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Common Instrument Interface Project

Hosted Payload Guidelines Document

Earth System Science Pathfinder Program Office
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Change Log

Version	Date	Section Affected	Description
Baseline	11/11/2011	4.0	Removed Class D from set of allowable Mission Risk Classifications
Baseline	11/11/2011	5.1.2	Updated guidelines related to OFF mode
Baseline	11/11/2011	5.1.3	Updated guidelines related to SURVIVAL mode
Baseline	11/11/2011	5.1.6	Updated guidelines related to SAFE mode
Baseline	11/11/2011	5.2.2	Changed paragraph title from “Allowed Mode Transitions” to “Preferred Mode Transitions”
Baseline	11/11/2011	12.8	Updated guidelines related to Contamination Analysis
Baseline	11/11/2011	12.10	Updated nomenclature and diction related to Particulate and Molecular Cleanliness

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1.0 OVERVIEW

1.1 Introduction

NASA's Earth Science Division (ESD) initiated the Common Instrument Interface (CII) Working Group and assigned implementation responsibility to the Earth System Science Pathfinder (ESSP) Program Office. ESD will develop Earth Science Instruments to be flown as hosted payloads on Hosted Payload Opportunities (HPO). In an effort to ensure broad representation from the NASA Centers involved in the development of Earth Science instrument, the ESSP Program Office formed a working group with representation from the Jet Propulsion Laboratory (JPL), the Goddard Space Flight Center (GSFC), the Ames Research Center (ARC), the Marshall Space Flight Center (MSFC), and the Langley Research Center (LaRC). Additionally, the CII Working Group held workshops in Fiscal Year 2011 to receive input from other NASA organizations, Department of Defense, National Oceanic and Atmospheric Administration, John Hopkins Applied Physics Lab, and numerous commercial spacecraft and instrument developers.

The development of the instruments as well as HPO will be conducted independently of each other with the goal of matching a specific instrument with a specific HPO by the instrument Key Decision Point (KDP) C timeframe. In an effort to facilitate this matching process, the CII Working Group is developing a set of common instrument-to-spacecraft interfaces. These common interfaces are captured as guidelines in this document, and the instrument developer is encouraged to use these guidelines to the extent practical for their specific instrument.

Initially, this effort has focused on Low Earth Orbit (LEO) missions based on historical data and an understanding of anticipated future trends. Recently, the CII Working Group has started investigating Geostationary Earth Orbit (GEO) missions, whose common interfaces are in the early stages of development. Consequently, the initial release of this document includes guidelines for LEO only. Subsequent releases will include GEO instrument-to-spacecraft interfaces, updates to LEO, and will also reflect payload interface data from actual HPO's.

1.2 Needs, Charter, Goals and Objectives

ESD wants to maximize its ability to collect science data, but does not have the resources to develop and operate single-mission projects in all cases. The increasing cost of launch vehicles and spacecraft forces ESD to drastically reduce mission scope, limiting its ability to answer their mandate to achieve systematic Earth Science objectives. ESD looks to broaden its approach and ability to acquire measurements from orbit by concentrating on instrument development. ESD intends to fly its instruments on non-NASA provided spacecraft in addition to future NASA platforms.

The charter for the CII Working Group is to work with industry, academia, and other government agencies to develop a set of common instrument-to-spacecraft interfaces that could serve as guidelines for instrument developers. If implemented, these guidelines would minimize interface complexities and improve the probability of matching a given instrument with an HPO.

In order to meet this need, the CII Working Group sets the following goals:

1) Develop a set of technical instrument interface guidelines that allow instrument and spacecraft developers to independently proceed in early life-cycle stages and successfully integrate following KDP C.
2) Identify hardware and software mitigations to address incompatibilities that may arise and implement these mitigations consistent with ESD approval
3) Maximize the science data returned given the constraints of the host spacecraft's dynamics

CII Working Group objectives that validate project success are:

1) Earth Science standalone instruments are matched with a host spacecraft no later than KDP C
2) All interface disconnects identified at KDP C are mitigated no later than CDR
3) The hosted payload instrument meets a high percentage of its own science objectives without impacting the host spacecraft's prime mission

1.3 Nomenclature

The term "should" denotes an optional, but desired, guideline. The term "may" denotes an optional specification. The words "is", "are", and "will" indicate statements of fact.

<u>Hosted Payload</u>: a payload manifested on a spacecraft bus flying on a primary space mission.

<u>Hosted Payload Opportunity</u>: a spacecraft bus flying on a primary space mission with surplus resources to accommodate a hosted payload.

<u>Secondary Payload</u>: a small spacecraft flying on a primary space mission, paying only the additive costs of integration, and willing to be deployed into the prime payload's insertion orbit after its separation.

1.4 Document Organization

This document is organized into two parts. Sections 1 through 14 describe the top-level guidelines. Section 15 lists acronyms and scientific unit abbreviations, and Section 16 includes reference material and best practices.

2.0 REFERENCE DOCUMENTS

2.1 NASA Documents

GSFC 420-05-01, Rev. A, *Earth Observing System (EOS) Performance Assurance Requirements for EOS General Instruments*, August 1991.

GSFC 420-05-04, *Earth Observing System (EOS) Performance Assurance Requirements for EOS Common Spacecraft*, 3 January 1994.

GSFC 422-11-122-01, Rev. B, *General Interface Requirements Document (GIRD) for EOS Common Spacecraft/Instruments*, August 1998.

GSFC- PPL-21, *Goddard Space Flight Center Preferred Parts List*, May 1996.

GSFC-S-311-P-4/09 Rev. C, *Connectors, Electrical, Polarized Shell, Rack and Panel, for Space Use*, 7 June 1991.

GSFC-STD-1000, *Rules for the Design, Development, Verification, and Operation of Flight Systems (GOLD Rules)*, 3 August 2009.

GSFC-STD-7000, *General Environmental Verification Standard (GEVS) For GSFC Flight Programs and Projects*, April 2005.

GSFC, *EOS: Common Spacecraft Radiation Environment*, April 1993.

NASA-HDBK-4001, *Electrical Grounding Architecture for Unmanned Spacecraft*, 17 February 1998

NASA-STD-4003, *Electrical Bonding for Launch Vehicles, Spacecraft, Payloads and Flight Equipment*, 8 September 2003

NPR 7150.2A, *NASA Software Engineering Requirements,* 19 November 2009.

NPR 8705.4, *Risk Classification for NASA Payloads,* 9 July 2008.

2.2 Government Documents

AFSPCMAN 91-710, *US Air Force Space Command Range Safety User Requirements Manual,* 1 February 2007.

Evolved Expendable Launch Vehicle Secondary Payload Adapter Rideshare Users Guide (ESPA RUG), Department of Defense Space Test Program, May 2010.

FED-STD-H28A, *Federal Standard: Screw-Thread Standards for Federal Services*, 10 September 2001.

MIL-DTL-24308, *General Specification for Connectors, Electric, Rectangular, Nonenvironmental, Miniature, Polarized Shell, Rack and Panel, w/ Amendment 1, Revision G*, 25 January 2011.

MIL-DTL-83513, *General Specification for Connectors, Electrical, Rectangular, Microminiature, Polarized Shell, Revision G,* 29 October 2008.

MIL-HDBK-454, *General Requirements for Electrical and Electronic Equipment, Revision B,* 15 April 2007.

MIL-HDBK-1547A, *Electronic Parts, Materials, and Processes for Space and Launch Vehicles,* 6 July 1998.

MIL-STD-461F, *Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment*, 10 December 2007.

MIL-STD-462, *Measurement of Electromagnetic Interference Characteristics*, 13 July 1967.

MIL-STD-464A, *Bonding, Electrical, and Lightning Protection for Aerospace Systems*, 18 March 1997.

MIL-STD-882, *Standard Practice for System Safety*, 10 February 2000.

MIL-STD-1246C, *Product Cleanliness Levels Contamination Control Program*, 15 February 2002.

MIL-STD-1512, *Electroexplosive Subsystems, Electrically Initiated, Design Requirements and Test Methods*, 30 September 1997.

MIL-STD-1553B, *Aircraft Internal Time Division Command/Response Multiplex Data Bus*, 30 April 1975.

Space Test Program – Standard Interface Vehicle (STP-SIV) Payload User's Guide, 15 June 2008.

2.3 **Other Documents**

417-R-RPT-0050, *GOES-R Reliable Data Delivery Protocol*, January 2008.

AIA/NAS ANA0156: *Metric Design Parameters for Male Threaded Fasteners*, 1 January 1982.

ANSI/IEST/ISO 14644-1, *Cleanrooms and Associated Controlled Environments*, 1999.

CCSDS 133.0-B-1, *Space Packet Protocol, Consultative Committee for Space Data Systems (CCSDS) Blue Book*, September 2003.

CCSDS 301.0-B-3, *Time Code Formats, Consultative Committee for Space Data Systems (CCSDS) Blue Book*, January 2002.

ECSS-E-ST-50-12C, *SpaceWire Links, Nodes, Routers and Networks, European Cooperation for Space Standardization (ECSS)*, July 2008.

ECSS-E-ST-50-51C, *SpaceWire Protocol Identification, European Cooperation for Space Standardization (ECSS)*, February 2010.

ECSS-E-ST-50-53C, *SpaceWire – CCSDS Packet Transfer Protocol, European Cooperation for Space Standardization (ECSS)*, February 2010.

EIA-RS-422-A, *Electronic Industries Association Standard Electrical Characteristics of Balanced Voltage Interface Circuits*, January 1978.

SpaceWire – CCSDS Unsegmented Code Transfer Protocol, proposed by Aeroflex at the 15th SpaceWire Working Group Meeting, October 2010.[1]

3.0 CONCEPT OF OPERATIONS

3.1 **Low Earth Orbit (LEO)**

Reserved for Future Use

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[1] All CCSDS Standards are freely available for download at the CCSDS website: http://public.ccsds.org/publications. All ECSS SpaceWire standards and the GOES-R Reliable Data Delivery Protocol are available for download at the ESA SpaceWire website: http://spacewire.esa.int/content/Standard.

4.0 LEVEL 1 DESIGN GUIDELINES

The Common Instrument Interface has seven Level 1 guidelines as shown in Table 4-1. These Level 1 guidelines are the highest guidelines in the hierarchy, and the rest of the lower-level guidelines depend on these.

Table 4-1: Top (Level 1) CII Guidelines

Function	Guideline
Priority	The instrument should be classified as a hosted payload.
Mass	The mass of the instrument should be less than or equal to 100 kg.
Power	The orbital average power required by the instrument should be less than or equal to 100 W.
Data Rate	The instrument data rate should be less than or equal to 10 Mbps.
Electrical Ground	The instrument should be electrically grounded to a single point on the spacecraft.
Thermal	The instrument should be thermally isolated from the spacecraft, such that the conductive heat transfer is less than 4W, and the radiative heat transfer is less than 3W.
Risk Classification	The instrument should be designed based upon a mission risk classification of Class C in accordance with NPR 8705.4

4.1 Supporting Analysis for Level 1 Design Guidelines

In order to provide Level 1 guidelines for future hosted payload instruments, we have examined the NASA Instrument Cost Model (NICM) remote sensing database to identify instrument characteristic parameters. The database has information on 102 different instruments that launched before 2009 from all four divisions of the Science Mission Directorate (SMD), as depicted in the Table 4-2. There are two significant characteristics of the data set that limit its statistical power to draw conclusions about Earth Science instruments. The first is the small sample size of Earth Science instruments (n=28). The second is that since more than half of the NICM instruments are Planetary, which tend to be smaller overall, the data are skewed. Nonetheless, analyzing the entire 102-instrument set provides some useful insight.

Table 4-2: Distribution of NICM Instruments Among Science Mission Directorate Divisions

SMD Division	Directed	Competed	Non-NASA	Total
Earth	18	5	5	28
Planetary	35	18	1	54
Heliophysics	5	3	1	9
Astrophysics	10	1	0	11
Total	**68**	**27**	**7**	**102**

In analyzing the data, one may easily conclude that the development cost of an instrument is a function of multiple parameters such as: mass, power, data rate, year built, SMD division and acquisition strategy. With further analysis, one will realize that these parameters are not independent of each other and are implicitly functions of mass. For example, Planetary Science instruments tend to be smaller than Earth Science instruments, and competed instruments tend to be smaller than their directed counterparts. As technology improves with time, the instruments get smaller and more capable. With this information, we have plotted the instrument cost as a function of mass as shown in Figure 4-1.

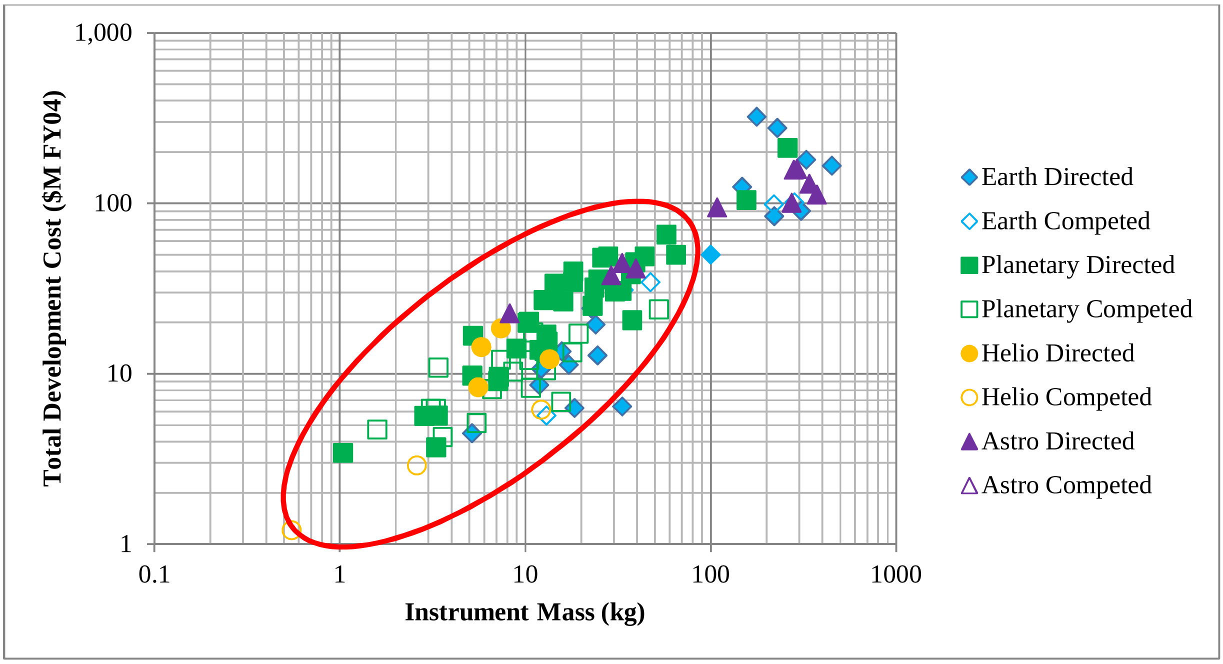


Figure 4-1: Instrument Mass vs. Development Cost

In further examination of the data, specifically the Earth Science instruments that are outside the ellipse in Figure 4-1, one realizes that they were primary instruments that the mission were built around, for example, the Aura mission with the MLS and TES instruments. Given that this document deals with instruments that are ***classified as hosted payloads*** without knowledge of what mission or spacecraft they will be paired with, the CII WG ***allocates 100 kg for the Level 1 mass guideline.*** Therefore, every effort should be made to keep the mass to less than 100 kg to increase the probability of matching up with an HPO.

Figure 4-2 shows the relationship between power and mass. The power consumed by an instrument is also approximately linearly correlated to the mass of the instrument. On this basis, ***we allocate 100 W for the Level 1 power guideline*** for a 100 kg instrument.

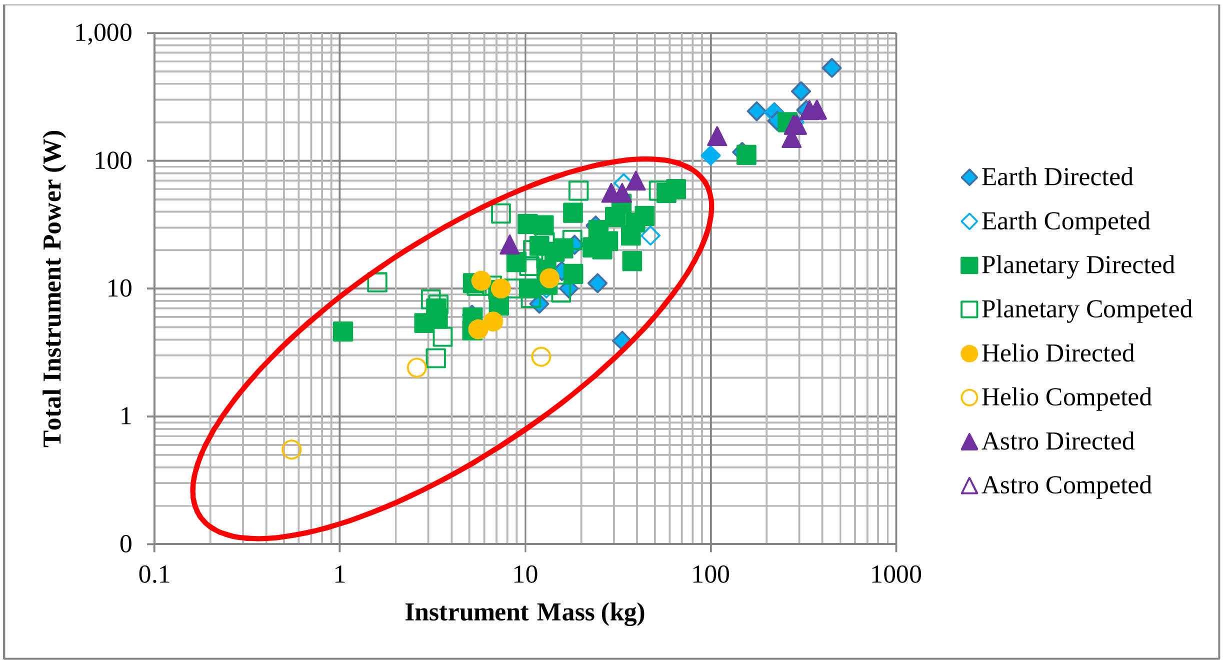


Figure 4-2: Power as a Function of Mass

As stated earlier, instruments over time have become smaller and more capable. Specifically, in Earth Science instruments this translates into generating more and more data. Figure 4-3 shows the data rates for all SMD instruments. This graph indicates that the data rate has increased by about an order of magnitude over two decades. Based upon this observation ***we set the Level 1 data rate guideline at 10 Mbps.*** It is clear that some instruments will generate more than 10 Mbps. This implies that the instruments should have the capability of on-board data analysis and or data compression or the capability of fractional time data collection. As with all guidelines contained within this document, once the instrument is paired with an HPO, the agreement between the two will supersede these guidelines.

