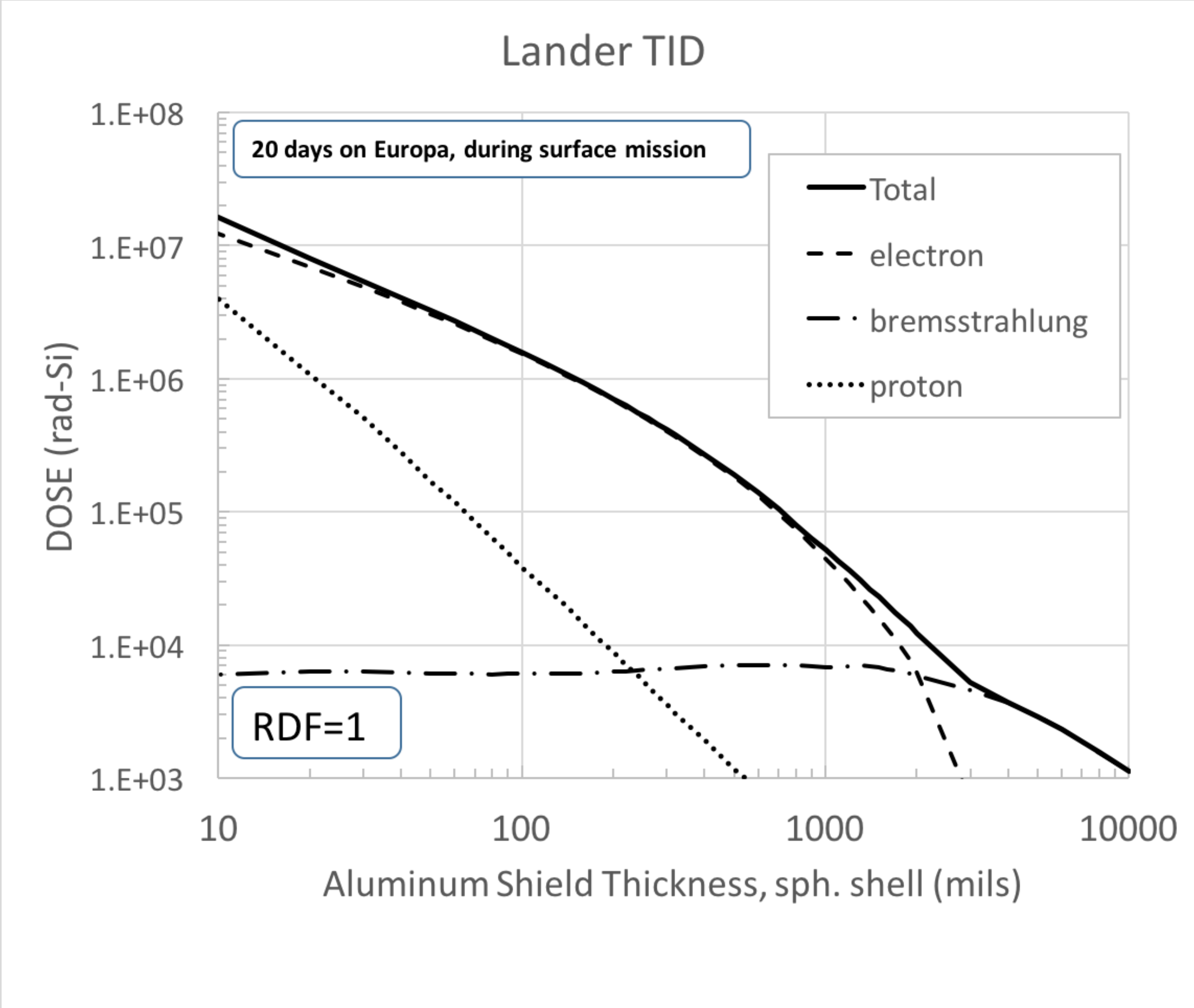


Figure 4.7.2-3 Ionizing dose-depth curves as a function of spherical shell aluminum shielding thickness for the Lander Stage for Trajectory 12L4_50km. RDF = 1. [TBR]

Table 4.7.2-4 gives maximum dose rates expected behind various aluminum shielding thicknesses. The maximum dose rate will be experienced near the orbital distance of Europa (9.5 R_J) by every flight system element.

Table 4.7.2-4 Maximum dose rate behind various thicknesses of aluminum shielding. RDF=1. [TBR]

Aluminum Shielding Thickness (mil)	Maximum Dose Rate (rad(Si)/s)
100	0.59
200	0.36
500	0.13

4.7.2.2 Displacement Damage Dose

The radiation degradation of certain electronic devices (solar cells and opto-couplers, among others), cannot be adequately characterized in terms of TID; the Displacement Damage Dose (DDD) is a more useful characterization.

Requirement: An assembly's electronic devices **shall** be selected such that the assembly operates within performance specification during and after the exposure to the radiation environment specified at a radiation design factor (RDF) of 2 times the DDD level present at the location of the device.

Table 4.7.2-5 Displacement damage dose (DDD) for the Carrier and Relay Stage as a function of spherical shell aluminum shielding thickness for trajectory 12L4_50km. RDF=1. [TBR]

	Displacement Damage Dose (DDD), MeV/g-Si		
mils Aluminum	electron	proton	Total
0.10	2.38E+10	2.12E+13	2.12E+13
1	2.34E+10	3.10E+12	3.12E+12
10	1.90E+10	1.52E+11	1.71E+11
20	1.56E+10	3.91E+10	5.47E+10
30	1.33E+10	1.87E+10	3.20E+10
40	1.16E+10	1.00E+10	2.16E+10
50	1.03E+10	5.89E+09	1.62E+10
60	9.29E+09	4.23E+09	1.35E+10
70	8.43E+09	3.02E+09	1.15E+10
80	7.73E+09	2.25E+09	9.97E+09
90	7.12E+09	1.76E+09	8.88E+09
100	6.58E+09	1.36E+09	7.94E+09
120	5.72E+09	9.48E+08	6.66E+09
140	5.03E+09	7.01E+08	5.73E+09
160	4.46E+09	5.26E+08	4.99E+09
180	3.98E+09	4.04E+08	4.38E+09
200	3.59E+09	3.21E+08	3.91E+09
220	3.26E+09	2.59E+08	3.52E+09
240	2.96E+09	2.20E+08	3.18E+09
260	2.72E+09	1.82E+08	2.90E+09
280	2.50E+09	1.56E+08	2.66E+09
300	2.30E+09	1.35E+08	2.44E+09
320	2.12E+09	1.16E+08	2.24E+09
400	1.58E+09	7.46E+07	1.65E+09
500	1.15E+09	4.67E+07	1.19E+09
600	8.52E+08	3.37E+07	8.86E+08
700	6.41E+08	2.53E+07	6.66E+08
800	4.97E+08	1.92E+07	5.16E+08
900	3.89E+08	1.55E+07	4.04E+08
1000	3.13E+08	1.28E+07	3.26E+08
1100	2.54E+08	1.09E+07	2.65E+08
1200	2.07E+08	9.54E+06	2.16E+08
1300	1.70E+08	8.49E+06	1.79E+08
1400	1.41E+08	8.26E+06	1.49E+08
1500	1.18E+08	7.20E+06	1.25E+08
1600	9.88E+07	4.53E+06	1.03E+08
1700	8.20E+07	2.88E+06	8.49E+07
1800	6.90E+07	1.98E+06	7.10E+07
1900	5.71E+07	1.32E+06	5.84E+07
2000	4.73E+07	8.03E+05	4.81E+07
3000	4.71E+06	6.39E+04	4.77E+06
4000	1.88E+05	8.92E+03	1.97E+05
5000	7.39E+03	2.01E+03	9.40E+03
6000	4.20E+02	5.46E+02	9.66E+02
8000	1.44E+00	6.68E+01	6.82E+01
10000	2.30E-03	0.00E+00	2.30E-03