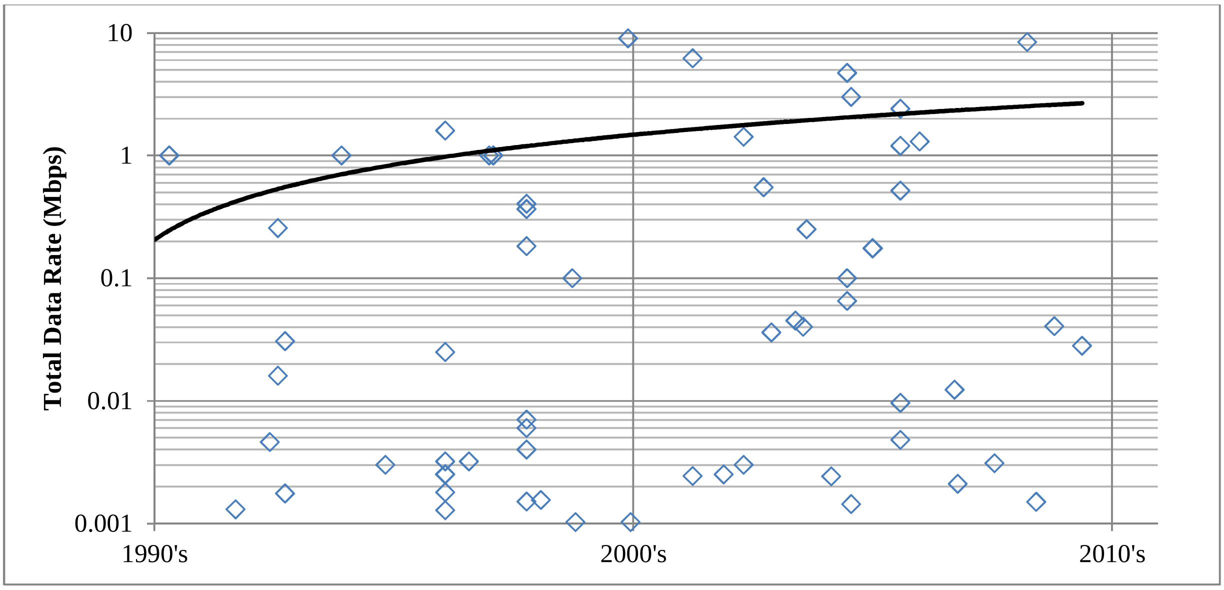


Figure 4-3: Trend of Mean Instrument Data Rates

Categorization of the instruments as hosted payloads implies that these instruments have a mission risk level of C as defined in NPR 8705.4. This in turn defines the 2 year operational life and software classification.

4.2 NICM Cumulative Distribution Analysis

Following up on the techniques in section 4.1, analyzing the distributions of various NICM database parameters using histograms and cumulative distribution functions allows instrument developers to understand how their own instruments relate to historical instruments.

Figure 4-4 plots the number of NICM database instrument that fall within mass bins from 0 to 450 kg. Specifically, bin 10 counts the instruments with masses from 0 to less than 10 kg [0, 10); bin 20 counts the instruments with masses 10 kg or greater up to 20 kg [10, 20), *etc*.

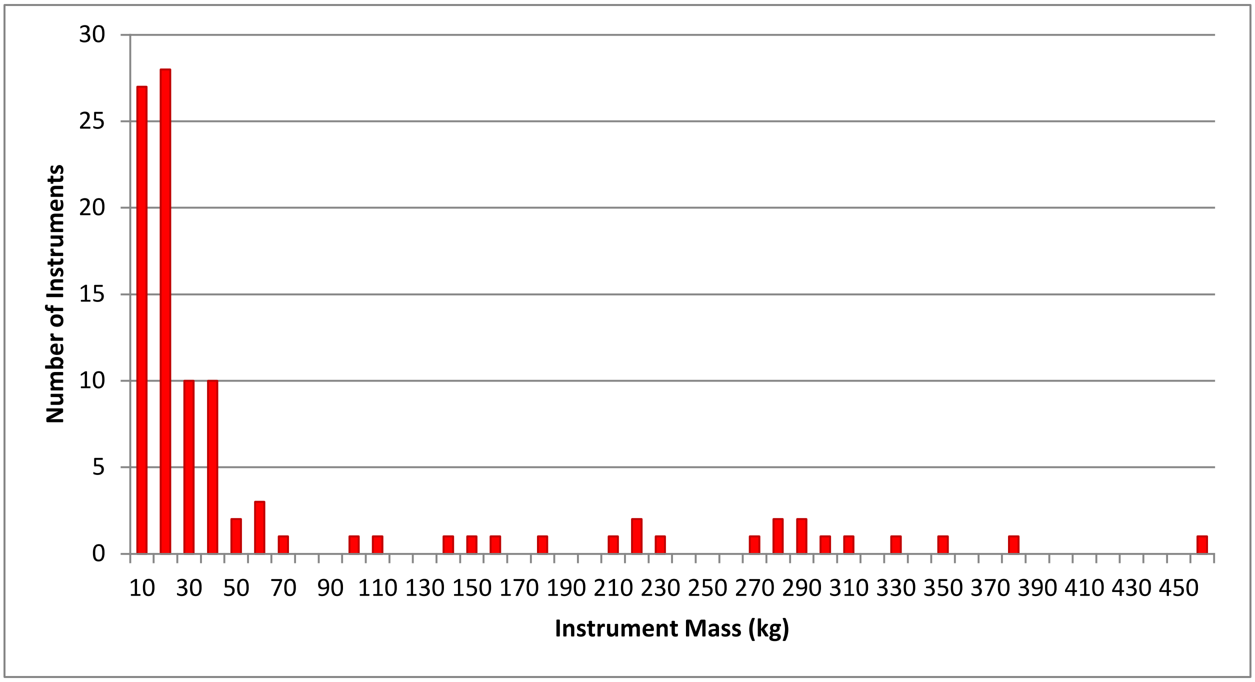


Figure 4-4: NICM Sample Instrument Mass Histogram

While a histogram is useful to visualize the overall distribution of mass within the NICM, a cumulative distribution plot helps one better understand how a particular mass value compares to the rest of the sample. Figure 4-5 shows the same mass parameter data plotted as a cumulative distribution. For a given instrument mass along the logarithmic *x*-axis, the value along the *y*-axis shows the fraction of the sample that exceeds that mass. As an example, in the NICM database, 74% of the instruments exceeded 10 kg. Said from the perspective of an instrument developer, a developer who delivered a 10 kg instrument would have found that it "fit" within 74% of the mass envelopes of the sample. Figure 4-5 also shows the masses of a few notable Earth Science instruments as vertical dashed lines to put them in perspective of the data set:

1) Thermal Infrared Sensor (TIRS): purple
2) Clouds and Earth's Radiant Energy System (CERES): green
3) Total and Spectral Solar Irradiance Sensor (TSIS): light blue
4) CALIPSO Wide Field Camera (WFC): orange

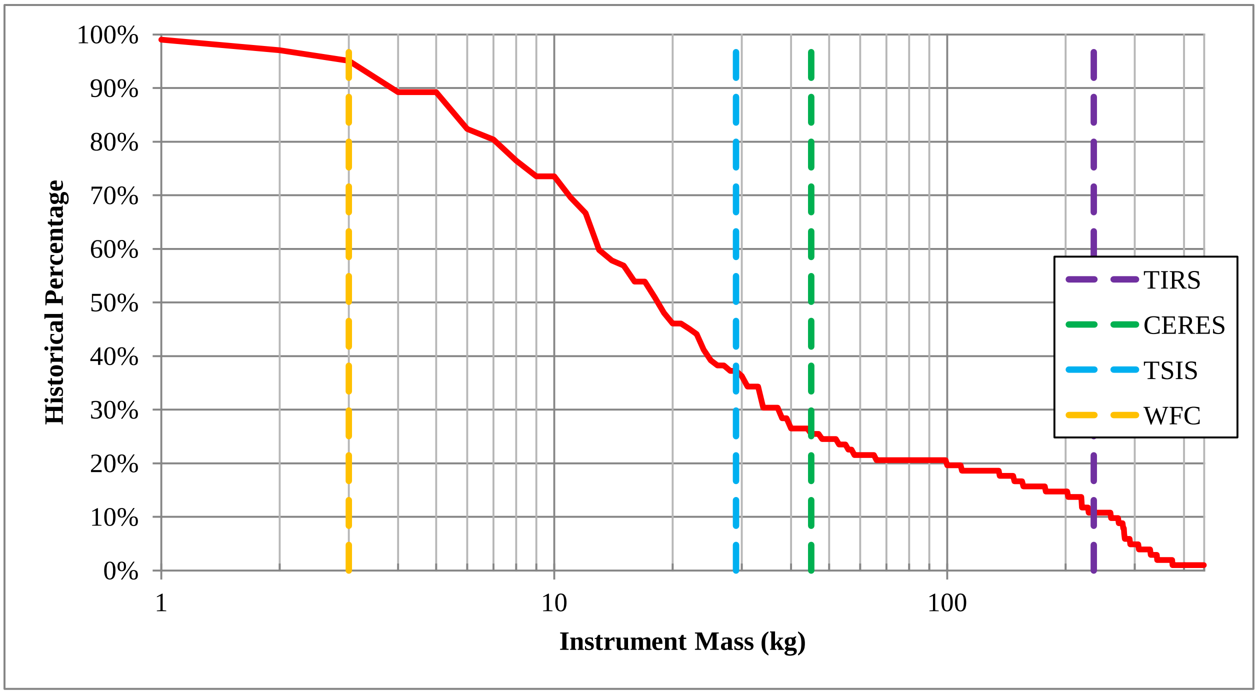


Figure 4-5: NICM Sample Mass Cumulative Plot

Readers should understand that these data are not representative of future HPO's, but only serve to provide a sense of relative standing for various instrument parameters. Figure 4-6 and Figure 4-7 show power and peak data rate, respectively.

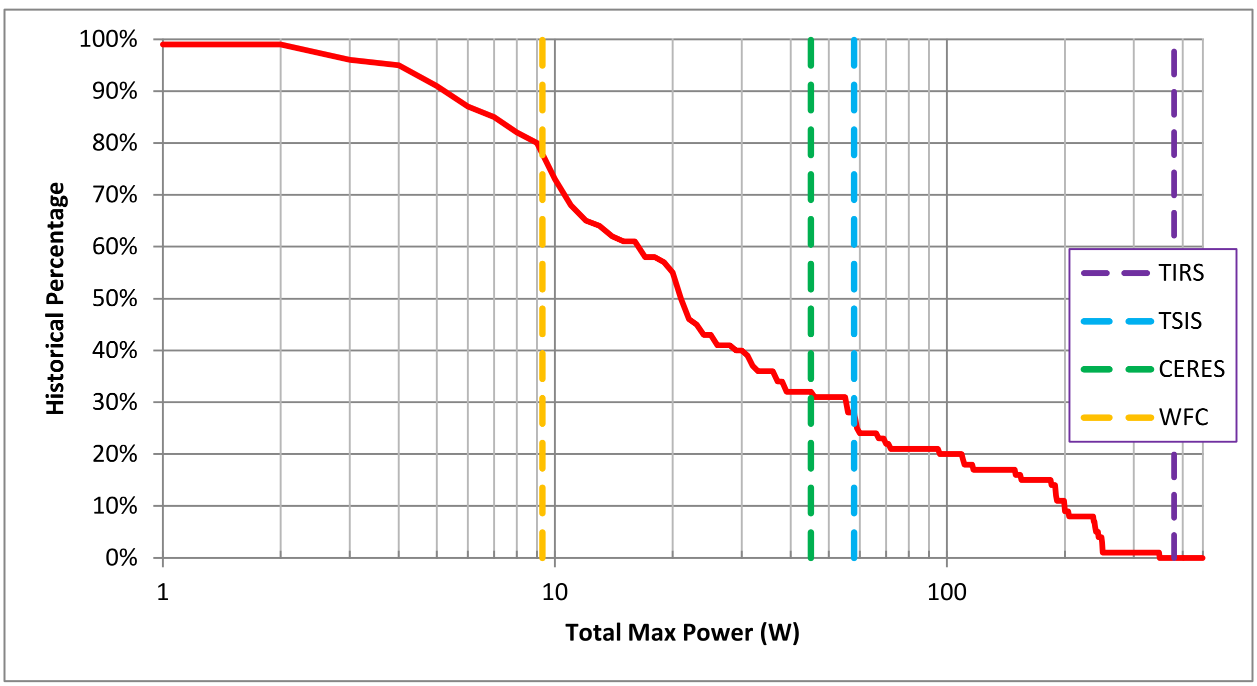


Figure 4-6: NICM Sample Power Cumulative Plot

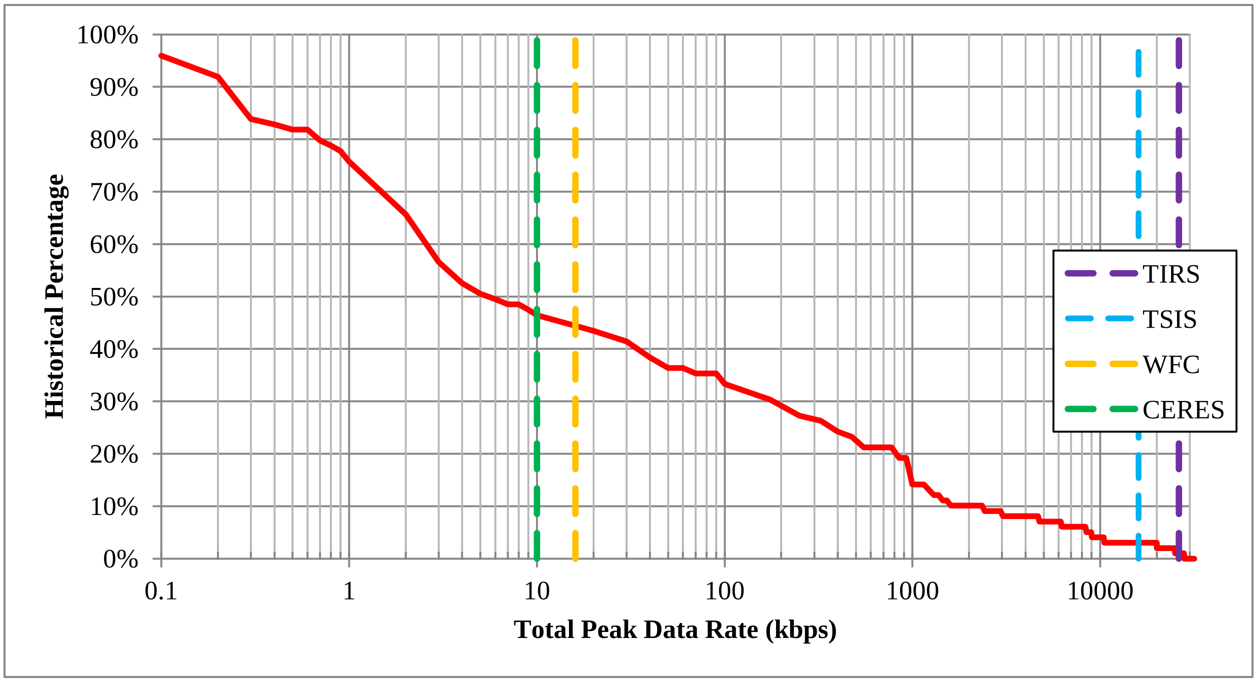


Figure 4-7: NICM Sample Data Rate Cumulative Plot

5.0 INSTRUMENT MODES

5.1 Mode Guidelines

5.1.1 Basic Modes

Instruments should support five basic modes of operation: OFF, INITIALIZATION, OPERATION, SAFE, and SURVIVAL (see Figure 5-1). Within the OPERATION mode, instruments may define additional sub-modes specific to their operation (e.g. STANDBY, DIAGNOSTIC, MEASUREMENT, etc.).