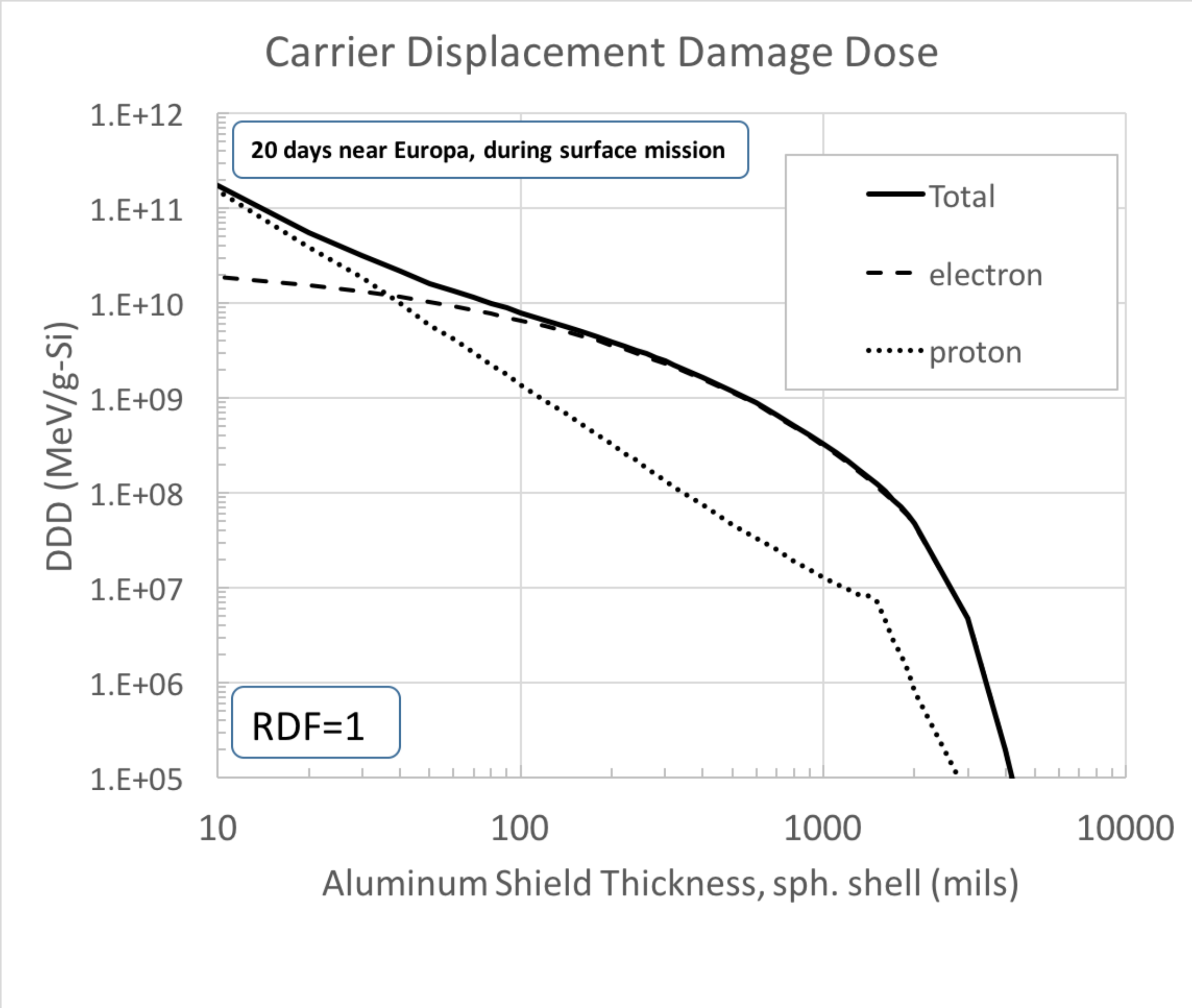


Figure 4.7.2-4 Aluminum spherical shell DDD-depth curve of silicon for the Carrier and Relay Stage for trajectory 12L4_50km. RDF = 1. [TBR]

Table 4.7.2-6 Displacement damage dose (DDD) for the Descent and Deorbit Stages as a function of spherical shell aluminum shielding thickness for trajectory 12L4_50km. RDF=1. [TBR]

	Displacement Damage Dose (DDD), MeV/g-Si		
mils Aluminum	electron	proton	Total
0.10	1.09E+10	9.85E+12	9.86E+12
1	1.08E+10	1.16E+12	1.17E+12
10	8.47E+09	5.18E+10	6.02E+10
20	6.78E+09	1.36E+10	2.04E+10
30	5.69E+09	6.56E+09	1.23E+10
40	4.88E+09	3.58E+09	8.46E+09
50	4.28E+09	2.13E+09	6.42E+09
60	3.82E+09	1.55E+09	5.37E+09
70	3.43E+09	1.12E+09	4.55E+09
80	3.12E+09	8.36E+08	3.95E+09
90	2.85E+09	6.59E+08	3.51E+09
100	2.61E+09	5.14E+08	3.13E+09
120	2.24E+09	3.61E+08	2.60E+09
140	1.95E+09	2.70E+08	2.21E+09
160	1.71E+09	2.04E+08	1.91E+09
180	1.51E+09	1.58E+08	1.67E+09
200	1.35E+09	1.27E+08	1.47E+09
220	1.21E+09	1.03E+08	1.31E+09
240	1.10E+09	8.79E+07	1.18E+09
260	9.99E+08	7.34E+07	1.07E+09
280	9.12E+08	6.32E+07	9.75E+08
300	8.35E+08	5.51E+07	8.90E+08
320	7.67E+08	4.73E+07	8.14E+08
400	5.60E+08	3.09E+07	5.91E+08
500	4.03E+08	1.98E+07	4.22E+08
600	2.97E+08	1.44E+07	3.12E+08
700	2.23E+08	1.09E+07	2.34E+08
800	1.73E+08	8.43E+06	1.81E+08
900	1.35E+08	6.87E+06	1.42E+08
1000	1.08E+08	5.72E+06	1.14E+08
1100	8.79E+07	4.90E+06	9.28E+07
1200	7.15E+07	4.33E+06	7.59E+07
1300	5.89E+07	3.89E+06	6.28E+07
1400	4.87E+07	3.81E+06	5.25E+07
1500	4.06E+07	3.33E+06	4.40E+07
1600	3.41E+07	2.10E+06	3.62E+07
1700	2.83E+07	1.34E+06	2.97E+07
1800	2.38E+07	9.25E+05	2.47E+07
1900	1.97E+07	6.19E+05	2.03E+07
2000	1.63E+07	3.78E+05	1.67E+07
3000	1.64E+06	3.06E+04	1.67E+06
4000	6.82E+04	4.34E+03	7.25E+04
5000	2.82E+03	9.88E+02	3.80E+03
6000	1.66E+02	2.71E+02	4.37E+02
8000	5.92E-01	3.35E+01	3.41E+01
10000	9.49E-04	0.00E+00	9.49E-04