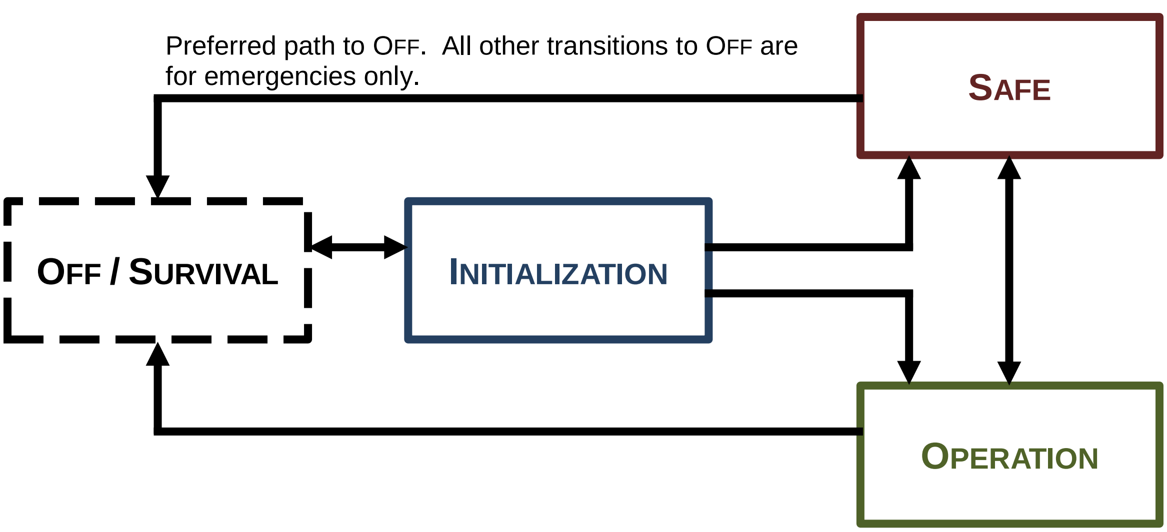


Figure 5-1: Instrument Mode Transitions

5.1.2 OFF Mode
The Instrument and the survival heaters are unpowered in the OFF mode.

5.1.2.1 *OFF Mode Power Draw*
The Instrument should draw no operational power while in OFF mode.

5.1.2.2 *Instrument Susceptibility to Unanticipated Power Loss*
The Instrument should be able to withstand the sudden and immediate removal of operational power by the Spacecraft at any time and in any instrument mode. This refers specifically to the sudden removal of operational power without the Instrument first going through an orderly shutdown sequence.

5.1.3 SURVIVAL Mode
The Instrument is unpowered and the survival heaters are powered-on in the SURVIVAL mode.

5.1.3.1 *Spacecraft Verification of Instrument Survival Power*
The Spacecraft should verify Instrument survival power is switched on during SURVIVAL mode.

5.1.3.2 *Post-Launch Instrument Survival Circuit Initiation*
The Spacecraft should enable power to the Instrument survival heater circuit(s) within 60 seconds after spacecraft separation from the launch vehicle, unless precluded by Spacecraft survival. The amount of time defined from spacecraft separation to enabling of the instrument survival heater circuit should be reviewed and revised as necessary after pairing with the host mission CONOPS, spacecraft and launch vehicle..

5.1.3.3 *Instrument Susceptibility to Unanticipated Transition to SURVIVAL Mode*
The Instrument should be able to withstand the sudden and immediate transition to instrument SURVIVAL mode by the Spacecraft at any time and in any Instrument mode. This refers specifically to the sudden removal of operational power without the Instrument first going through an orderly shutdown sequence and the sudden activation of the survival heater power circuit(s).

5.1.4 INITIALIZATION Mode
When first powered-on, the Instrument enters INITIALIZATION mode and conducts all internal operations necessary in order to eventually transition to OPERATION (or SAFE) mode.

5.1.4.1 *Power Application*
The Instrument should be in INITIALIZATION mode upon application of electrical power.

5.1.4.2 *Thermal Conditioning*
When in INITIALIZATION mode, the Instrument should conduct Instrument component warm-up or cool-down to operating temperatures.

5.1.4.3 *Command and Telemetry*
When in INITIALIZATION mode, the command and telemetry functions of the Instrument should be powered up first.

5.1.4.4 *Health and Status Telemetry*
When in INITIALIZATION mode, the Instrument should send the Spacecraft health and status telemetry.

5.1.5 OPERATION Mode
The Instrument OPERATION mode covers all nominal Instrument operations and science observations.

5.1.5.1 *Science Observations and Data Collection*
The Instrument should have one OPERATION mode for science observations and data collection. Within the OPERATION mode, an instrument may define additional sub-modes specific to their operation (*e.g.* STANDBY, DIAGNOSTIC, MEASUREMENT, etc.).

5.1.5.2 *Data Transmission*
When in OPERATION mode, the Instrument should be fully functional and capable of providing all health and status and science data originating within the instrument to the Spacecraft and ground operations team.

5.1.5.3 *Resources*
When in OPERATION mode, the Instrument should have all allocated Spacecraft resources available to it.

5.1.6 SAFE Mode
The Instrument SAFE mode is a combined Instrument hardware and software configuration meant to protect the Instrument from possible internal or external harm while making minimal use of Spacecraft resources (e.g. power).

5.1.6.1 *Data Collection and Transmission*
When in SAFE mode, the Instrument should limit data collection and transmission to health and status information only.

5.1.6.2 *Notification*
The Instrument should notify the Spacecraft when it has completed a transition to SAFE mode.

5.2 **Mode Transitions**

5.2.1 Impacts to other instrument and spacecraft bus
The Instrument should transition from its current mode to any other mode without harming itself, other instruments, or Spacecraft bus.

5.2.2 Preferred Mode Transitions
The Instrument should follow the mode transitions depicted in Figure 5-1. The preferred transition to OFF mode is through SAFE mode. All other transitions to OFF are to be exercised in emergency situations only.

5.2.3 SURVIVAL Mode Transitions
5.2.3.1 *Trigger*
The Spacecraft should transition the Instrument to SURVIVAL mode in the event of a severe Spacecraft emergency.

5.2.3.2 *Instrument Operational Power*
The Spacecraft should remove Instrument operational power during transition to SURVIVAL mode.

5.2.3.3 *Instrument Notification*
Transition to SURVIVAL mode should not require notification or commands be sent to the Instrument.

5.2.4 INITIALIZATION Mode Transitions
5.2.4.1 *Transition from OFF Mode*
The Instrument should transition from OFF mode to INITIALIZATION mode before entering either OPERATION or SAFE modes.

5.2.4.2 *Exiting initialization Mode*
When in INITIALIZATION mode, the Instrument should remain in INITIALIZATION mode until a valid command is received from the Spacecraft or ground operations team to transition to OPERATION (or SAFE) mode.

5.2.5 SAFE Mode Transitions
5.2.5.1 *Command Trigger*
The Instrument should transition to SAFE mode upon receipt of a command from the Spacecraft or ground operations team.

5.2.5.2 *Missing Time Message Trigger*
The Instrument should transition to SAFE mode upon the detection of 10 consecutive missing time messages.

5.2.5.3 *On-Orbit Anomaly Trigger*
The Instrument should transition to SAFE mode autonomously upon any instance of an Instrument-detected on-orbit anomaly, where failure to take prompt corrective action could result in damage to the Instrument or Spacecraft.

5.2.5.4 *Orderly Transition*
The Instrument should conduct all transitions to SAFE mode in an orderly fashion.

5.2.5.5 *Duration of SAFE Mode Transition*

The Instrument should complete SAFE mode configuration within 10 seconds after SAFE mode transition is initiated.

5.2.5.6 *Instrument Inhibition of SAFE Mode Transition*

The Instrument should not inhibit any SAFE mode transition, whether by command from the Spacecraft or ground operations team, detection of internal Instrument anomalies, or lack of time messages from the Spacecraft.

5.2.5.7 *Deliberate Transition from SAFE Mode*

When in SAFE mode, the instrument should not autonomously transition out of SAFE mode, unless it receives a mode transition command from the Spacecraft or ground operations team.

5.2.6 OPERATION Mode Transitions

5.2.6.1 *Trigger*

The Instrument should enter OPERATION mode only upon reception of a valid OPERATION mode (or sub-mode) command from the Spacecraft or ground operations team.

5.2.6.2 *Maintenance of OPERATION Mode*

When in OPERATION mode, the Instrument should remain in the OPERATION mode until a valid command is received from the Spacecraft or ground operations team to place the Instrument into another mode, or until an autonomous transition to SAFE mode is required due to internal Instrument anomalies or lack of time messages from the Spacecraft.

6.0 DATA INTERFACE GUIDELINES

6.1 Introduction

This section provides key data interface guidelines. For more detailed recommendations and best practices see Section 16.1: Data Interface Reference Material / Best Practices.

6.1.1 Goals

The CII Data Interface meets several design goals:

1) Low-power data transmission
2) High data rate transmission with scalable capacity
3) Lightweight messaging protocol
4) Leverage existing standards and practices as much as practical
5) Minimally intrusive to spacecraft provider and instrument designer

The use of SpaceWire allows the CII Data Interface to meet goals (1) and (2). SpaceWire offers higher data rates at much lower power consumption than common aerospace alternatives such as MIL-STD-1553 bus or RS-422 serial. While goal (3) is subjective, the CII Data Interface defines only five basic message types to communicate spacecraft status (ephemeris and time), instrument commands, their acknowledgement, engineering telemetry, and science data. Goal

(4) is set forth to avoid reinventing the proverbial wheel. The protocol itself uses or builds upon existing SpaceWire and packet communication standards, including the industry standard Consultative Committee for Space Data Systems (CCSDS) for packet sequencing and control. Goals (1)-(4) taken together lead to Goal (5) and help minimize the implementation and integration impact on both the spacecraft provider and instrument designer.

6.1.2 Assumptions

The CII Data Interface guidelines assume the Instrument has some reasonable autonomous capabilities. In particular, the Instrument includes an onboard processor and its own capabilities for packet processing, telemetry monitoring, command and data handling, stored command sequences, (science) data storage, data compression, and data playback. Further, when necessary, the Instrument is operated by a ground team with the spacecraft acting largely as a passive relay of instrument commands and engineering and science data.

Autonomous instruments and passive relay spacecraft require far fewer changes to spacecraft flight and ground software, which helps maximize the number of HPO’s that can be realized.

6.1.3 Drivers

The CII Data Interface guidelines recommend SpaceWire to achieve a low-power-consumption, high-data-rate interface with a scalable capacity in the range of 2–400 Mbps. SpaceWire systems have achieved industry acceptance for LEO spacecraft.

The use of CCSDS packets for data communication is common practice across aerospace flight and ground data systems. CII Data Interface messages communicate standard information about Spacecraft and Instrument status, time, telemetry, and science data in a well-accepted fashion (*e.g.* CRCs for error detection, command acknowledgements to assist in fault detection, SAFE mode for contingency situations). These factors can help reduce the overall costs and technical challenges of integrating the CII data interface guidelines into existing mission operating systems.

6.1.4 Conventions

To remain consistent with the CCSDS specifications referenced throughout the data portions of this specification, conventions for data packet diagrams, byte orderings, and byte numbering follow those established in Section 1.6.3 of the Space Packet Protocol Blue Book (CCSDS 133.0-B-1). The CCSDS standards use the term “octet” to refer to an 8-bit word while the SpaceWire standards use the term “byte” to refer to the same quantity. This document uses the term “byte” to refer to a grouping of 8 bits.

All binary numbers are prefixed with “`0b`”. All hexadecimal numbers are prefixed with “`0x`”.

Instrument modes are rendered in SMALL CAPS. For more information on CII instrument modes see Section 5.0: Instrument Modes.

6.2 Data Interface

6.2.1 Data Bus

The Instrument should adopt a common instrument-to-spacecraft data interface (e.g. MIL-STD-1553B, RS-422, or SpaceWire) for commanding, telemetry, and science data transmission.

Rationale: MIL-STD-1553B, RS-422, and SpaceWire are the most common LEO data interfaces.

Considerations: Common spacecraft command and data buses include MIL-STD-1553B, RS-422, and SpaceWire. MIL-STD-1553B typically offers a total data rate less than 1 Mbps shared among all devices on the data bus (*e.g.* primary spacecraft and instruments). Based on feedback from commercial spacecraft vendors, hosted payloads can expect a data rate of only kilobits per second or even bits per second in the best case. RS-422 data rates range from 100 Kbps to a maximum of 10 Mbps in the (theoretical) best case. SpaceWire provides high data-rate asynchronous serial communications in the range of 2–400 Mbps. SpaceWire power consumption and heat dissipation is significantly lower than either MIL-STD-1553B or RS-422.

The space community has been and deployed SpaceWire worldwide. ESA, JAXA, and NASA missions using SpaceWire include: BepiColombo Mercury Planetary Orbiter (MPO) and Mercury Magnetospheric Orbiter (MMO), Gaia, James Webb Space Telescope (JWST), Herschel, Corot, Gravity field and steady-state Ocean Circulation Explorer (GOCE), Rosetta, Mars Express, Non-thermal Energy eXploration Telescope (NeXT), Lunar Reconnaissance Orbiter (LRO), and Swift. Several commercial vendors offer support, engineering test equipment, and flight-qualified SpaceWire hardware and software.

For detailed information on the data standards described, see Section 2.0: Reference Documents.

6.2.2 Data Bus Redundancy (optional)

The Instrument may employ two or four data links for communication with the Spacecraft in a simple parallel or cross-strapped configuration.

6.2.2.1 *Active Links*

The Instrument should only activate one link at any given time.

6.2.2.2 *Secondary Links*

The Instrument should use secondary link(s) in the event that the primary active link becomes inoperative or the Instrument swaps to second string components.

Rationale: Hosted payload instruments are expected to fall into NASA risk classifications C where reduced mission assurance standards are permitted. Furthermore, instrument providers will likely *not* be able to afford data link-redundancy and all of its associated hardware and software complexity. Data bus redundancy is therefore purely optional. Single-string instrument providers are strongly encouraged to minimize risk using other measures (e.g. design, analysis, parts selection).

6.2.3 Packet Data

Data transmitted over the Instrument-to-Spacecraft data interface should be encoded as discrete digital packet data. For more detailed information and recommended best practices, see Section 16.1.6: CII Messages, Packet Formats, and Protocol.

Rationale: Best practice.

6.2.4 Packet Data Control

Data transmitted over the Instrument-to-Spacecraft data interface should adopt CCSDS primary and secondary headers [CCSDS 133.0-B-1] to communicate basic packet control and time information.

Rationale: CCSDS headers are the established industry standard for space packet communication. The CCSDS primary header provides basic packet control data (e.g. packet sequence count, packet length, virtual channels, *etc*.). The CCSDS secondary header provides time information.

6.3 Spacecraft Status

6.3.1 Spacecraft Time and Ephemeris

The Instrument should receive time and ephemeris information from the spacecraft over the Instrument-to-Spacecraft data interface at regular intervals, no less than once per second. The precise resolution and specific period of such information will be negotiated between Spacecraft and Instrument providers.

Rationale: Instruments require time and position information for nominal operations and to coordinate and correlate science observations.

6.4 Command and Telemetry

6.4.1 Command Verification

The Instrument should reject any command that does not pass command verification or is otherwise deemed invalid.

Rationale: Command verification is a best practice. It allows the Spacecraft and ground operations team to devise and operate fault detection, isolation, and recovery (FDIR) procedures.

6.4.2 Command Acknowledgement

The Instrument should explicitly acknowledge the receipt of and indicate the execution status of all commands.

Rationale: Explicit commanding and acknowledgement is best practice. It allows the Spacecraft and ground operations team to devise and operate FDIR procedures.

6.4.3 Command Dictionary

A command dictionary should document all Instrument commands, the format and detail of which will be negotiated with the Spacecraft Provider.

6.4.3.1 *Command (SAFE Mode)*

The Instrument should enter SAFE mode when it receives an Enter SAFE Mode command from the Spacecraft or ground operations team.

6.4.3.2 *Command (SAFE Mode) Priority*

The Instrument should immediately process the Enter SAFE Mode command message ahead of any queued, but not yet processed, commands.

Rationale: Transition to SAFE mode is a high priority event. For the Instrument's, and possibly Spacecraft's, own health and safety, SAFE mode entry should be immediately accommodated.

6.4.3.3 *Command (Data Flow Control)*

Instrument providers should define commands to disable, enable, and resume the transmission of Instrument telemetry and Instrument science data.

Rationale: Data flow control is best practice. It allows the Spacecraft and ground operations team to devise and operate FDIR procedures.

6.4.4 Telemetry Dictionary

A telemetry dictionary should document all Instrument telemetry, the format and detail of which will be negotiated with the Spacecraft Provider.

6.5 **Data Management**

6.5.1 Onboard Science Data Storage

The Instrument should be responsible for its own science data onboard storage capabilities.

6.5.2 Onboard Science Data Compression

The Instrument should be responsible for its own science data onboard compression capabilities.

6.5.3 Science Data Playback

The Instrument should be responsible for its own science data playback capabilities.

6.5.4 Science Data Playback Coordination

The Instrument should coordinate science data playback with the spacecraft operations team.

Rationale: Buffering all data on the Instrument imposes no excess storage requirements on the Spacecraft's capacity to accommodate the Instrument and its data. A spacecraft need only enough buffer capacity to relay Instrument telemetry and science data transmission. As a result, far fewer modifications are required to Spacecraft flight software and operations procedures.

6.6 **Flight Software**

6.6.1 Flight Software Update

Instrument control flight software should be updatable on orbit through ground command.

Rationale: On-orbit flight software updates are a best practice that facilitates improvements and/or workarounds deemed necessary through operational experience.

6.6.2 Flight Software Update (Partial)

Individual memory addresses of instrument control software should be updatable on orbit through ground command.

Rationale: On-orbit flight software updates are a best practice that facilitates improvements and/or workarounds deemed necessary through operational experience.

7.0 ELECTRICAL POWER GUIDELINES

7.1 Introduction

The following electrical power guidelines define a common basis for the provision of electrical power by spacecraft hosts to potential hosted payload Earth science instruments. This is necessary to progress both the design of any instrument prior to the selection of a hosted payload opportunity and launch vehicle and to progress the design of the host payload accommodations. The major sources from which these guidelines were developed are the *General Interface Requirements Document (GIRD) for EOS Common Spacecraft/Instruments, Electrical Grounding Architecture for Unmanned Spacecraft*, (NASA-HDBK-4001), and *Electrical Bonding for Launch Vehicles, Spacecraft, Payloads and Flight Equipment* (NASA-STD-4003).