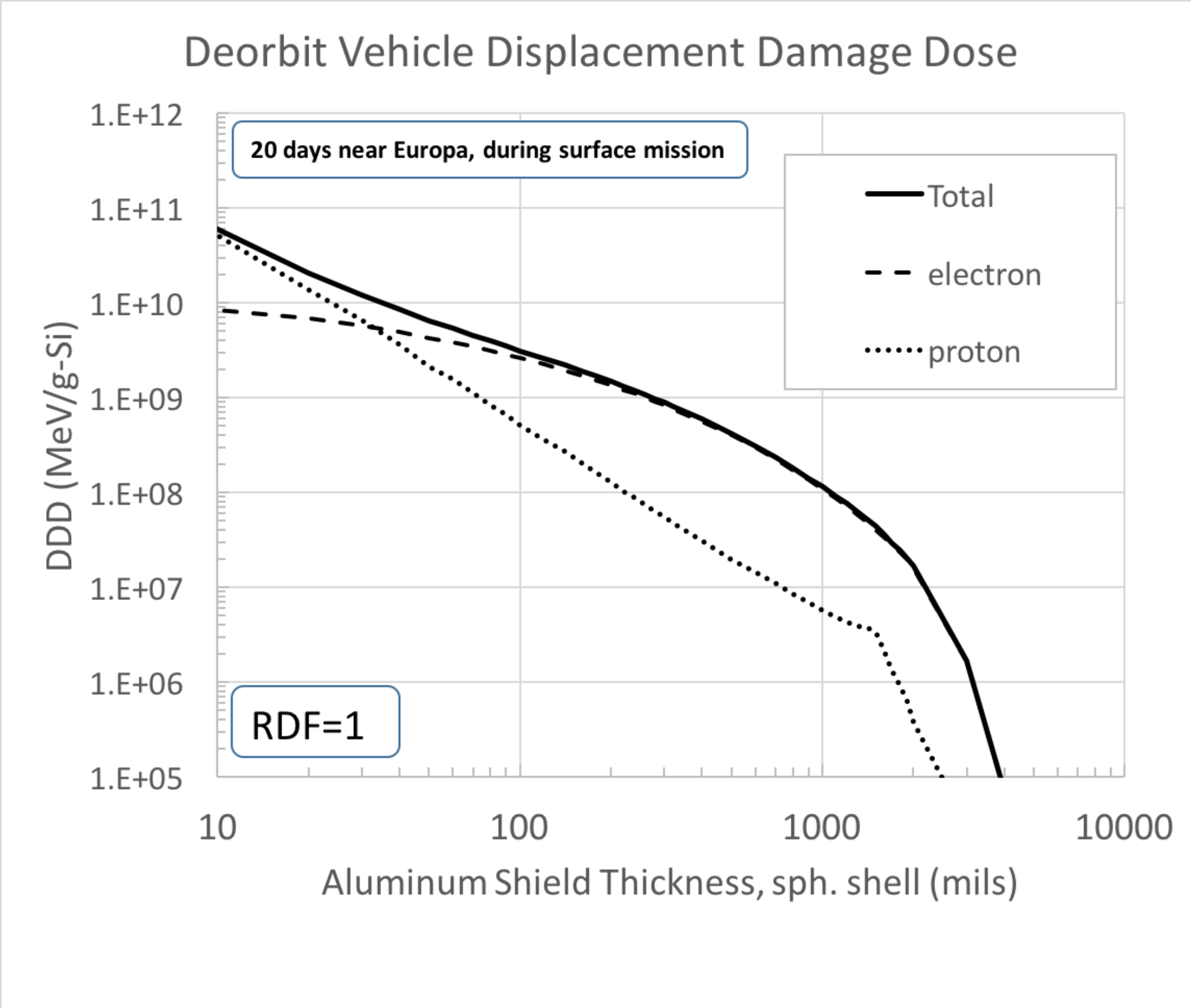


Figure 4.7.2-5 Aluminum spherical shell DDD-depth curve of silicon for the Descent and Deorbit Stages for trajectory 12L4_50km. RDF = 1. [TBR]

Table 4.7.2-7 Displacement damage dose (DDD) for the Lander as a function of spherical shell aluminum shielding thickness for trajectory 12L4_50km. RDF=1. [TBR]

	Displacement Damage Dose (DDD), MeV/g-Si			
mils Aluminum	bremsstrahlung	electron	proton	Total
0.10	0.00E+00	1.36E+10	1.22E+13	1.22E+13
1	0.00E+00	1.34E+10	1.68E+12	1.70E+12
10	0.00E+00	1.07E+10	8.14E+10	9.21E+10
20	0.00E+00	8.71E+09	2.10E+10	2.97E+10
30	0.00E+00	7.40E+09	1.00E+10	1.74E+10
40	0.00E+00	6.42E+09	5.40E+09	1.18E+10
50	0.00E+00	5.68E+09	3.18E+09	8.86E+09
60	0.00E+00	5.11E+09	2.29E+09	7.40E+09
70	0.00E+00	4.63E+09	1.64E+09	6.26E+09
80	0.00E+00	4.23E+09	1.22E+09	5.45E+09
90	0.00E+00	3.89E+09	9.55E+08	4.85E+09
100	0.00E+00	3.59E+09	7.41E+08	4.33E+09
120	0.00E+00	3.12E+09	5.16E+08	3.63E+09
140	0.00E+00	2.74E+09	3.82E+08	3.12E+09
160	0.00E+00	2.42E+09	2.88E+08	2.71E+09
180	0.00E+00	2.16E+09	2.21E+08	2.38E+09
200	0.00E+00	1.94E+09	1.76E+08	2.12E+09
220	0.00E+00	1.76E+09	1.42E+08	1.90E+09
240	0.00E+00	1.60E+09	1.21E+08	1.72E+09
260	0.00E+00	1.47E+09	1.00E+08	1.57E+09
280	0.00E+00	1.35E+09	8.61E+07	1.43E+09
300	0.00E+00	1.24E+09	7.48E+07	1.31E+09
320	0.00E+00	1.14E+09	6.40E+07	1.21E+09
400	0.00E+00	8.47E+08	4.14E+07	8.88E+08
500	0.00E+00	6.14E+08	2.61E+07	6.40E+08
600	0.00E+00	4.56E+08	1.89E+07	4.75E+08
700	0.00E+00	3.43E+08	1.42E+07	3.57E+08
800	0.00E+00	2.66E+08	1.08E+07	2.76E+08
900	0.00E+00	2.07E+08	8.78E+06	2.16E+08
1000	0.00E+00	1.67E+08	7.26E+06	1.74E+08
1100	0.00E+00	1.35E+08	6.18E+06	1.41E+08
1200	0.00E+00	1.10E+08	5.44E+06	1.16E+08
1300	0.00E+00	9.06E+07	4.86E+06	9.54E+07
1400	0.00E+00	7.50E+07	4.74E+06	7.97E+07
1500	0.00E+00	6.25E+07	4.13E+06	6.66E+07
1600	0.00E+00	5.25E+07	2.60E+06	5.51E+07
1700	0.00E+00	4.35E+07	1.66E+06	4.52E+07
1800	0.00E+00	3.66E+07	1.14E+06	3.77E+07
1900	0.00E+00	3.03E+07	7.61E+05	3.10E+07
2000	0.00E+00	2.51E+07	4.64E+05	2.55E+07
3000	0.00E+00	2.49E+06	3.71E+04	2.53E+06
4000	0.00E+00	9.93E+04	5.20E+03	1.04E+05
5000	0.00E+00	3.88E+03	1.18E+03	5.06E+03
6000	0.00E+00	2.20E+02	3.20E+02	5.40E+02
8000	0.00E+00	7.53E-01	3.93E+01	4.00E+01
10000	0.00E+00	1.20E-03	0.00E+00	1.20E-03