7.1.1 Goals

The goal of this section is to specify the characteristics, connections, and control of the Spacecraft power provided to each Instrument as well as the guidelines that each Instrument should meet at this interface. This section applies equally to the Power Buses and the Survival Heater Power Buses. Section 16.2 of this document details “best practices” to be implemented for the achievement of the noted goals

7.1.2 Assumptions

These electrical power guidelines assume the following:

1) The Spacecraft Electrical Power Subsystem (EPS) is unregulated (sun regulated) and provides power at a nominal voltage of +28 volts direct current (VDC)
2) The Instrument EPS employs selective redundancy as part of risk classification C.

7.1.3 Drivers

The guidelines define a 50 W increment, stepped power provision architecture. This incremental provision “right sizes” the amount of power draw for any particular instrument and drives the implementation of a modular instrument electrical power subsystem. Selective redundancy of power provision also drives the implementation of a modular instrument electrical power subsystem. The use of space flight qualified D connectors for power connections drives the use of a low cost and readily available connector.

7.2 **Electrical Interface**

All guidelines in this section should be met at the electrical interface.

7.2.1 Electrical Interfaces

The electrical interfaces (see Figure 7-1) should include the following:

1) Operational Power Interface
2) Survival Heater Power Interface
3) Grounding Interface

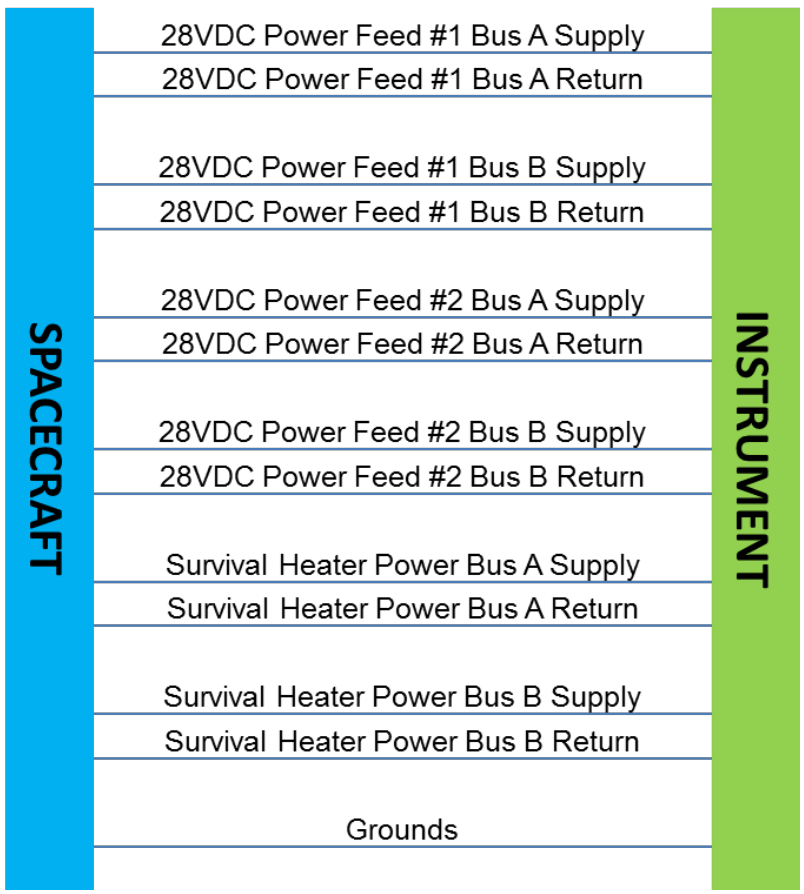


Figure 7-1: Spacecraft-Instrument Electrical Interface

7.3 **Power Specifications**

This section specifies the characteristics, connections, and control of the Spacecraft power provided to each Instrument as well as the requirements that each Instrument must meet at this interface. This section applies equally to the Power Buses and the Survival Heater Power Buses.

7.3.1 Instrument Power Harness

Instrument power harnesses should be appropriately sized to support the peak allocated power levels and both spacecraft and Instrument fusing.

7.3.2 Orbital Average Power Consumption
Orbital average power (OAP) is the total power consumed averaged over 2 or more orbits.

7.3.3 Peak Power Consumption
Peak power is the maximum power consumed averaged over any 10 ms period.

7.3.4 Allocation of Instrument Power
Each Instrument mode should be allocated peak not to exceed (NTE) and average NTE power shown in Table 7-1.

Rationale: The Level 1 guideline defines power allocation for the OPERATION mode. The assumption that the instrument requires 100% of the power required in the OPERATION mode defines the power allocation for the ACTIVATION mode. The assumption that the instrument requires 50% of the power required in the OPERATION mode defines the power allocation for the SAFE mode. The assumption that the instrument only requires survival heater power defines the power allocation for the SURVIVAL mode.

Table 7-1: Instrument Power Allocation

Mode	Peak (W)	Average (W)
OFF	0	0
SURVIVAL	40	20
ACTIVATION	200	100
SAFE	100	50
OPERATION	200	100

7.3.5 Voltage
7.3.5.1 *Primary Instrument Voltage*
The Spacecraft power bus should provide voltage within the range of 28 ±6 VDC, including ripple and normal transients as defined below, and power distribution losses due to switching, fusing, harness and connectors.

7.3.5.2 *Unannounced Removal of Power*
Unannounced removal of power should not cause damage or degraded Instrument performance.

7.3.5.3 *Reversal of Power*
Reversal of power (positive) and ground (negative) should not cause damage or degraded Instrument performance.

7.3.6 Power-Up and Power-Down
The Instrument should be designed for nominal and anomalous power-up and power-down sequences where the bus voltage change from +28 to 0 VDC or from 0 to +28 VDC will be a step function.

7.3.7 Overcurrent Protection

The Spacecraft should protect of the Spacecraft power system by providing overcurrent protection devices on each Instrument power connection.

7.3.7.1 *Overcurrent Protection Device Type*

The type of overcurrent protection device(s), such as latching current limiters, fuses, or other, should be consistent for each Instrument.

7.3.7.2 *Abnormal Operation Steady-State Voltage Limits*

The Instrument should survive under abnormal conditions, steady-state voltages (V) in the range of 0 to 50 VDC, without permanent degradation.

7.3.7.3 *Power Source Impedance*

The Spacecraft power source impedance should be as indicated in Table 7-2.

Table 7-2: Power Source Impedance

Maximum Source Impedance (Ω)	Frequency
0.1	1 Hz to 1 kHz
1.0	1 kHz to 20 kHz
2.0	20 kHz to 100 kHz
20.0	100 kHz to 10 MHz

7.3.8 Current Transients

7.3.8.1 *Instrument Turn-on Transients Limit*

For turn-on, the transient current on any Power Feed bus should not exceed 100 percent (that is, two times the steady state current) of the maximum steady-state current and should not be greater than 50 ms surge duration. There is no turn-on transient restriction on the Survival Heater Bus.

7.3.9 Power Connections

The Spacecraft should provide the Instrument separate connections to each redundant power source, as illustrated in Figure 7-1. These connections are designated as Power Bus #1, Power Bus #2, and the Survival Heater Power Bus. All buses have prime and redundant sides designated as Power Bus A and Power Bus B. Each power bus connection/return will supply a maximum of 50 W OAP (100 W instantaneous power) with selective redundancy.

7.4 **Grounds, Returns and References**

To ensure that the Instrument grounding configuration will be compatible with the spacecraft, the Instrument grounds should be wired as described in the following sections. Instrument grounding should appear in the Electrical Interface drawing.

7.4.1 Power Leads and Returns

Each Instrument primary power service should have a distinct and isolated return.

7.4.2 Power Reference
The Signal Reference Plane is the Spacecraft conducting plate or other structure to which all ground planes are connected. The Primary Power Reference is the reference point for Spacecraft voltage control. The Spacecraft should connect all power returns to the Signal Reference Plane only at the Spacecraft Primary Power Reference Point.

7.4.3 Grounding, DC Isolation
In each unit before connection to the harness, a DC isolation of at least 1 MΩ should exist from each line of primary power to each line of secondary power, and between all grounds which are not common.

7.4.4 Grounding, AC Isolation
AC isolation should be at least 0.5 μF.

7.4.5 Grounding, DC Resistance
Grounds which are common should have a measured DC resistance of less than 10 mΩ between connector pins.

7.4.6 Shield Grounding
Multi-point grounding should be provided as necessary for shielding connections. The reference for shields should be chassis ground.

7.4.7 Component Bounding
The Instrument component housings supporting or containing electrical/electronic assemblies should be electrically bonded to chassis to preclude electrostatic discharge from causing inadvertent initiation of deployments or damage to sensitive electronics.

7.4.7.1 *Component Ground Connection*
The Spacecraft Contractor should electrically bond the chassis ground terminal of each Instrument component to the Instrument ground plane.

7.4.7.2 *Component Bonding Straps*
Where direct bonding is not possible and/or movable metal-to-metal joints are present, bonding straps should be used.

7.4.7.3 *Component Bonding Strap Design Guidance*
Bonding straps should be designed and applied in accordance with NASA-STD-4003.

7.4.8 Ground Loops
The instrument electrical power subsystem grounding architectures should be designed to preclude ground loops.

7.5 **Harness**

7.5.1 Spacecraft to Instrument Harness Provider
The Spacecraft Provider should furnish all Spacecraft and Spacecraft-to-Instrument harnessing.

7.5.2 Intra-Instrument Harness Provider
The Instrument Provider should furnish all Instrument harnessing.

7.5.3 Harness Hardware Documentation
The appropriate Electrical Interface Control Drawing should document all harnesses, connectors, ground straps, and associated service loops.

7.5.4 Harness Wiring Requirements
All requirements for harness construction, pin-to-pin wiring, cable type, etc. should be documented in the appropriate Electrical Interface Control Drawing.

7.5.5 Power Harnessing
7.5.5.1 *Power Routing and Shielding*
The delivery of Instrument power by the Spacecraft should be via twisted conductor (pair, quad, etc.) cables with both power and return leads enclosed by an electrical overshield.

7.5.5.2 *Power Shield Bonding*
The Instrument should provide the capability to form a low impedance electrical path (a bond) between the power conductor overshield and the Instrument chassis via a 360° connector back shell at the Instrument primary power input.

8.0 MECHANICAL GUIDELINES

8.1 Introduction

8.1.1 Goals
The key goals of mechanical guidelines are:

1) To allow interfacing of a hosted payload instrument to be minimally intrusive to spacecraft provider.
2) Once paired for a mission, a Mechanical Interface Control Document (MICD) should establish implementation details between Spacecraft and Instrument Providers.

8.1.2 Assumptions
The CII mechanical guidelines make several assumptions:

1) The Instrument is a self-contained assembly with on board electronics and thermal control.
2) The MICD will define instrument location with fields-of-view (FOV) that equal or exceed the Instrument science and radiator requirements at the pairing of Spacecraft and Instrument.

8.1.3 Drivers
1) Parametric studies of a variety of similar class instruments drive mass guidelines
2) Engineering analysis determines guideline payload volume based on mass guidelines and comparisons to spacecraft envelopes in the NASA RSDO catalog.

8.2 **Instrument Envelopes**

8.2.1 Dimensions

Instrument components in the launch and on-orbit configurations should be contained within the detailed instrument envelope of 0.4m×0.5m×0.85m (H×W×L) for a 100 kg (max) instrument or 0.3m×0.4m×0.7m for a 50 kg instrument.

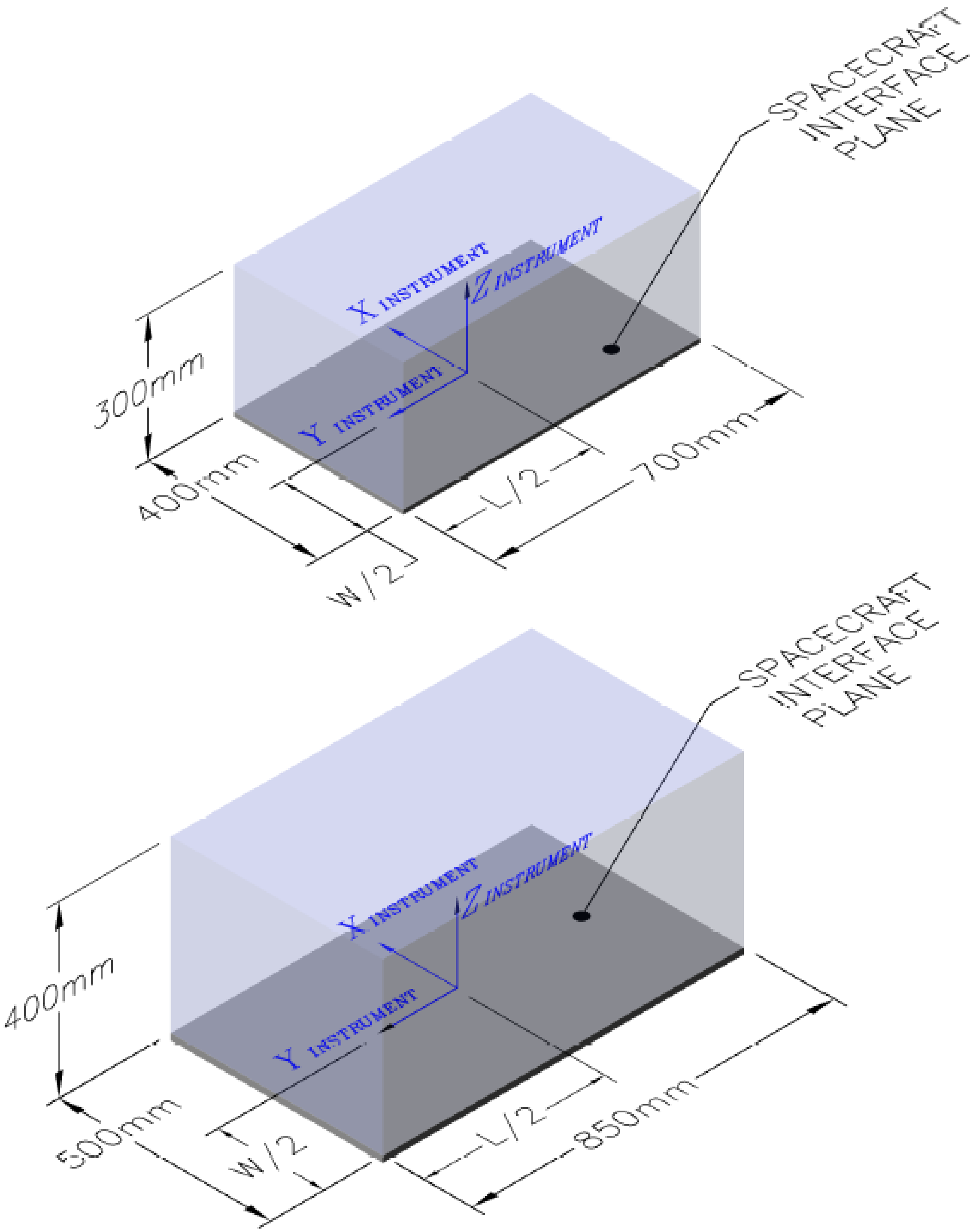


Figure 8-1: Spatial Envelopes for 50 kg and 100 kg Instruments with Instrument Payload Coordinate Frame

8.3 Instrument Mass

8.3.1 Mass Limit

The Instrument mass should be less than or equal to 100 kg.

8.3.2 Mass Centering

For a 50 kg Instrument, the center of mass should be less than 4 cm radial distance from the $Z_{instrument}$ axis and less than 15 cm above ($+Z_{instrument}$) the Instrument mounting plane.

For a 100 kg Instrument, the center of mass should be less than 5 cm radial distance from the $Z_{instrument}$ axis and less than 20 cm above ($+Z_{instrument}$) the Instrument mounting plane.

8.3.3 Inertia

The MICD should document moments and products of inertia.

8.4 Instrument Mounting

8.4.1 Mounting

The Instrument should be hard mounted to the spacecraft unless the Instrument Provider demonstrates that kinematic mounts are required.

8.4.2 Functionality in 1 g Environment

The Instrument should be fully functional in any 1 g orientation.

8.5 Mechanisms

8.5.1 Caging During Launch

Instrument mechanisms that require restraint during launch should be caged during launch without requiring power to maintain the caged condition.

8.6 Structural Characteristics

8.6.1 Minimum Fixed-Base Frequency - mass of less than 100 kg

The Instrument component with a mass of less than 100 kg, configured for launch, should have a fixed-base frequency greater than 35 Hz. Fixed-based is defined as follows: Each mounting point should be constrained in those degrees of freedom which are rigidly attached to the Spacecraft, and should be free in those degrees of freedom for which kinematic mounts or flexures provide flexibility.

9.0 THERMAL GUIDELINES

9.1 Introduction

9.1.1 Goals

The key goal of thermal guidelines is:

1) To minimize thermal perturbations on the spacecraft system or let the spacecraft perturb the instrument design

9.1.2 Assumptions

The thermal guidelines assume:

1) The Instrument is nadir-pointed in order to have a clear view of the earth and/or earth's limb.
2) The instrument is thermally isolated from the spacecraft and the neighboring instruments.
3) The instrument volume is capped at 0.4m×0.5m×0.85 m.
4) The mass is capped at 100kg.