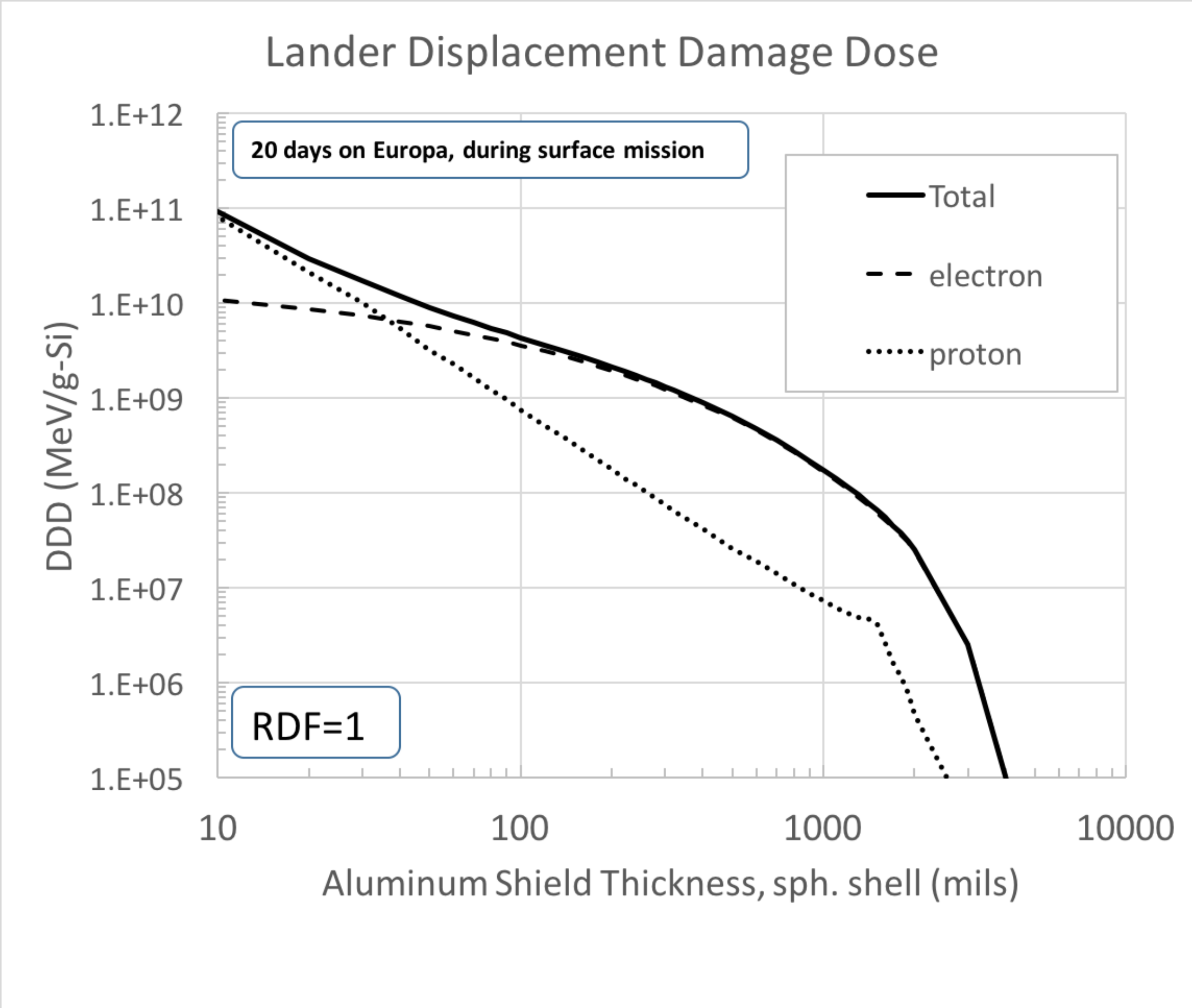


Figure 4.7.2-6 Aluminum spherical shell DDD-depth curve of silicon for the Lander for trajectory 12L4_50km. RDF = 1. [TBR]

4.7.3 Peak Flux of Jovian Electrons and Protons

Requirement: The flight system **shall** be designed to survive and operate within specifications under 5 [**TBR**] times the peak average flux environment shown in Table 4.7.3-1. Each element of the flight system will be subject to the same peak flux in free space near the orbital distance of Europa's orbit. While on the surface, the Lander Stage will experience less than half the expected flux at any given moment compared to the Carrier and Relay Stage due to the shielding by Europa, itself.

Since the radiation model provides a time-averaged value of the flux at a given position, the maximum flux seen along the trajectory is the peak average flux, rather than the peak instantaneous flux. Flux averaging is based on the statistical mean of 10-minute energetic particle measurements made at the Jovian magnetic equator.

An estimate of the statistical variation of energetic charged particle flux and potential worst-case environments can be deduced from Figure4.7.3-1. This figure provides super-imposed plots of the Galileo Energetic Particle Detector (EPD) DC3 channel (E>11 MeV) flux versus radial distance compared to the average GIRE2p model.

Table 4.7.3-1 GIRE2p Peak average flux of Jovian electrons and protons for trajectory 12L4_50km. RDF = 1. [TBR]

	Electrons		Protons	
Energy	Integral	Differential	Integral	Differential
(MeV)	($cm^{-2}\ s^{-1}$)	($cm^{-2}\ MeV^{-1}\ s^{-1}$)	($cm^{-2}\ s^{-1}$)	($cm^{-2}\ MeV^{-1}\ s^{-1}$)
0.0001	5.177E+10	1.951E+14	2.937E+08	7.714E+10
0.0003	2.484E+10	1.011E+14	2.830E+08	3.996E+10
0.0005	1.383E+10	2.928E+13	2.787E+08	1.157E+10
0.001	1.011E+10	1.199E+12	2.772E+08	4.929E+08
0.003	8.143E+09	8.509E+11	2.761E+08	5.882E+08
0.005	6.461E+09	8.328E+11	2.746E+08	9.172E+08
0.01	3.494E+09	4.401E+11	2.683E+08	1.553E+09
0.03	8.977E+08	3.596E+10	2.280E+08	2.393E+09
0.05	4.931E+08	1.122E+10	1.839E+08	2.048E+09
0.1	2.352E+08	2.311E+09	1.220E+08	7.748E+08
0.2	1.290E+08	4.760E+08	7.786E+07	2.581E+08
0.3	9.600E+07	2.360E+08	5.923E+07	1.357E+08
0.5	6.590E+07	9.690E+07	4.125E+07	6.037E+07
1	3.990E+07	2.850E+07	2.406E+07	2.011E+07
2	2.460E+07	8.520E+06	1.260E+07	6.700E+06
3	1.850E+07	4.410E+06	7.590E+06	3.710E+06
5	1.240E+07	2.210E+06	3.010E+06	1.310E+06
10	5.090E+06	9.220E+05	4.650E+05	1.440E+05
20	1.040E+06	1.350E+05	4.610E+04	7.900E+03
30	3.610E+05	3.130E+04	1.110E+04	1.290E+03
50	9.900E+04	4.880E+03	1.810E+03	1.270E+02
100	1.890E+04	4.390E+02	1.510E+02	5.340E+00
200		4.390E+01	1.250E+01	2.220E-01
300			2.920E+00	3.450E-02
500			4.650E-01	3.290E-03
1000			3.840E-02	1.360E-04