9.1.3 Thermal Interface Drivers

A key thermal interface driver is:

1) The required instrument radiator size can vary by a factor of four depending on the location of the instrument on the spacecraft

9.2 Thermal Design at the Mechanical Interface

The Instrument thermal design should be decoupled from the Spacecraft at the mechanical interface spacecraft and neighboring payloads as much as possible.

The conductive heat transfer at the instrument-spacecraft mechanical interface should be less than 15 W/m^2 or 4 W.

The net radiative heat transfer between the instrument and spacecraft should be less than 3 W.

The MICD should document the isolation implementation.

9.3 Thermal Design

9.3.1 Responsibility

The Instrument Provider should be responsible for the thermal design of the instrument.

9.3.2 Instrument Placement on Spacecraft

The Instrument Designer should expect the instrument to be mounted to any of the six sides of the spacecraft (nadir, zenith, East, West, North, and South).

9.4 Temperature Maintenance

9.4.1 Responsibility

The Instrument Provider should be responsible for maintaining all instrument temperature requirements.

9.4.2 Allowable Temperature Range

The spacecraft temperature range at the mounting interface should be: [0°C, +30°C] operating and [-20°C, +50°C] non-operating.

9.5 Thermal Hardware

9.5.1 Thermal Control Hardware Responsibility

The Instrument Provider should provide and install all Instrument thermal control hardware including blankets, temperature sensors, louvers, heat pipes, radiators, and coatings.

9.6 Thermal Design Verification and Validation

The Instrument thermal control system's ability to maintain hardware within allowable temperature limits should be verified and validated empirically by thermal balance testing and if necessary, by analysis for conditions that cannot be ground tested.

10.0 ENVIRONMENTAL GUIDELINES

10.1 Introduction

The following environmental guidelines are established to provide potential Earth Science instrument and HPO spacecraft providers a common design and analysis basis in order to progress the design of instrument prior to selection of a host spacecraft and launch vehicle for same. The major sources for these are the *General Environmental Verification Standard (GEVS) for GSFC Flight Programs and Projects*, the *General Interface Requirements Document (GIRD) for EOS Common Spacecraft/Instruments*) and the latest revision of MIL-STD-461, *Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment*.

10.2 Goals

The goal of this section is provide sufficient definition of the environments to which an Earth science instrument will be subjected during integration, testing, launch, and operation to allow a successful progression of the instrument design from conception until the pairing of the Instrument with an HPO spacecraft and launch vehicle.

10.3 Drivers

The following drive these environmental guidelines:

1) The Instrument will be paired with a defined Spacecraft no later than the Instrument Preliminary Design Review (PDR)
2) The Instrument/Spacecraft will be paired with a defined launch vehicle no later than PDR.

10.4 Launch Environment

The Instrument should be designed for the following launch environments.

10.4.1 Launch Pressure Profile

The Instrument should withstand an atmospheric pressure decay rate less than or equal to 4 kPa/s (30 Torr/s).

10.4.2 Explosive Atmosphere

The instrument should operate in the presence of flammable vapors without initiating an explosion or fire.

10.4.3 Atmospheric Pressure Range

The instrument should be designed for a range of atmospheric pressures from 101 kPa (~760 Torr) at sea level to 1.3×10^{-15} kPa (10^{-14} Torr) in space. This guideline defines the level of vacuum that an instrument will encounter in an orbital environment. For verification testing, the vacuum test level should be $\leq 1\times10^{-5}$ Torr

10.5 **Dynamics Environment**

Sinusoidal vibration design guidelines cover low-frequency (typically 5-50 Hz) launch vehicle-induced transient loading. The random vibration design guidelines are derived from: (a) launch vehicle-induced acoustic excitations during liftoff, transonic and max-q events; and (b) mechanically transmitted vibration from the engines during upper stage burns. Acoustic design guidelines are based on maximum internal payload fairing sound pressure level spectra. Pyroshock design guidelines are intended to represent the structurally transmitted transients from explosive devices used to achieve various separations, including spacecraft separation from the upper stage motor.

10.5.1 Quasi-Static Acceleration

The Instrument should survive and fulfill its mission after the application of a launch vehicle-induced quasi-static acceleration environment represented by the Mass Acceleration Curve (MAC) defined in Figure 10-1. The "Mass" is the mass of the entire instrument or any component of the instrument.

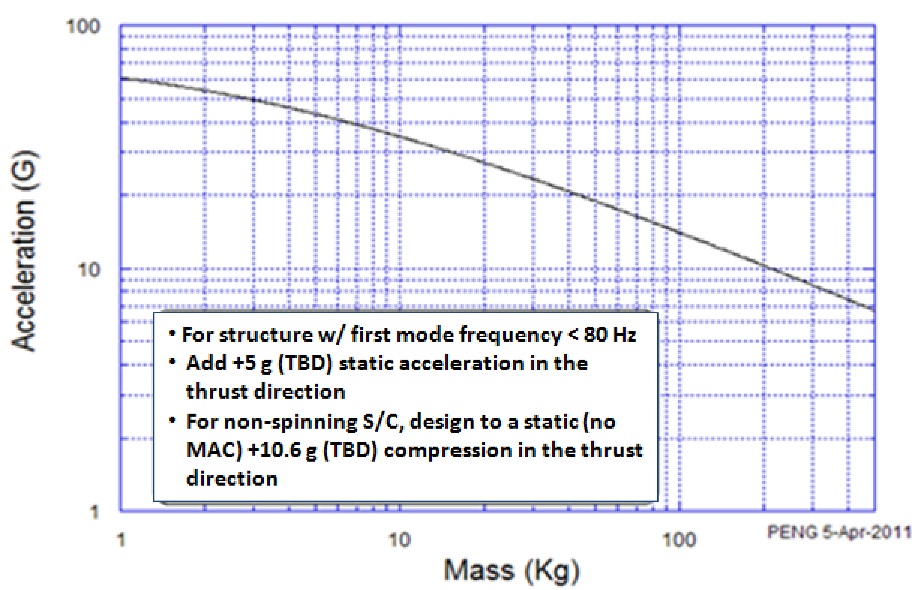


Figure 10-1: Mass Acceleration Curve

The MAC applies to worst single direction, which might not be aligned with coordinate directions, to produce the greatest load component (axial load, bending moment, reaction component, stress level, etc.) being investigated.

10.5.2 Sinusoidal Vibration

The Instrument should survive and fulfill its mission after application of launch vehicle-induced transient environments represented by the sinusoidal vibration environment defined in Table 10-1.

Table 10-1: Sinusoidal Vibration Environment

Frequency (Hz)	Displacement or Acceleration Amplitudes	
	Acceptance	Qualification
5 – 18	8 mm (0.31 inches) DA	12 mm (0.48 inches) DA
18 – 50	5.6 g peak	8 g peak
One sweep from 5 to 50 Hz at a rate of 4 octaves/minute except from 25 to 35 Hz at a rate of 1.5 octaves/minute		

Double Amplitude (DA) is the overall value from the highest maximum to the lowest minimum of a waveform in one period and is equivalent to twice the peak amplitude.

10.5.3 Random Vibration

The instrument should survive and fulfill its mission after application of launch vehicle-induced transient environments represented by the random vibration environment defined in Table 10-2. All flight article test durations are to be 1 minute per axis. Non-flight article qualification test durations are to be 2 minutes per axis.

Table 10-2: Random Vibration Environment

Zone/Assembly	Frequency (Hz)	Protoflight / Qualification	Acceptance
Instrument	20	0.026 g^2/Hz	0.013 g^2/Hz
	20 – 50	+6 dB/octave	+6 dB/octave
	50 - 800	0.16 g^2/Hz	0.08 g^2/Hz
	800 - 2000	-6 dB/octave	-6 dB/octave
	2000	0.026 g^2/Hz	0.013 g^2/Hz
	Overall	14.1 g_{rms}	10.0 g_{rms}

Table 10-2 represents the random vibration environment for instruments with mass less than or equal to 25 kg. Instruments with mass greater than 25 kg may apply the following random vibration environment reductions:

1) The acceleration spectral density (ASD) level may be reduced for components weighing more than 25 kg according to:

$$ASD_{new} = ASD_{original}*(25/M)$$

where M = instrument mass in kg

2) The slope is to be maintained at ±6 dB/octave for instruments with mass less than or equal to 65 kg. For instruments greater than 65 kg, the slope should be adjusted to maintain an ASD of 0.01 g^2/Hz at 20 Hz and at 2000 Hz for qualification testing and an ASD of 0.005 g^2/Hz at 20 Hz and at 2000 Hz for acceptance testing.

10.5.4 Acoustic Noise

The instrument should survive and fulfill its mission after application of launch vehicle-induced transient environments represented by the acoustic noise spectra defined in Table 10-3. The acoustic noise design requirement for both the instrument and its assemblies is a reverberant random-incidence acoustic field specified in 1/3 octave bands. The design/qualification exposure time is 2 minutes; PF/FA exposure time is one minute.

Table 10-3: Acoustic Noise Environment

1/3 Octave Band Center Frequency (Hz)	Design/Qual/PF Sound Pressure Levels (dB re 20 μPa)	Flight Acceptance Sound Pressure Levels (dB re 20 μPa)
31.5	122.5	119.5
40	125.5	122.5
50	129.5	126.5
63	131.0	128.0
80	131.5	128.5
100	132.5	129.5
125	133.0	130.0
160	133.0	130.0
200	133.5	130.5
250	134.5	131.5
315	135.5	132.5
400	134.5	131.5
500	131.0	128.0
630	128.0	125.0
800	125.0	122.0
1000	123.0	120.0
1250	121.0	118.0
1600	120.0	117.0
2000	119.5	116.5
2500	119.0	116.0
3150	118.0	115.0
4000	116.5	113.5
5000	114.0	111.0
6300	110.0	107.0
8000	106.0	103.0
10000	103.0	100.0
Overall	**143.8**	**140.8**

10.5.5 Mechanical Shock

The instrument should survive and fulfill its mission after application of spacecraft to launch vehicle separation or other shock transient accelerations represented by Figure 10-2 using Q = 10. The time-history of the Shock Response Spectrum (SRS) should be oscillatory and decay to within 10% of the peak value within 20 ms. Spacecraft to launch vehicle separation or other shock transients are defined at the instrument/spacecraft interface and the instrument should be designed to survive two shocks. The guideline assumes that shock transients are generated externally to the instrument and attenuated to the level defined at the notated interface.

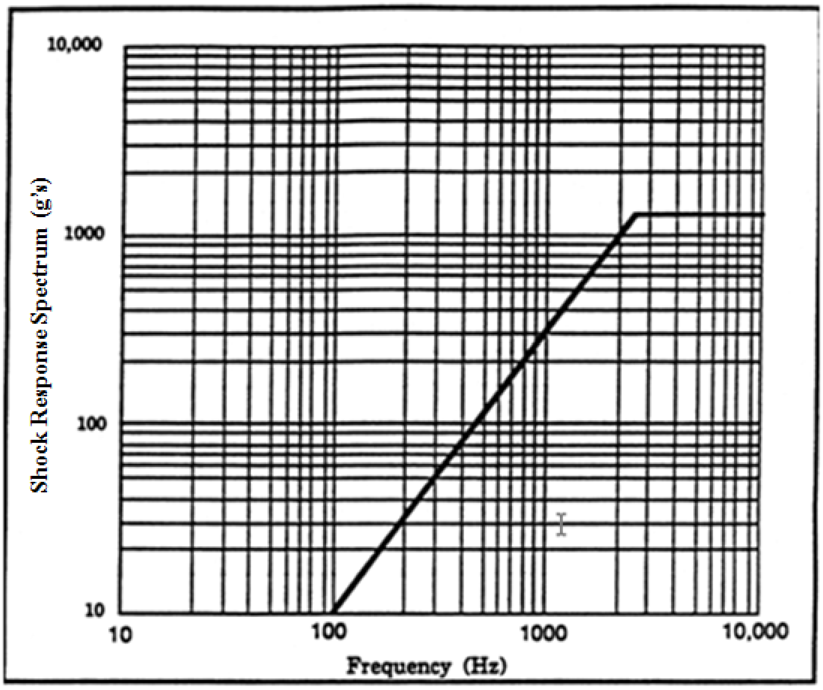


Figure 10-2: Mechanical Shock Environment

10.5.6 Orbital Acceleration

The Instrument should withstand a maximum acceleration of 0.015 g on orbit without permanent degradation of performance.

10.6 Thermal Environment

10.6.1 Ground Handling Temperatures

The Instrument should maintain assembly allowable flight temperatures within the non-operating temperature range in accordance with Section 9.4.2 during ground handling up to and including pre-launch fairing enclosure.

10.6.2 Ascent Temperatures

The Instrument should maintain assembly allowable flight temperatures within the non-operating temperature range in accordance with Section 9.4.2 during ascent.

10.6.3 Corona

The instrument should prevent corona or other forms of electrical breakdown at pressures less than 1.33×10^{-6} kPa (10^{-5} Torr). RF/high voltage circuitry is subject to multipacting/arcing damage at critical pressure.

10.6.4 Thermal Radiation

The Instrument should maintain assembly allowable flight temperatures within the non-operating temperature range in accordance with Section 9.4.2 during the mission when exposed to the worst case expected thermal radiation environments defined in Table 10-4.

Table 10-4: Thermal Radiation Environment

Mission Phase	Solar Flux	Earth Albedo	Earth IR (Long Wave)
Earth Orbit	1290 to 1420 W/m^2	0.275 to 0.375	222 to 243 W/m^2

10.6.5 Thermal Radiation

The thermal control design should control the instrument temperatures within the Protoflight levels when exposed to the applicable thermal radiation levels in Section 10.6.4 with a 20% test margin applied to the high levels.

10.6.6 Electronics Operational Temperatures

All instrument electronic assemblies should be designed to operate within specification at the greater of the launch vehicle payload fairing temperature range (typically -20°C to +75°C) or the allowable flight temperature plus 25°C.

10.7 Electromagnetic Interference & Compatibility Environment

The Electromagnetic Interference and Electromagnetic Compatibility (EMI/EMC) environments are based upon the latest revision to MIL-STD-461. More conventional, and typical of heritage, EMI/EMC environmental requirements are based upon revision C of the noted specification with testing requirements defined by and in MIL-STD-462. Either approach is acceptable. Section 16.5 of these guidelines presents details of the latest revision of the cited standard. Tailoring of the cited standard for the host spacecraft and launch vehicle will be required upon pairing.

10.8 Radiation Environment

A Radiation Design Margin (RDM) for a given electronic part (with respect to a given radiation environment) is defined as the ratio of that part’s capability (with respect to that environment and its circuit application) to the environment level at the part’s location. The hardware should be designed so as to provide all parts an RDM of two (2) or greater.

The following guidelines are based upon a 98 degree inclination, 705 km altitude circular orbit.

10.8.1 Total Ionizing Dose

The instrument should operate within specification during and after exposure to the Total Ionizing Dose (TID) radiation environment of Table 10-4 over a two year mission life. The TID of Table 10-4 contains no margin or uncertainty factors.

Table 10-4: Total Ionizing Dose Radiation Environment

Shield Thickness mils	Electron rads-si	Bremsstrahlung rads-si	Trapped Proton rads-si	Solar Flare Proton 4 Events rads-si	Total rads-si
1	1.29E+06	6.76E+02	1.59E+04	1.40E+03	1.31E+06
3	7.56E+05	4.92E+02	8.72E+03	1.42E+03	7.67E+05
4	4.88E+05	3.83E+02	6.12E+03	1.43E+03	4.96E+05
6	3.38E+05	3.14E+02	4.60E+03	1.44E+03	3.45E+05
7	2.46E+05	2.65E+02	3.64E+03	1.46E+03	2.52E+05
9	1.87E+05	2.28E+02	3.07E+03	1.46E+03	1.92E+05
10	1.46E+05	1.99E+02	2.65E+03	1.48E+03	1.50E+05
12	1.16E+05	1.75E+02	2.35E+03	1.48E+03	1.20E+05
13	9.40E+04	1.56E+02	2.14E+03	1.49E+03	9.78E+04
15	7.64E+04	1.41E+02	1.96E+03	1.50E+03	8.00E+04
29	2.11E+04	7.40E+01	1.17E+03	1.52E+03	2.39E+04
44	1.08E+04	5.28E+01	9.04E+02	1.32E+03	1.31E+04
58	6.72E+03	4.16E+01	7.80E+02	1.15E+03	8.69E+03
73	4.52E+03	3.44E+01	6.92E+02	1.02E+03	6.27E+03
87	3.13E+03	2.93E+01	6.28E+02	9.24E+02	4.71E+03
117	1.55E+03	2.28E+01	5.48E+02	7.72E+02	2.89E+03
146	8.00E+02	1.87E+01	5.00E+02	6.60E+02	1.98E+03
182	3.45E+02	1.54E+01	4.60E+02	5.60E+02	1.38E+03
219	1.40E+02	1.32E+01	4.24E+02	4.76E+02	1.05E+03
255	5.36E+01	1.16E+01	4.00E+02	4.20E+02	8.85E+02
292	2.05E+01	1.04E+01	3.79E+02	3.70E+02	7.79E+02
365	2.03E+00	8.80E+00	3.43E+02	2.80E+02	6.33E+02
437	0.00E+00	7.68E+00	3.15E+02	2.34E+02	5.57E+02
510	0.00E+00	6.88E+00	2.90E+02	1.88E+02	4.86E+02
583	0.00E+00	6.24E+00	2.70E+02	1.55E+02	4.32E+02
656	0.00E+00	5.76E+00	2.54E+02	1.34E+02	3.93E+02
729	0.00E+00	5.32E+00	2.38E+02	1.13E+02	3.56E+02
875	0.00E+00	4.64E+00	2.12E+02	8.28E+01	3.00E+02
1167	0.00E+00	3.66E+00	1.75E+02	4.80E+01	2.27E+02
1458	0.00E+00	2.99E+00	1.48E+02	2.90E+01	1.80E+02

The TID versus shielding thickness curve for an aluminum spherical shell shield exposed to these fluences is defined in the Figure 10-3.

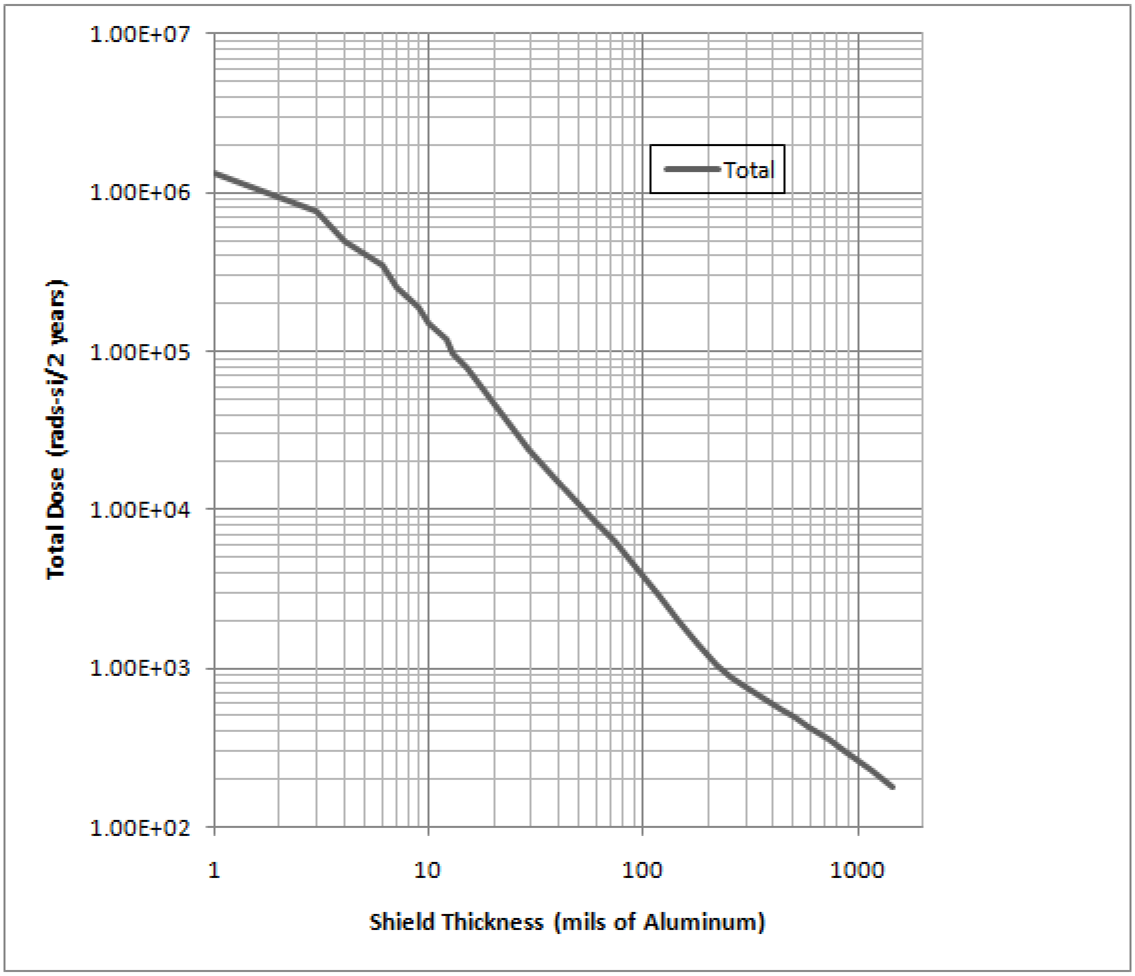


Figure 10-3: TID versus Shielding Thickness

10.8.2 Cosmic Ray and High Energy Proton Environment

10.8.2.1 *Displacement Damage Effects*

Prediction of proton-induced displacement damage (also known as bulk damage) to sensitive electronic parts should be based on the differential proton fluences defined in Table 10-5 and Table 10-6 and the integral proton fluence defined in Figure 10-4 and Figure 10-5. An RDM of 2 should be applied to this fluence.

Table 10-5: Total Differential Fluence of Trapped Protons and Galactic Cosmic Ray Protons

Surface Incident Energy (MeV)	Surface Incident Fluence (p/sqcm*MeV)	Emerging Energy (MeV)	Particles Emerging Behind Tantalum Spheres			Aluminum Spheres
			30 mils-TA (p/sqcm*MeV*5yrs)	150 mils-TA (p/sqcm*MeV*5yrs)	500 mils-TA (p/sqcm*MeV*5yrs)	100 mils-AL (p/sqcm*MeV*5yrs)
4.00E-02	1.41E+13	1.30E-01	-	2.07E+07	5.91E+06	1.12E+07
7.00E-02	1.10E+13	1.60E-01	8.20E+07	1.84E+07	5.27E+06	1.20E+07
1.00E-01	8.31E+12	2.00E-01	7.31E+07	1.64E+07	4.70E+06	1.29E+07
5.00E-01	6.11E+11	4.00E-01	5.17E+07	1.16E+07	3.33E+06	1.70E+07
1.00E+00	1.31E+10	5.00E-01	4.61E+07	1.04E+07	2.97E+06	1.93E+07
2.00E+00	3.36E+10	6.30E-01	4.23E+07	9.50E+06	2.72E+06	2.19E+07
3.00E+00	9.87E+09	7.90E-01	4.19E+07	9.41E+06	2.70E+06	2.49E+07
4.00E+00	8.10E+09	1.00E+00	4.35E+07	9.77E+06	2.80E+06	2.86E+07
5.00E+00	4.94E+09	2.00E+00	4.99E+07	1.28E+07	3.66E+06	4.53E+07
6.00E+00	3.12E+09	3.98E+00	7.39E+07	1.89E+07	5.41E+06	7.45E+07
8.00E+00	1.48E+09	5.01E+00	8.56E+07	2.19E+07	6.27E+06	8.35E+07
1.00E+01	1.06E+09	6.31E+00	8.56E+07	2.55E+07	7.30E+06	9.25E+07
2.00E+01	3.02E+08	7.94E+00	8.89E+07	2.99E+07	8.55E+06	1.10E+08
3.00E+01	1.66E+08	1.00E+01	1.01E+08	3.51E+07	1.00E+07	9.91E+07
4.00E+01	1.34E+08	2.00E+01	1.05E+08	4.58E+07	1.66E+07	1.08E+08
5.00E+01	1.21E+08	3.98E+01	1.03E+08	6.08E+07	2.12E+07	1.07E+08
6.00E+01	1.16E+08	5.01E+01	1.03E+08	5.99E+07	2.37E+07	1.04E+08
8.00E+01	8.79E+07	6.31E+01	8.67E+07	5.97E+07	2.54E+07	8.96E+07
1.00E+02	7.25E+07	7.94E+01	7.18E+07	5.11E+07	2.37E+07	7.35E+07
2.00E+02	2.44E+07	1.00E+02	5.66E+07	4.10E+07	2.33E+07	5.78E+07
3.00E+02	8.82E+06	2.00E+02	1.91E+07	1.54E+07	1.05E+07	1.92E+07
4.00E+02	3.39E+06	3.98E+02	2.82E+06	2.07E+06	1.49E+06	2.64E+06
5.00E+02	1.40E+06	5.01E+02	8.43E+05	3.94E+05	2.65E+05	6.37E+05
6.00E+02	1.58E+05	6.31E+02	1.51E+05	1.47E+05	1.43E+05	1.50E+05
8.00E+02	1.37E+05	7.94E+02	1.34E+05	1.28E+05	1.23E+05	1.31E+05
1.00E+03	1.14E+05	1.00E+03	1.10E+05	1.03E+05	9.76E+04	1.07E+05
2.00E+03	5.15E+04	2.00E+03	4.97E+04	4.59E+04	4.27E+04	4.72E+04
3.00E+03	2.81E+04	3.98E+03	1.68E+04	1.52E+04	1.40E+04	1.59E+04
4.00E+03	1.77E+04	5.01E+03	1.18E+04	1.09E+04	1.01E+04	1.12E+04
5.00E+03	1.23E+04	6.31E+03	8.03E+03	7.40E+03	6.86E+03	7.62E+03
8.00E+03	5.84E+03	7.94E+03	5.60E+03	5.13E+03	4.58E+03	5.33E+03
1.00E+04	4.09E+03	1.00E+4	1.07E-11	-	-	-

Table 10-6: Total Differential Fluence of Trapped Protons, Galactic Cosmic Ray Protons, and Solar Flare Protons

Surface Incident Energy (MeV)	Surface Incident Fluence (p/sqcm*MeV)	Emerging Energy (MeV)	Particles Emerging Behind Tantalum Spheres 30 mils-TA (p/sqcm*MeV *5yrs)	150 mils-TA (p/sqcm*MeV *5yrs)	500 mils-TA (p/sqcm*MeV *5yrs)	Aluminum Spheres 100 mils-AL (p/sqcm*MeV*5yrs)
4.00E-02	1.33E+13	1.30E-01	-	2.83E+07	5.11E+06	2.03E+07
7.00E-02	1.03E+13	1.60E-01	1.49E+08	2.52E+07	4.56E+06	2.17E+07
1.00E-01	7.64E+12	2.00E-01	1.33E+08	2.25E+07	4.06E+06	2.33E+07
5.00E-01	5.05E+11	4.00E-01	9.39E+07	1.59E+07	2.87E+06	3.09E+07
1.00E+00	8.22E+10	5.00E-01	8.37E+07	1.42E+07	2.56E+06	3.50E+07
2.00E+00	1.15E+10	6.30E-01	7.68E+07	1.30E+07	2.35E+06	3.98E+07
3.00E+00	5.64E+09	7.90E-01	7.61E+07	1.29E+07	2.33E+06	4.51E+07
4.00E+00	3.80E+09	1.00E+00	7.90E+07	1.34E+07	2.42E+06	5.20E+07
5.00E+00	2.54E+09	2.00E+00	9.44E+07	1.75E+07	3.16E+06	8.22E+07
6.00E+00	1.64E+09	3.98E+00	1.40E+08	2.59E+07	4.68E+06	1.35E+08
8.00E+00	1.28E+09	5.01E+00	1.62E+08	3.00E+07	5.42E+06	1.54E+08
1.00E+01	1.05E+09	6.31E+00	1.65E+08	3.49E+07	6.31E+06	1.75E+08
2.00E+01	5.19E+08	7.94E+00	1.74E+08	4.09E+07	7.39E+06	2.08E+08
3.00E+01	3.47E+08	1.00E+01	2.00E+08	4.81E+07	8.68E+06	1.94E+08
4.00E+01	2.62E+08	2.00E+01	2.12E+08	5.98E+07	1.43E+07	2.20E+08
5.00E+01	2.03E+08	3.98E+01	1.65E+08	6.99E+07	1.78E+07	1.76E+08
6.00E+01	1.61E+08	5.01E+01	1.41E+08	6.45E+07	1.96E+07	1.44E+08
8.00E+01	1.00E+08	6.31E+01	1.01E+08	5.80E+07	2.05E+07	1.07E+08
1.00E+02	6.80E+07	7.94E+01	7.15E+07	4.57E+07	1.84E+07	7.37E+07
2.00E+02	1.72E+07	1.00E+02	5.04E+07	3.43E+07	1.75E+07	5.16E+07
3.00E+02	5.97E+06	2.00E+02	1.34E+07	1.07E+07	7.19E+06	1.34E+07
4.00E+02	2.20E+06	3.98E+02	1.80E+06	1.30E+06	8.92E+05	1.68E+06
5.00E+02	8.51E+05	5.01E+02	4.30E+05	1.50E+05	8.60E+04	2.85E+05
6.00E+02	3.65E+04	6.31E+02	3.73E+04	3.77E+04	3.78E+04	3.76E+04
8.00E+02	3.83E+04	7.94E+02	3.82E+04	3.79E+04	3.76E+04	3.81E+04
1.00E+03	3.70E+04	1.00E+03	3.63E+04	3.52E+04	3.44E+04	3.59E+04
2.00E+03	2.57E+04	2.00E+03	2.52E+04	2.39E+04	2.28E+04	2.43E+04
3.00E+03	1.72E+04	3.98E+03	1.17E+04	1.08E+04	1.02E+04	1.12E+04
4.00E+03	1.22E+04	5.01E+03	8.85E+03	8.28E+03	7.77E+03	8.46E+03
5.00E+03	9.16E+03	6.31E+03	6.45E+03	6.02E+03	5.65E+03	6.17E+03
8.00E+03	4.93E+03	7.94E+03	4.76E+03	4.41E+03	3.98E+03	4.56E+03
1.00E+04	3.61E+03	1.00E+04	1.26E-11	-	-	-

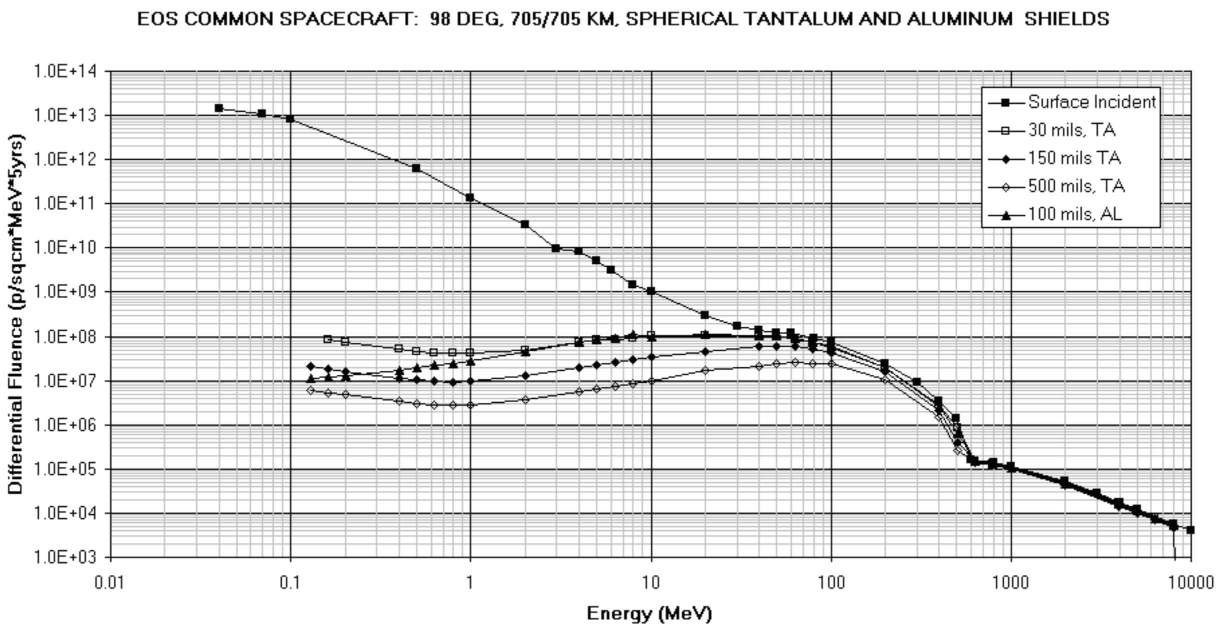


Figure 10-4: Total Differential Fluence of Trapped Protons and Galactic Cosmic Ray Protons

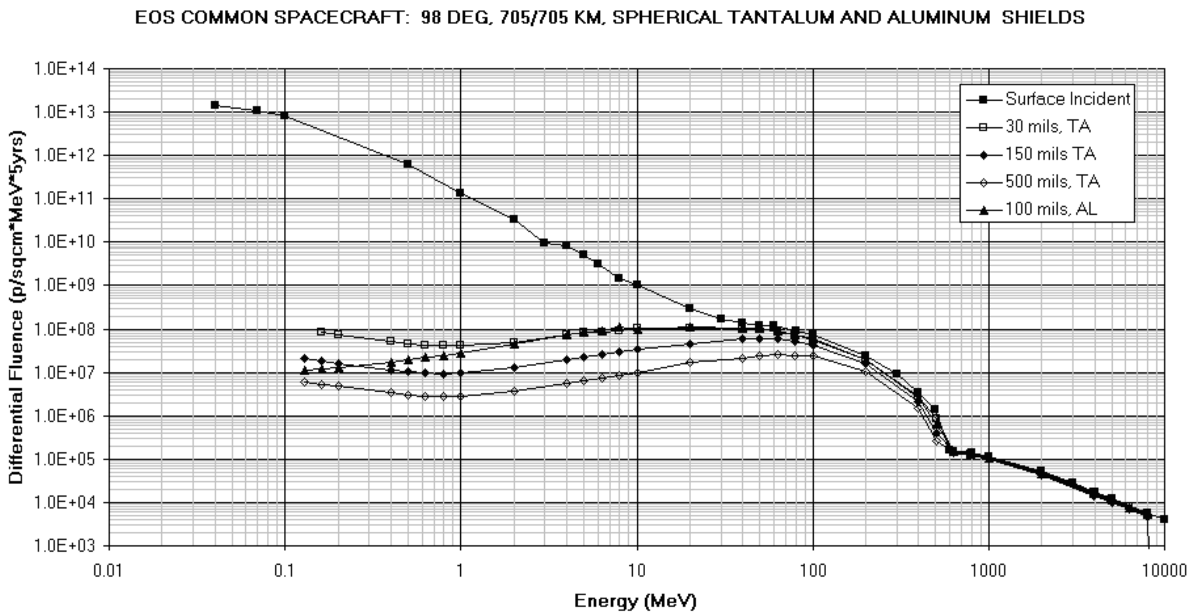


Figure 10-5: Total Differential Fluence of Trapped Protons, Galactic Cosmic Ray Protons, and Solar Flare Protons

10.8.2.2 *Single Event Effects*

The instrument should operate when exposed to Galactic Cosmic Ray and High Energy Protons.