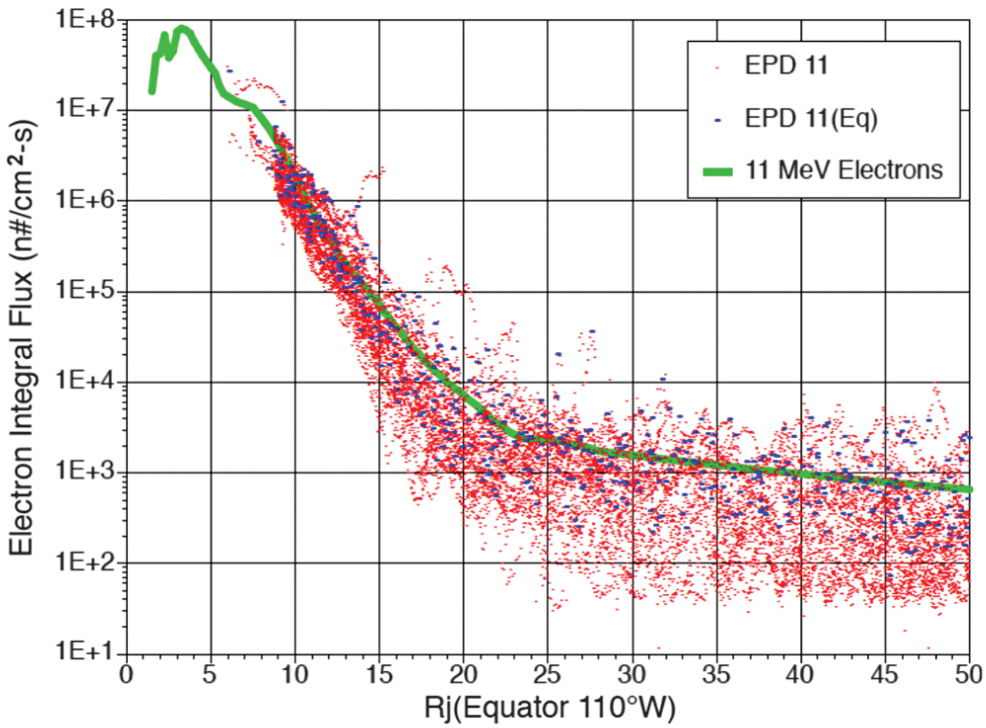


Figure 4.7.3-1 Galileo EPD data (small red dots) with magnetic equator crossings indicated (larger, blue dots), and GIRE2p (green line) integral electron flux at 11 MeV. From *Garrett, Kim, and Evans* [2016].

4.7.4 Single Event Effects (SEE)

Requirement: The flight system **shall** be designed to survive and operate within specifications under the single event effect (SEE) environment Figure 4.7.4-1 and Figure 4.7.4-2.

Electronics may be susceptible to Single Event Effects, or SEE, which include reversible, non-destructive actions (Single Event Upsets, or SEUs) such as memory bit-flips; or potentially destructive actions such as device latch-up. SEEs are caused by high-energy ions. The term “heavy ion”, as used below, refers to any ion having atomic number Z > 1; i.e. anything larger than a proton. If the part’s SEE threshold LET (linear energy transfer) is less than 15 MeV-cm^2/mg, then high-energy protons can also cause SEE. These types of high-energy particles are found in galactic cosmic rays and solar particle events.

In electronic sensors, SEE can manifest itself as spurious signals, i.e. radiation-induced background noise.

Requirement: An assembly’s electronic devices **shall** be chosen such that the assembly operates within performance specification during and after exposure to the high-energy radiation environments with respect to SEE.

The subsystem/assembly and system-level requirements regarding performance with respect to SEE during operation are as follows:

Requirement: Temporary loss of function or loss of data **shall** be permitted provided that the loss does not compromise subsystem/system health, full performance can be recovered rapidly, and there is no time in the mission that the loss is mission critical.

Requirement: Normal operation and function **shall** be restored via internal/flight system correction methods without external intervention in the event of an SEU.

Requirement: Fault traceability **shall** be provided in the telemetry stream to the greatest extent practical for all anomalies involving SEEs.

An RDF = 1 will be applied to the environments specified in the following sub-sections (4.7.4.1 Solar Proton Peak Flux, 4.7.4.2 Solar Heavy Ion Peak Flux, 4.7.4.3 Galactic Cosmic Ray Proton Flux, and 4.7.4.4 Galactic Cosmic Ray Heavy Ion Flux).

4.7.4.1 Solar Proton Peak Flux

The solar proton peak flux environment, which is to be used for proton-induced SEE in parts susceptible to proton-induced SEE, is given by the CREME96 model for the worst-case (5 minute average) solar event protons. For information purposes, the flux behind 25 mils of aluminum shielding at 1 AU is provided in **Error! Reference source not found.**. Peak fluxes for other shielding thicknesses should be determined using CREME96 models located at: https://creme.isde.vanderbilt.edu/.

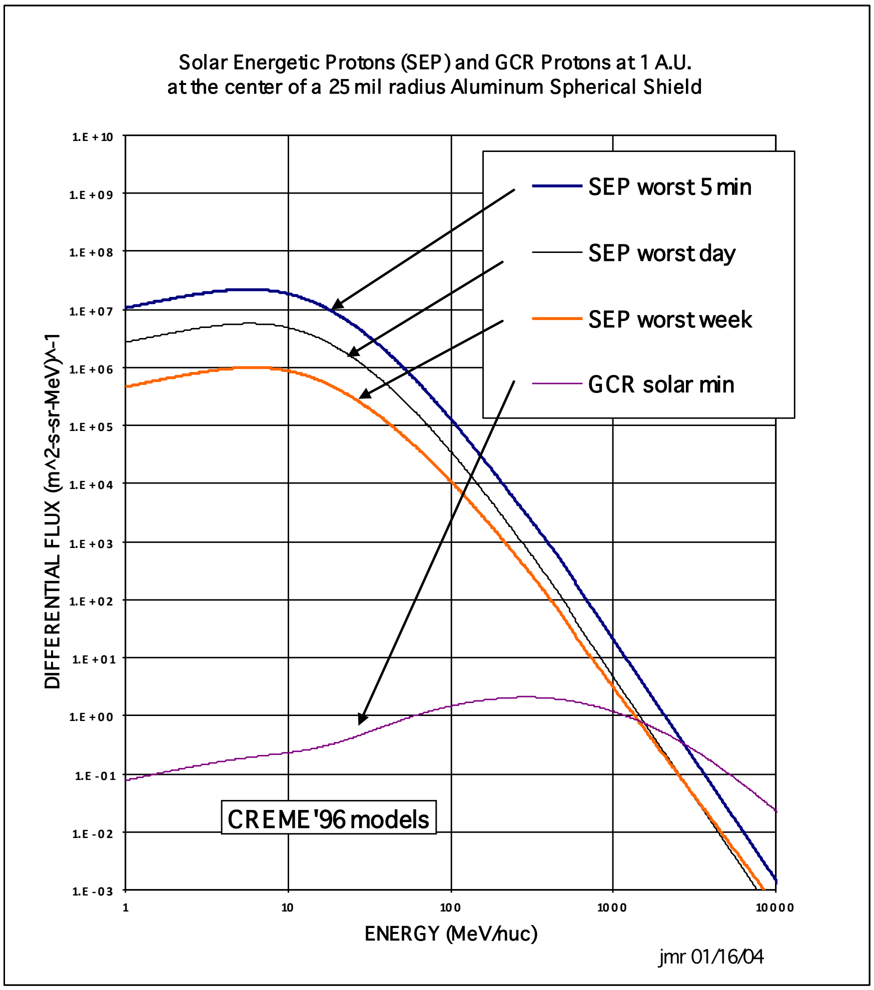


Figure 4.7.4-1 Solar energetic proton (at 1 AU) and galactic cosmic ray (GCR) proton fluxes.