Predictions of single events (*i.e.* single event latch-up, single event upset, and single event burn-out) induced by galactic cosmic ray ions and high energy protons should be performed separately and the results combined.

The integral Galactic Cosmic Ray Linear Energy Transfer (LET) spectrum provided in Figure 10-6 and Table 10-7 should be used to predict ion-induced single events. The difference proton fluence in Figure 10-4 and Table 10-5, which consists of trapped protons and galactic cosmic ray protons, should be used to predict proton-induced single events in the absence of solar flares. The difference proton fluence in Figure 10-5 and Table 10-6, which consists of trapped protons, galactic cosmic ray protons, and solar flare protons, should be used to predict proton-induced single events with solar flares.

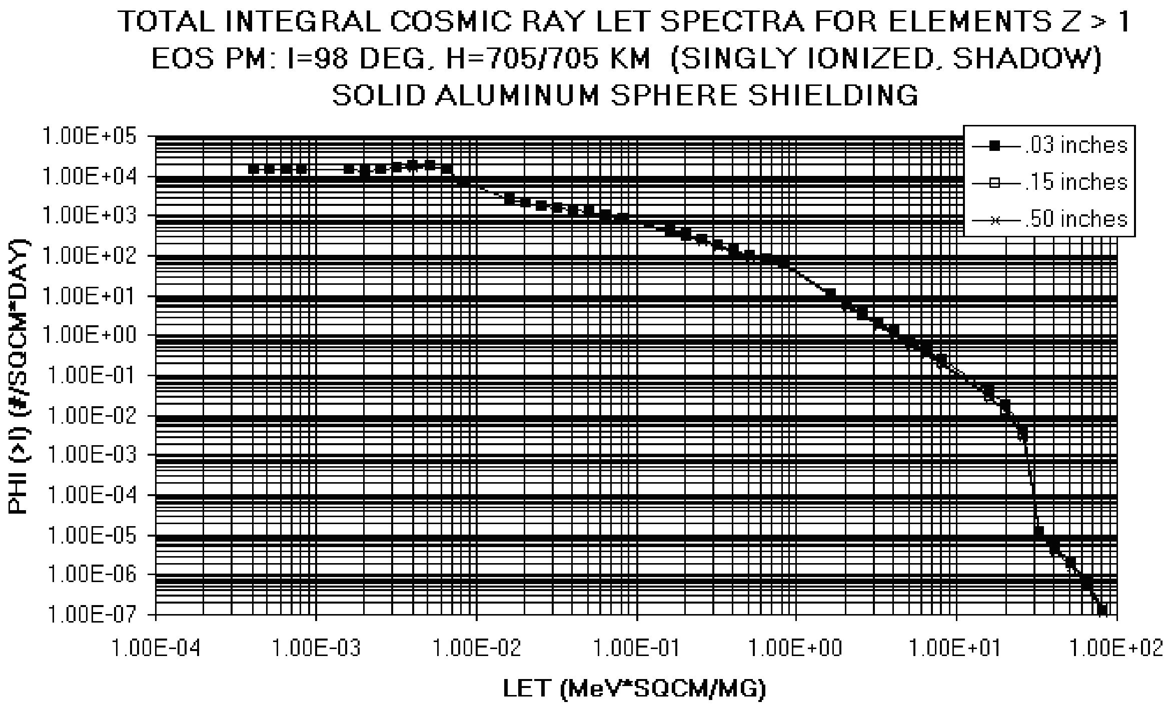


Figure 10-6: Total Integral Galactic Cosmic Ray LET Spectrum

Table 10-7: Total Integral Galactic Cosmic Ray LET Spectrum

INTEGRAL COSMIC RAY LET SPECTRA FOR ELEMENTS Z > 1
I=98 DEG, H=705/705 KM (SINGLY IONIZED WITH SHADOW)
SOLID ALUMINUM SPHERE SHIELDING

LET				LET SPECTRA		
MEV/CM	MEV*SQCM/MG	PC/CM	PC*SQCM/MG	0.03 inches #/SQCM*DAY	0.15 inches #/SQCM*DAY	0.50 inches #/SQCM*DAY
1.00E+00	4.10E-04	4.55E-02	1.86E-05	1.39E+04	1.38E+04	1.38E+04
1.26E+00	5.16E-04	5.73E-02	2.35E-05	1.39E+04	1.38E+04	1.38E+04
1.58E+00	6.48E-04	7.18E-02	2.94E-05	1.39E+04	1.38E+04	1.38E+04
2.00E+00	8.20E-04	9.09E-02	3.73E-05	1.39E+04	1.38E+04	1.38E+04
3.98E+00	1.63E-03	1.81E-01	7.41E-05	1.44E+04	1.44E+04	1.44E+04
5.01E+00	2.05E-03	2.28E-01	9.33E-05	1.36E+04	1.36E+04	1.35E+04
6.31E+00	2.59E-03	2.87E-01	1.18E-04	1.47E+04	1.46E+04	1.45E+04
7.94E+00	3.25E-03	3.61E-01	1.48E-04	1.60E+04	1.60E+04	1.58E+04
1.00E+01	4.10E-03	4.55E-01	1.86E-04	1.70E+04	1.70E+04	1.67E+04
1.26E+01	5.16E-03	5.73E-01	2.35E-04	1.75E+04	1.75E+04	1.72E+04
1.58E+01	6.48E-03	7.18E-01	2.94E-04	1.43E+04	1.42E+04	1.40E+04
2.00E+01	8.20E-03	9.09E-01	3.73E-04	7.49E+03	7.38E+03	7.23E+03
3.98E+01	1.63E-02	1.81E+00	7.41E-04	2.69E+03	2.64E+03	2.55E+03
5.01E+01	2.05E-02	2.28E+00	9.33E-04	2.14E+03	2.10E+03	2.02E+03
6.31E+01	2.59E-02	2.87E+00	1.18E-03	1.81E+03	1.78E+03	1.69E+03
7.94E+01	3.25E-02	3.61E+00	1.48E-03	1.61E+03	1.57E+03	1.51E+03
1.00E+02	4.10E-02	4.55E+00	1.86E-03	1.44E+03	1.39E+03	1.34E+03
1.26E+02	5.16E-02	5.73E+00	2.35E-03	1.30E+03	1.28E+03	1.23E+03
1.58E+02	6.48E-02	7.18E+00	2.94E-03	1.05E+03	1.03E+03	9.82E+02
2.00E+02	8.20E-02	9.09E+00	3.73E-03	8.85E+02	8.63E+02	8.32E+02
3.98E+02	1.63E-01	1.81E+01	7.41E-03	4.06E+02	3.97E+02	3.78E+02
5.01E+02	2.05E-01	2.28E+01	9.33E-03	3.27E+02	3.21E+02	3.05E+02
6.31E+02	2.59E-01	2.87E+01	1.18E-02	2.47E+02	2.44E+02	2.31E+02
7.94E+02	3.25E-01	3.61E+01	1.48E-02	1.81E+02	1.78E+02	1.68E+02
1.00E+03	4.10E-01	4.55E+01	1.86E-02	1.32E+02	1.30E+02	1.22E+02
1.26E+03	5.16E-01	5.73E+01	2.35E-02	1.02E+02	1.00E+02	9.42E+01
1.58E+03	6.48E-01	7.18E+01	2.94E-02	8.14E+01	8.01E+01	7.55E+01
2.00E+03	8.20E-01	9.09E+01	3.73E-02	6.51E+01	6.41E+01	6.04E+01
3.98E+03	1.63E+00	1.81E+02	7.41E-02	1.11E+01	1.06E+01	9.50E+00
5.01E+03	2.05E+00	2.28E+02	9.33E-02	6.09E+00	5.84E+00	5.26E+00
6.31E+03	2.59E+00	2.87E+02	1.18E-01	3.49E+00	3.31E+00	3.02E+00
7.94E+03	3.25E+00	3.61E+02	1.48E-01	2.07E+00	1.94E+00	1.77E+00
1.00E+04	4.10E+00	4.55E+02	1.86E-01	1.24E+00	1.12E+00	1.03E+00
1.26E+04	5.16E+00	5.73E+02	2.35E-01	7.04E-01	6.38E-01	5.88E-01
1.58E+04	6.48E+00	7.18E+02	2.94E-01	4.12E-01	3.62E-01	3.39E-01
2.00E+04	8.20E+00	9.09E+02	3.73E-01	2.44E-01	2.07E-01	1.93E-01
3.98E+04	1.63E+01	1.81E+03	7.41E-01	3.84E-02	2.90E-02	2.67E-02
5.01E+04	2.05E+01	2.28E+03	9.33E-01	1.73E-02	1.27E-02	1.17E-02
6.31E+04	2.59E+01	2.87E+03	1.18E+00	4.48E-03	3.12E-03	2.83E-03
7.94E+04	3.25E+01	3.61E+03	1.48E+00	1.11E-05	1.22E-05	9.94E-06
1.00E+05	4.10E+01	4.55E+03	1.86E+00	4.44E-06	4.99E-06	3.98E-06
1.26E+05	5.16E+01	5.73E+03	2.35E+00	1.81E-06	2.06E-06	1.62E-06
1.58E+05	6.48E+01	7.18E+03	2.94E+00	5.51E-07	6.46E-07	4.91E-07
2.00E+05	8.20E+01	9.09E+03	3.73E+00	1.28E-07	1.49E-07	1.12E-07
1.00E+06	4.10E+02	4.55E+04	1.86E+01	0.00E+00	0.00E+00	0.00E+00

10.8.2.3 *Solar Flare Events*

The instrument should survive without permanent degradation or damage and meet all performance requirements when exposed to trapped proton (E $\geq$ 5 MeV) flux of 6.4×10^3 particles/cm^2/sec, trapped electron (E $\geq$ 0.5 MeV) flux of 3.2×10^5 particles/cm^2/sec, and solar flare events specified in Table 10-8. If necessary, external intervention may restore functionality after the solar flare event. The total event integral fluence is accumulated within a time interval of a few hours to two days.

Table 10-8: Solar Flare Proton Peak Fluxes and Associated Total Event Integral Fluences

Energy (MeV)	Flux (Particles/cm^{2}sec)	Total Event Integral Fluence (Particles/cm^2)
>10	1.1×10^6	3.4×10^9
>30	2.6×10^5	1.8×10^9
>60	8.0×10^4	6.4×10^8
>100	-	1.5×10^8

10.9 **Electrostatic Discharge Environment**

10.9.1 External Potential Difference

The voltage between any instrument surface area greater than 0.5 cm^2 and the instrument structure should be less than or equal to 200 V, when exposed to a 5 nA/cm^2 flux level of electrons with 20 keV energy.

10.9.2 Electrostatic Discharge Susceptibility

The instrument should not be susceptible to a 6 mJ arc discharge (13 kV from 70 pF capacitor or equivalent) applied at a distance of 25 cm from the exterior surface of the instrument.

10.9.3 Internal Charging

Metallic elements, including wires, unused conductors of cable, connectors, circuit board traces, and spot shields greater than 3 cm^2 in area or 25 cm in length should have a conductive path to ground of less than 10^8 Ω when measured in air and less than 10^{12} Ω when measured in vacuum. Flight hardware that is shielded from the external radiation environment by greater than 0.32 cm (125 mils) of aluminum (or equivalent) need not be grounded.

10.10 **Solid Particles Environment**

10.10.1 Micrometeoroids

Table 10-9 defines the design level fluence (nominal levels, no margin, and omnidirectional fluence) for the Instrument mission micrometeoroid environment (cumulative).

Table 10-9: Solid Particle Fluence

Particle mass, M (g)	Omnidirectional Fluence (particles/m^2 of mass > M) Micrometeoroids	Artificial Space Debris
1.0×10^{-12}	1.0×10^{4}	2.3×10^{6}
1.0×10^{-10}	1.9×10^{3}	5.0×10^{4}
1.0×10^{-8}	3.6×10^{2}	1.1×10^{3}
1.0×10^{-6}	1.4×10^{1}	2.3×10^{1}
1.0×10^{-5}	1.4	3.4
1.0×10^{-4}	9.9×10^{-2}	5.0×10^{-1}
1.0×10^{-3}	5.7×10^{-3}	7.3×10^{-2}
1.0×10^{-2}	2.9×10^{-4}	1.1×10^{-2}
1.0×10^{-1}	1.4×10^{-5}	1.6×10^{-3}
1.0	6.7×10^{-8}	2.7×10^{-4}
1.0×10^{1}	-	6.9×10^{-5}
1.0×10^{2}	-	3.1×10^{-5}
Average Particle Density	2.5 g/cm^3	2.5 to 3.0 g/cm^3
Mean Impact Velocity	19 km/s	7 to 10 km/s

10.10.2 Artificial Space Debris

Table 10-9 defines the design level fluence (nominal levels, no margin, and omnidirectional fluence) for the Instrument mission artificial space debris environment (cumulative).

10.11 Atomic Oxygen Environment

The design level cumulative fluence for the Instrument mission atomic oxygen (AO) environment is 3.3×10^{20} AO/cm^2 for a 705 km circular polar orbit over two years.

11.0 SOFTWARE AND ELECTRONIC GROUND SUPPORT EQUIPMENT GUIDELINES

11.1 Introduction

The CII WG recommends that all instrument flight software and ground software be developed according to NASA Class C software development requirements and guidelines. More specific software development and implementation details are outside the scope of the CII Guidelines. High-level instrument flight software behavior flows from the CII Instrument Modes and Data Interface guidelines.

11.2 Instrument Flight Software Guidelines

11.2.1 Flight Software Development

CII flight software should be developed according to all NASA Class C software development requirements and guidelines.

Rationale: Appendix E of the *NASA Software Engineering Requirements* (NPR 7150.2A) classifies CII software as Class C. NASA Class C software is any flight or ground software which contributes to mission objectives, but whose correct functioning is not essential to the accomplishment of primary mission objectives. In the case of CII, primary mission objectives are exclusively those of the host spacecraft providing the HPO.

11.3 Instrument Ground Support Equipment Software Guidelines

11.3.1 Ground Software Development

CII ground support software should be developed according to all NASA Class C software development requirements and guidelines.

Rationale: Appendix E of the NPR 7150.2A classifies CII software as Class C. NASA Class C software is any flight or ground software which contributes to mission objectives, but whose correct functioning is not essential to the accomplishment of primary mission objectives. In the case of CII, primary mission objectives are exclusively those of the host spacecraft providing the HPO.

11.4 Instrument GSE to Spacecraft I&T GSE Interface

12.0 CONTAMINATION

12.1 Introduction

12.1.1 Goals

The Contamination Control Guidelines are comprehensive for hosted payload instruments. Excessive contamination, whether molecular films or particulate matters, can degrade the performance of the system by compromising the function of contamination-sensitive components such as the optics, focal planes, thermal control surfaces, solar cells, *etc*.

12.1.2 Assumptions

Guidelines apply to the instrument through all phases of development, design, integration, and test, and through the flight mission.

1) Guidelines contain reporting criteria.
2) Documents assumptions on contamination control of the instrument during spacecraft integration and launch operations.

12.1.3 Drivers

The sources of contamination and the migration mechanisms vary across the hardware phases of the project. Manufacturing, integration, testing, and mission operations each present unique contamination environments and constraints. The allocation of a contamination threshold level for each hardware phase should reflect the most cost- and time-efficient way of mitigating the risk of contamination-induced degradation. Materials selection, cleaning procedures, controlled assembly environments, storage and handling methods, venting, purges, design configuration, contamination transport analysis, hardware bake-outs, on-orbit decontamination cycles, *etc*. are some of the methods used to mitigate the risk of contamination.

12.2 Contamination Control Guidelines

12.2.1 Documentation Guidelines

The Instrument Provider should submit the cleanliness guidelines for all sensitive instrument surfaces that are exposed during Spacecraft I&T and launch site processing and document them in the MICD.

12.2.2 Verification of Cleanliness

Prior to integration with the Spacecraft, the Instrument Provider should verify the cleanliness of instrument exterior surfaces by test.

12.3 Instrument Sources of Contamination

The Instrument Provider should identify all sources of contamination that can be emitted from the instrument and should document these in the MICD.

12.4 Instrument Venting

12.4.1 Instrument Venting Documentation

The MICD should define number, location, size, vent path, and operation time of vents.

12.4.2 Location of Vent Path

The Spacecraft Contractor should place the instrument such that the contamination products from the vents of hosted payload instrument do not directly impinge on the primary instrument's contamination-sensitive surface nor directly enter another instrument's aperture.

12.4.3 Sealed Hardware

Electro-explosive devices (EEDs), hot-wax switches, or other devices should be sealed devices which do not allow escape of the actuating materials.

12.5 Protective Covers

12.5.1 Covering of Sensitive Instrument Surfaces

Sensitive instrument surfaces should be covered during Spacecraft I&T, shipment, launch site processing, and launch.

12.5.2 Responsibility for Covers

The Instrument Provider should provide Instrument protective covers and specify procedures for their use. The Instrument Provider should specify in the MICD if and when protective covers (such as bags, draping materials, or hard covers) are required to keep the instrument clean during I&T.

12.6 Instrument Purge Requirements

The MICD should document instrument purge requirements, including type of purge gas, flow rate, gas purity specifications, filter pore size, type of desiccant (if any), and whether interruptions in the purge are tolerable.

12.7 Instrument Inspection and Cleaning During I&T

The MICD should define any required inspections or cleaning of the instrument during I&T. The Instrument Provider is responsible for cleaning the instrument.