4.7.4.2 Solar Heavy Ion Peak Flux

The solar particle event heavy ion peak flux environment is given by the CREME96 model for the worst-case (5 minute average) heavy ions. For information purposes, the flux behind 25 mils of aluminum shielding is provided in Figure 4.7.4-2. Peak fluxes for other shielding thicknesses should be determined using CREME96 models located at: https://creme.isde.vanderbilt.edu/.

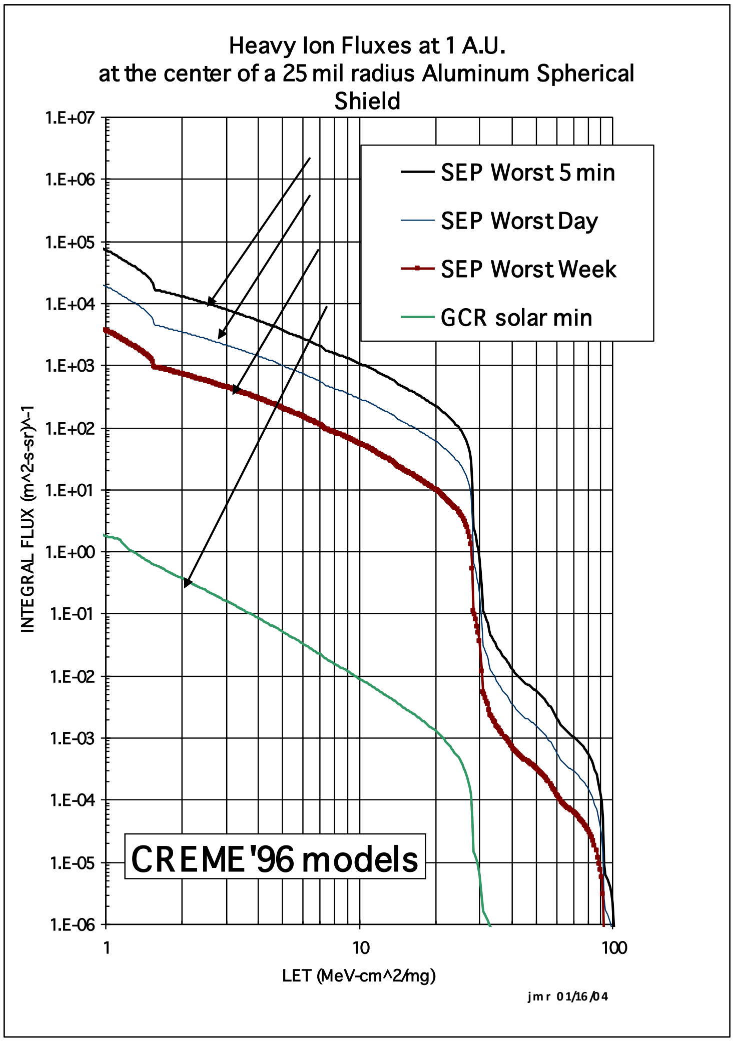


Figure 4.7.4-2 Solar and galactic cosmic ray (GCR) heavy ion fluxes.

4.7.4.3 Galactic Cosmic Ray (GCR) Proton Flux

The GCR proton environment, which is to be used for proton-induced SEE in parts susceptible to proton-induced SEE, is given by the CREME96 model for GCR protons. For information purposes, the flux behind 25 mils of aluminum shielding is provided in Figure 4.7.4-1.

4.7.4.4 Galactic Cosmic Ray (GCR) Heavy Ion Flux

The GCR heavy ion environment is given by the CREME96 model for heavy ions at solar minimum. For information purposes, the flux behind 25 mils of aluminum shielding is provided in Figure 4.7.4-2.

4.8 Atomic Oxygen Environments

Requirement: The flight system **shall** be designed to survive and operate within specifications under the atomic oxygen environment in Table 4.8-1.

Table 4.8-1 Europa Lander mission fluence of atomic oxygen (AO). [TBD; Placeholder]

[The baseline Europa Lander mission fluence of atomic oxygen (AO) will be provided at a later date.]

Requirement: Materials exposed to space with a vector in the ram direction **shall** survive the mission exposure to AO in Table 4.8-1 with acceptable property characteristics.

4.9 Europa Gravity

Europa's surface gravitational acceleration is approximately 1.31 m/s^2, or nearly 2/15 that of Earth's gravitational acceleration.

4.10 Solar Spectral Irradiance

The solar electromagnetic environment mean flux for each mission phase is as stated in Table 4.4.3-1 with a solar irradiance spectrum is shown in Table 4.10-1.

Table 4.10-1 Solar Spectral Irradiance 0.0850 – 7.0 Microns.

λ (μm)	P (%)	λ (μm)	P (%)	λ (μm)	P (%)
0.0850	3.8×10^{-4}	0.36	5.317	0.67	43.745
0.0900	3.9×10^{-4}	0.365	5.723	0.68	44.816
0.0950	4.0×10^{-4}	0.37	6.151	0.69	45.856
0.1000	4.1×10^{-4}	0.375	6.583	0.70	46.880
0.1050	4.2×10^{-4}	0.38	7.003	0.71	47.882
0.1100	4.2×10^{-4}	0.385	7.413	0.72	48.865
0.1150	4.3×10^{-4}	0.39	7.819	0.73	49.827
0.1200	4.4×10^{-4}	0.395	8.242	0.74	50.769
0.1250	4.7×10^{-4}	0.40	8.725	0.75	51.691
0.1320	4.9×10^{-4}	0.405	9.293	0.80	56.019
0.1350	5.2×10^{-4}	0.41	9.920	0.85	59.890
0.1400	5.4×10^{-4}	0.415	10.572	0.90	63.358
0.1450	5.6×10^{-4}	0.42	11.222	0.95	66.544
0.1500	5.8×10^{-4}	0.425	11.858	1.0	69.465
0.1550	6.3×10^{-4}	0.43	12.474	1.1	74.409
0.1600	6.9×10^{-4}	0.435	13.084	1.2	78.386
0.1650	8.2×10^{-4}	0.44	13.726	1.3	81.638
0.1700	1.01×10^{-3}	0.445	14.415	1.4	84.343
0.1750	1.31×10^{-3}	0.45	15.141	1.5	86.645
0.1800	1.70×10^{-3}	0.455	15.892	1.6	88.607
0.1850	2.33×10^{-3}	0.46	16.653	1.7	90.256
0.1900	3.16×10^{-3}	0.465	17.414	1.8	91.590
0.1950	5.2×10^{-3}	0.47	18.168	1.9	92.643
0.2000	8.1×10^{-3}	0.475	18.921	2.0	93.489
0.2050	1.34×10^{-2}	0.48	19.682	2.1	94.202
0.2100	2.05×10^{-2}	0.485	20.430	2.2	94.827
0.2150	3.53×10^{-2}	0.49	21.156	2.3	95.370
0.22	0.0502	0.495	21.878	2.4	95.858
0.225	0.0729	0.50	22.599	2.5	96.294
0.23	0.0972	0.505	23.313	2.6	96.671
0.235	0.1205	0.51	24.015	2.7	97.007
0.24	0.1430	0.515	24.702	2.8	97.310
0.245	0.1681	0.52	25.379	2.9	97.584
0.25	0.1944	0.525	26.060	3.0	97.828
0.255	0.2267	0.53	26.743	3.1	98.038
0.26	0.270	0.535	29.419	3.2	98.218
0.265	0.328	0.54	28.084	3.3	98.372
0.27	0.405	0.545	28.738	3.4	98.505
0.275	0.486	0.55	29.381	3.5	98.620
0.28	0.465	0.555	30.017	3.6	98.725
0.285	0.644	0.56	30.648	3.7	98.819
0.29	0.811	0.565	31.276	3.8	98.906
0.295	1.008	0.57	31.908	3.9	98.985
0.30	1.211	0.575	32.542	4.0	99.058
0.305	1.417	0.58	33.176	4.1	99.125
0.31	1.656	0.585	33.809	4.2	99.186
0.315	1.924	0.59	34.440	4.3	99.241
0.32	2.219	0.595	35.065	4.4	99.291
0.325	2.552	0.60	35.683	4.5	99.337
0.33	2.928	0.61	36.902	4.6	99.379
0.335	3.324	0.62	38.098	4.7	99.416
0.34	3.722	0.63	39.270	4.8	99.450
0.345	4.118	0.64	40.421	4.9	99.482
0.35	4.517	0.65	41.550	5.0	99.511
0.355	4.919	0.66	42.658	6.0	99.718
				7.0	99.819

λ (μm) is wavelength; and P is the percentage of the solar constant associated with wavelengths shorter than λ.

5 Appendix A: Acronyms and Abbreviations

AC Alternating Current

AFT	Allowable Flight Temperature
AM	Amplitude Modulation
AO	Announcement of Opportunity
AO	Atomic Oxygen
ATLO	Assembly, Test, and Launch Operations
CE	Conducted Emissions
c.g.	Center of Gravity
CLA	Coupled Loads Analysis
CS	Conducted Susceptibility
CogE	Cognizant Engineer
dB	Decibel
DC	Direct Current
DDD	Displacement Damage Dose
EELV	Evolved Expendable Launch Vehicles
EM	Europa Mission
EM	Engineering Model
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
ERD	Environmental Requirements Document
ERE	Environmental Requirements Engineer
ESD	Electrostatic Discharge
EACS	Environmental Analysis Completion Statement (form)
ETAS	Environmental Test Authorization Summary (form)
EVEEGA	Earth Venus Earth Earth Gravity Assist
FA	Flight Acceptance
FMH	Free Molecular Heating
g	Acceleration of Gravity
g_{rms}	Acceleration Root Mean Square
GCR	Galactic Cosmic Ray
GHz	Giga-Hertz
GN2	Gaseous Nitrogen
Hi-Rtn	High-Return
Hrs	Hour(s)
HV	High Voltage
Hz	Hertz
IESD	Internal Electrostatic Discharge
JPL	Jet Propulsion Laboratory
K	Kelvin
kHz	kilo-Hertz
LISN	Line Impedance Simulation Network
LV	Launch Vehicle
LS	Launch Site
LVDS	Low Voltage Differential Signaling
MAC	Mass Acceleration Curve
MAM	Mission Assurance Manager
MeV	Million electron Volts
MEFL	Maximum Expected Flight Level
min	Minute
MinEFL	Minimum Expected Flight Level
MHz	Mega-Hertz
MIL-STD	Military Standard
mΩ	Milli-Ohm

MΩ	Meg-Ohm
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
Non-Op	Non-Operating
nT	nano-Testla
Op	Operating
PAF	Payload Adapter Fairing
PF	Protoflight
PLF	Payload Fairing
PSI	Pounds per Square Inch
QA	Quality Assurance
Qual	Qualification (Model)
RACS	Radiation Analysis Completion Statement (form)
RDF	Radiation Design Factor
RE	Radiated Emissions
RF	Radio Frequency
Rj	Jupiter Radius
RMS	Root Mean Square
RS	Radiated Susceptibility
s	Second
SEE	Single Event Effects
SEP	Solar Energetic Particle (or Proton)
SEU	Single Event Upset
SLS	Space Launch System
SPL	Sound Pressure Level
SRS	Shock Response Spectrum
TAM	Test and Analysis Matrix
TBC	To Be Completed or Confirmed
TBD	To Be Determined
TBR	To Be Reviewed/Revised
TID	Total Ionizing Dose
TRT	Temperature Requirements Table
Vac	Vacuum
W/m2	Watt per Square Meter
WC	Worst Case

6 Appendix B: ETAS Form (Environmental Test Authorization Summary)

ETAS LOG#* ______

ENVIRONMENTAL TEST AUTHORIZATION AND SUMMARY (ETAS)

AUTHORIZATION SECTION

PROJECT:*	SUBSYSTEM/ASSEMBLY (TEST ARTICLE):*	SERIAL #:*
CONFIGURATION NAME:		
SUPPLIER:*	PART # & REV:* (NOTE: if applicable, list lower tier h/w on continuation sheet)	YEAR MFG:
H/W TYPE:* ☐EM ☐QUAL ☐FLIGHT ☐FLT SPARE ☐ OTHER ______	WIRING HARNESS (IF APPLICABLE): ☐ FLIGHT ☐ EM ☐ GSE ☐ OTHER	

ENVIRONMENTAL TEST(S) PLANNED:* CHECK ALL APPLICABLE, INDICATE TEST LEVEL: Q= Qual. , PF= Protoflight , FA= Flight Acceptance

RANDOM VIB. ☐Q ☐PF ☐FA	PYROSHOCK ☐Q ☐PF ☐FA	ACOUSTIC ☐Q ☐PF ☐FA	SINE VIB. ☐Q ☐PF ☐FA
QUASI-STATIC ☐Q ☐PF ☐FA	TEMP. ATM. ☐Q ☐PF ☐FA	THERMAL VAC. ☐Q ☐PF ☐FA	

☐ EMC: ☐Cond.Susc. ☐Cond. Emis. ☐Rad. Emis. ☐Rad. Susc. ☐Isolation ☐ESD ☐Magnetics ☐ Other ______

☐ OTHER ENV TEST: ______

ARE ANY OF THE TESTS BEING AUTHORIZED RETEST(S)?* ☐ YES ☐ NO If YES, explain: ______

FOR THE FOLLOWING QUESTIONS, PLEASE PROVIDE EXPLANATION FOR ANY "NO" ANSWERS (USE SPACE PROVIDED ON PAGE 2 AS NECESSARY)

1. DO ALL TESTS/LEVELS/DURATIONS COMPLY WITH PROJECT ENVIRONMENTAL REQUIREMENTS?* PROJ. DOC. NO. AND REV (ERD OR OTHER): ______ Comments:	☐ YES ☐ NO
2. IS THE TEST ARTICLE IDENTICAL TO THE FLIGHT CONFIGURATION?* Comments: .	☐ YES ☐ NO
3. HAS THE TEST ARTICLE PASSED ALL PRE-ENVIRONMENTAL FUNCTIONAL TESTS?* Comments: .	☐ YES ☐ NO ☐ NA
4. HAVE ALL DESIGN ANALYSES BEEN COMPLETED AND REQUIRED CHANGES INCORPORATED?* Comments:	☐ YES ☐ NO
5. ARE ALL PFRs AGAINST THIS HARDWARE CLOSED?* List Open PFRs:	☐YES or None Generated ☐ NO
6. HAVE ALL WAIVERS AND ECRs BEEN APPROVED AND REQUIRED CHANGES INCORPORATED?* List Open Items:	☐YES or None Generated ☐ NO
7. ARE ALL INSPECTION REPORTS CLOSED AND REQUIRED CHANGES INCORPORATED?* List Open Items:	☐YES or None Generated ☐ NO
8. HAS THE TEST ARTICLE PASSED ITS PRE-ENVIRONMENTAL INSPECTION?* AIDS and/or IR# ______ Comments:	☐ YES ☐ NO
9. HAS THE FUNCTIONAL TEST PROCEDURE BEEN APPROVED?* Comments:	☐YES ☐ NO
10. IS THE REQUIRED GSE (INCLUDING TEST AND HANDLING FIXTURES) AVAILABLE AND FUNCTIONING PROPERLY? Comments:	☐YES ☐ NO
11. HAVE THE OPERATIONAL SAFETY SURVEYS FOR EACH TEST BEEN SCHEDULED? Comments:	☐YES ☐ NO
12. IS A PLANETARY PROTECTION DRY HEAT MICROBIAL REDUCTION REQUIRED FOR THIS TEST ARTICLE?* If YES, indicate date completed (or planned):	☐YES ☐ NO
13. IS A CONTAMINATION CONTROL BAKEOUT REQUIRED FOR THIS TEST ARTICLE?* If YES, indicate date completed (or planned):	☐YES ☐ NO
THE FOLLOWING QUESTIONS APPLY FOR SYSTEM TESTS ONLY	
14. HAVE ALL HRCR ACTION ITEMS BEEN CLOSED AND APPROVED? If NO, please provide list of exceptions and explanations:	☐YES ☐ NO
15. HAVE ALL WAIVERS BEEN REVIEWED FOR SYSTEM TEST IMPACT? Explain any system test impacts (Use additional sheets as necessary):	☐YES ☐ NO

Please continue to page 2.

ETAS LOG#*______

ENVIRONMENTAL TEST AUTHORIZATION AND SUMMARY (ETAS)

AUTHORIZATION SECTION (Cont'd)

PROJECT:*	SUBSYSTEM/ASSEMBLY (TEST ARTICLE):*	SERIAL #:*

AUTHORIZED PROVISIONS AND EXPLANATIONS

AS A MINIMUM INCLUDE: 1) TEST PLANS/TEST PROCEDURES/OTHER TEST DOCUMENTATION, 2) TEST AGENCIES AND LOCATIONS, 3) TEST LEVELS AND DURATIONS. *

TESTS AUTHORIZED BY

COGNIZANT ENGINEER* (CTM for non-JPL h/w)	PDM/TECHNICAL MGR./INSTR MGR*	ENVIRONMENTAL REQTS. ENG*
Print Name	Print Name	Print Name
Signature Date	Signature Date	Signature Date

ETAS LOG#*______

ENVIRONMENTAL TEST AUTHORIZATION AND SUMMARY (ETAS)

TEST RESULTS SECTION

PROJECT:*	SUBSYSTEM/ASSEMBLY (TEST ARTICLE):*	SERIAL #:*

ENTER TEST SUMMARY DATA FOR EACH TEST AUTHORIZED ON PAGE 1.

TEST ENVIRONMENT* LEVELS & DURATION	TEST START/END DATES*	TEST AGENCY & LOCATION*	TEST REPORTS, PFRS, WAIVERS, TEST WITNESS & OTHER COMMENTS*	PASS/ FAIL*

Please continue to page 4.

ETAS LOG#*______

ENVIRONMENTAL TEST AUTHORIZATION AND SUMMARY (ETAS)

TEST RESULTS SECTION (Cont'd)

PROJECT:*	SUBSYSTEM/ASSEMBLY (TEST ARTICLE):*	SERIAL #:*

FOR THE FOLLOWING QUESTIONS, PLEASE USE SPACE PROVIDED BELOW, AS NECESSARY, FOR FURTHER EXPLANATION

1. WERE ALL PLANNED TESTS/LEVELS/DURATIONS ACHIEVED?* (IF NO, ATTACH EXCEPTIONS LIST)	☐ YES ☐ NO
2. WERE THERE ANY ANOMALIES OBSERVED DURING OR FOLLOWING ENVIRONMENTAL TESTS?* (IF YES PROVIDE EXPLANATIONS AND PFR #S)	☐ YES ☐ NO
3. HAS THE TEST ARTICLE PASSED ITS POST-ENVIRONMENTAL DAMAGE INSPECTIONS?* INSPECTION AIDS OR IR #: _____	☐ YES ☐ NO ☐ N/A
4. HAS THE TEST ARTICLE PASSED ITS POST-ENVIRONMENTAL FUNCTIONAL TESTS?* REPORT #: _____	☐ YES ☐ NO ☐ N/A
5. WERE ANY WAIVERS GENERATED AS A RESULT OF THE TEST(S)?* WAIVER #S*: _____	☐ YES ☐ NO

TEST SUMMARY AND EXPLANATIONS (attach test data as necessary)

TEST RESULTS DISPOSITION: *

☐Pass ☐ Pass with Waiver ☐Fail

COGNIZANT ENGINEER* (CTM for non-JPL h/w)	PDM/TECHNICAL MGR./INSTR MGR*	ENVIRONMENTAL REQTS. ENG*
Print Name	Print Name	Print Name
Signature Date	Signature Date	Signature Date

ETAS LOG#*______

ENVIRONMENTAL TEST AUTHORIZATION AND SUMMARY (ETAS)		
TEST RESULTS CONTINUATION SHEET (use as necessary)		
PROJECT:*	SUBSYSTEM/ASSEMBLY (TEST ARTICLE):*	SERIAL #:*

7 Appendix C: ECAS Form (Environmental Analysis Completion Statement)

PROJECT	ASSEMBL/SUBASSEMBLY TITLE
ENVIRONMENT SPECIFICATION: (ERD paragraph # or attach as necessary).	
ANALYSIS APPROACH :	
ANALYSIS CONCLUSIONS:	
Hardware compliance with design environment? _______yes _________no (explain)	
LIST OF PERTINENT DOCUMENTS (IOM'S, REPORTS): ATTACH AS APPROPRIATE	
______________________________ ______________________________ ______________________________ ______________________________	
PREPARED BY: ______________________________ ______________________________ __________ **Cognizant Engineer date phone #**	
APPROVED BY: ______________________________ ______________________________ **Technical Manager / Instrument Mgr. or P.I. Environmental Requirements Engineer (ERE)**	
After Technical Manager approval, send statement and supporting documentation to the Project ERE.	