12.8 Contamination Analysis Guidelines

The instrument contamination analysis should be based on the following:

1) The cleanliness of the external surfaces and surface molecular levels should adhere to MIL-STD-1246C.
2) Worst case on-orbit redistribution of particles for spacecraft and instrument external surfaces should not exceed Level 650 for velocity vector surfaces and Level 600 for all other surfaces.
3) The analysis of long-term degradation should consider orbital debris.

12.9 Spacecraft Contractor Supplied Analysis Inputs

The Spacecraft Contractor should provide the Instrument Provider with plume flow field analyses for all thrusters, including any launch vehicle stages which fire after the payload fairing is jettisoned. The flow field analysis results should include:

1) Identity and quantity of each chemical species emitted,
2) Density as a function of spatial position, including the "backflow region" at angles greater than 90 degrees from the plume centerline,
3) Velocity or flux as a function of spatial position, including the backflow region,
4) An equation or group of equations describing the plume.

12.10 Particulate and Molecular Cleanliness

12.10.1 Cleanroom Specifications

The instruments should be integrated with the Spacecraft in a Class 10,000 (ISO Class 5.5) cleanroom environment and maintained in that environment as much as possible during the integration and test flow.

12.10.2 Cleanroom Performance Verification

Air cleanliness, NVR, and particle fallout rates should be measured at regular intervals, satisfying, as a minimum, the requirements in ANSI/IEST/ISO 14644-1.

12.10.3 Transitions between Cleanrooms

Hardware manufactured in areas that do not meet Class 100,000 clean room (ANSI/IEST/ISO 14644-1) standards should undergo proper cleaning, including vacuum baking.

12.11 GSE Cleanliness Guidelines

12.11.1 GSE in Cleanroom Area

Any GSE that accompanies the instrument into a cleanroom area should be cleaned and cleanroom-compatible.

12.11.2 GSE in Vacuum Chamber

Any GSE in the vacuum chamber during thermal vacuum testing should be cleaned and vacuum-compatible.

12.12 Wiring and MLI Cleanliness Guidelines

All wiring harnesses and thermal blankets should be baked out and screened for outgassing.

13.0 COORDINATE REFERENCE FRAMES

This section defines guidelines for establishing the coordinate reference frames for a spacecraft host and instrument payload in Low Earth Orbit (LEO).

13.1 Spacecraft Host Coordinate Reference Frame

The spacecraft host will define the location and orientation of this reference frame.

13.1.1 Spacecraft Host Coordinate Frame Location

The subject reference frame is typically defined with the origin located at the center of the separation plane between the spacecraft host and the launch vehicle. The subject reference frame "typical" location is depicted in Figure 13-1.

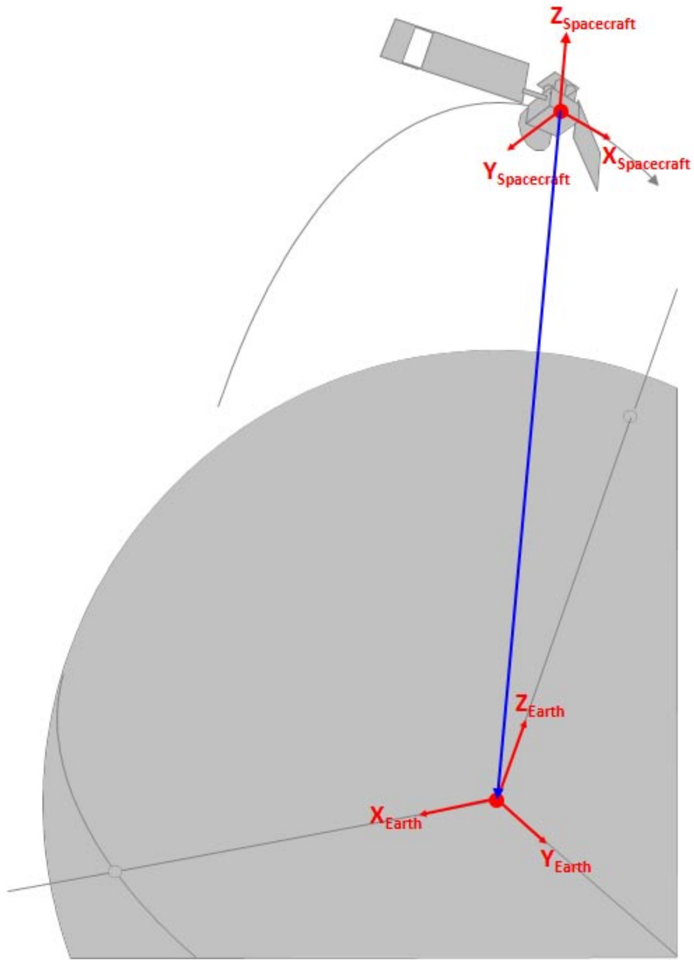


Figure 13-1: Spacecraft Coordinate Reference Frame Location & Orientation

13.1.2 Spacecraft Host Coordinate Frame Orientation

The orientation of this coordinate frame is typically defined with the $\vec{Z}_{Spacecraft}$ axis extending from the origin along the axis pointed in the zenith (anti-nadir) direction with the spacecraft in the deployed orbital configuration, assuming that the spacecraft is inertially stabilized. The $\vec{Y}_{Spacecraft}$ axis of the spacecraft is orthogonal to the orbital plane, and the $\vec{X}_{Spacecraft}$ axis is the resultant of the normalized product $|\vec{Y}_{Spacecraft}\ X\ \vec{Z}_{Spacecraft}|$ oriented in the direction of the spacecraft velocity vector. The subject reference frame "typical" orientation is depicted in Figure 13-1 in the orbital configuration.

13.2 Instrument Payload Coordinate Reference Frame

The instrument payload defines the location and orientation of this reference frame relative to the instrument. The Spacecraft defines the location and orientation of the instrument payload coordinate reference frame relative to the Spacecraft Host Coordinate Reference Frame.

13.1.2 Instrument Payload Coordinate Frame Location

The subject reference frame is defined with the origin located at the center of the interface plane between the instrument payload and the spacecraft host. The subject reference frame "typical" location is depicted in Figure 13-2.

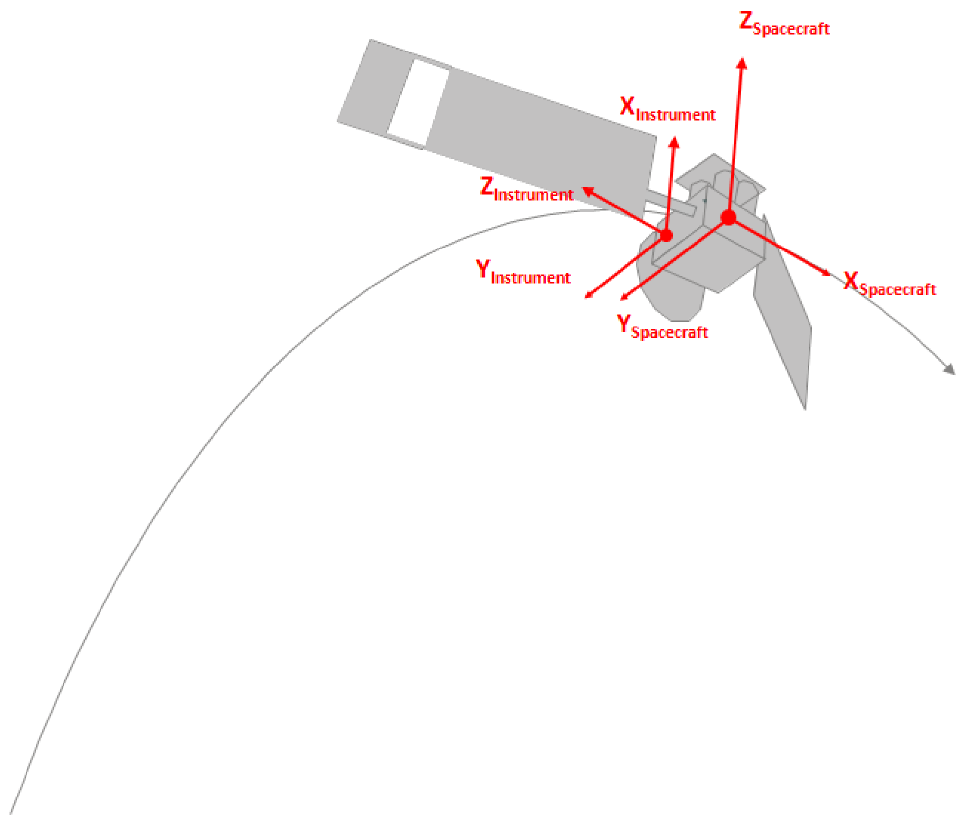


Figure 13-2: Instrument Coordinate Reference Frame Location & Orientation

13.2.2 Instrument Payload Coordinate Frame Orientation

The orientation of this coordinate frame is defined with the $\vec{Z}_{Instrument}$ axis extending from the origin of the Instrument Payload Coordinate Frame along the axis orthogonal to the interface plane between the instrument and the spacecraft host in the direction away from the spacecraft host. The remaining Instrument Payload Coordinate Frame axes, the $\vec{X}_{Instrument}$ axis and the $\vec{Y}_{Instrument}$ axis, are located in the interface plane between the instrument payload and the spacecraft host and are typically oriented so that at least one of the axes is aligned with one of the Spacecraft Host Coordinate Frame axes. Such aligned orientations are preferred in order to simplify coordinate system transformations. The subject reference frame "typical" orientation is depicted in Figure 13-2.

13.3 Nominal Orbit

The CII guidelines assume a nominal LEO mission orbit. A particular orbital altitude, eccentricity, inclination and other parameters are not defined. LEO is defined to be between 350 kilometers and 2000 kilometers orbital altitude. The altitude lower bound is considered to be the beginning of the Earth's atmosphere and the limit at which an orbit may continue to be propagated with reasonable drag makeup requirements. The upper bound is defined by the beginning of the inner Van Allen radiation belt.

14.0 MODEL GUIDELINES AND SUBMITTAL DETAILS

14.1 Finite Element Model Submittal

This section defines the finite element model and model submittal guidelines.

14.1.1 Model Representation

Components which have fixed base frequencies greater than 100 Hz can be represented as rigid masses.

14.1.2 Finite Element Model (FEM) Guidelines

1) The FEM should utilize the NASTRAN format.
2) The model will be used for dynamic analysis rather than stress analysis and therefore should be as small as possible. Models should be 'full" models with no symmetry assumptions made to reduce model size. "Super elements" should not be used.
3) Model should adequately represent all dynamic modes up to 100 Hz when rigidly supported at the interface.
4) The local coordinate system should be rectangular with the same orientation as the Spacecraft coordinate system. Coordinate cards should be provided to establish the local system. The origin of the local coordinate system should be documented in the ICD.
5) All constraints internal to instruments and output should be in the local coordinate system. There should be no constraints external to the model upon submittal.
6) The bulk data deck should not contain BAROR, GRDSET, PARAM K6ROT or PARAM AUTOSPC NASTRAN cards.
7) The specification of vector components for element coordinate system definition should be used in lieu of referenced grid (i.e., CBAR, CBEAM).
8) The Système international d’unités (SI) is the preferred system of units.
9) An instrument-unique numbering system for all NASTRAN model identification numbers should be used (for grids, coordinates systems, elements, property and material IDs, as well as constraint and loading IDs). This number should be of the form XXX001 through XXX999, where XXX is the instrument identification number provided by the Spacecraft Integrator. All instrument components which are physically separate units should be modeled separately with unique identification numbers within the unique XXX instrument number allocation. No ID number duplications should be allowed.

10) The NASTRAN model should be capable of representing different dynamic response characteristics (if any) resulting from planned configuration changes between launch and deployment.

14.1.3 Deliverable NASTRAN Model Data

The following data and information should be supplied in fulfillment of the guideline of a deliverable NASTRAN math model:

1) NASTRAN structural math model of the instrument which is able to:

   a) Define the loads at the instrument/support structure interface due to gravity loads and structural element temperature changes.

   b) Provide sufficient detail to accurately represent the dynamic behavior of the instrument up to 100 Hz. In general, no more degrees-of-freedom than necessary should be used. Models which employ static or dynamic reduction techniques should refer to the validation guidelines in Section 14.1.4.

2) The method of data transfer is negotiable with NASA.
3) Description of the model, any special modeling features used, and rationale for the modeling methodology.
4) List of all material properties used in the model.
5) The mass and stiffness calculations used in generating the input data for the model and a description of how the masses were distributed throughout the structure.
6) A description of the dynamic ASET degrees-of-freedom.
7) Detailed plots of the NASTRAN model clearly showing all grid points, element numbers, and element types. Hardcopy plots showing element connectivity should also be supplied. Identify clearly all Spacecraft attachment points with respective degrees-of-freedom.
8) Mechanical and functional description of all mechanisms in the instrument, whether they are modeled or not.
9) Mode plots of the first three significant modes.
10) The following are to be delivered with the test verified model: Design drawings of the instrument to manufacturing standards. Copy of the stress analysis (input and output) of all loading combinations performed on the detailed model, with cross references to the NASTRAN model and drawing numbers. All fracture critical parts must be clearly labeled as such.

14.1.4 Deliverable Model Validity Checks

The following computer runs and data checks should be made and delivered to NASA to confirm the mathematical validity of the model:

1) Static analysis for unit gravity loading in each of the three axis directions. Reaction forces given by this analysis should equal the weight of the structure as given by the grid-

point weight generator (using PARAM WTMASS card). No large displacements or forces should be generated.

2) Eigenvalue analysis for modes up to 100 Hz with the instrument constrained at the instrument/support structure interface. Only those degrees-of-freedom actually used to attach the instrument to the support should be constrained in this analysis.

3) Eigenvalue analysis for the modes up to 100 Hz of the instrument in the free-free (unconstrained) condition. This analysis should be made both with and without the SUPORT NASTRAN bulk-data card. The rigid body modes from the analysis without the SUPORT card should yield frequencies less than 0.01 Hz.

4) Static analysis with unit-enforced displacements in all six degrees-of-freedom at one grid point. This analysis should yield equivalent unit values of displacement for all grid points whose displacement coordinate system is defined as being parallel to the input coordinate system of the referenced point. In addition, no element forces greater than 0.05 kg or moments greater than 0.1 N-M should be observed. A grid-point force balance should reveal no significant forces on any point in the model.

5) Grid-point weight generator (using PARAM WTMASS card) should yield correct weight, center-of-mass location, and moments of inertia.

6) Use of case-control command "SPCFORCES=ALL" should reveal no constraint forces at points other than legitimate boundary condition constraint points.

7) "Epsilon Sub E" error check for each static subcase should be less than 10^{-11}.

8) Static analysis with a unit temperature increase from ambient. The model should contain a stress-free boundary condition, (*i.e.*, kinematic mount or single constrained grid point), with all the thermal expansion coefficients changed to simulate identical materials. The grid-point force balance should reveal no force greater than 0.005 kg. Care should be taken to ensure that rigid elements do not constrain thermal expansion.

9) A rigid-body or stiffness-equilibrium check should be performed. Terms should be less than 10^{-5}.

10) Modal results for reduced models should agree with full-up model as follows:

    a) Fundamental frequency should agree to within 3%.

    b) All remaining modes to 100 Hz should have frequencies to within 5%.

    c) Cross-orthogonality check with full-up model must have diagonal terms greater than .95 and off-diagonal terms less than .05.

14.1.5 NASTRAN Model Verification

If either the analysis or a frequency verification test shows any significant mode below 70 Hz, these modes should be verified through a modal survey.

The analytical frequency predictions should agree with dynamic test data to within 5 percent for the fundamental mode and to within 10 percent for all remaining significant modes to 70 Hz.

A cross orthogonality check between test and analytical mode shapes should have diagonal terms above 0.9 and off diagonal terms below 0.1 for all significant modes up to 50 Hz.

14.2 Thermal Math Model Submittal

14.2.1 Thermal Math Model

The Instrument Provider should supply the Spacecraft Provider with a reduced node geometric and thermal math model in compliance to the following:

14.2.1.1 *Model Format*

Model format should be in Thermal Desktop version 5.2 or later or NX Space Systems Thermal version 7.x or later.

14.2.1.2 *Units of Measure*

Model units should be SI.

14.2.1.3 *Radiating Surface Element Limit*

Radiating surface elements should be limited to less than 200.

14.2.1.4 *Thermal Node Limit*

Thermal nodes should be limited to less than 500.

14.2.1.5 *Model Verification*

The Geometric Math Model and Thermal Math Model should be documented with a benchmark case in which the Spacecraft Developer may use to verify the model run.

14.2.1.6 *Steady-State and Transient Analysis*

The model should be capable of steady-state and transient analysis.

14.2.2 Reduced Node Thermal Model Documentation

The Instrument Provider should supply the Spacecraft Developer with documentation describing the reduced node thermal model. The documentation should contain the following:

14.2.2.1 *Node(s) Location*

The node(s) location at which each temperature limit applies.

14.2.2.2 *Electrical Heat Dissipation*

A listing of electrical heat dissipation and the node(s) where applied.

14.2.2.3 *Active Thermal Control*

A listing of active thermal control, type of control (*e.g.*, proportional heater), and the node(s) where applied.

14.2.2.4 *Boundary Notes*

A listing and description of any boundary nodes used in the model.

14.2.2.5 *Environmental Heating*
A description of the environmental heating (Beta angle, heliocentric distance, planetary albedo, planetary emissive power, *etc*.).

14.2.2.6 *User Generated Logic*
A description of any user generated software logic.

14.3 **Mechanical Computer Aided Design (CAD) Model Submittal**

14.3.1 <u>Mechanical CAD Model</u>
14.3.1.1 *Model Format*
The Instrument Provider should provide Mechanical CAD models that have been created in a file format compatible with the spacecraft vendor, *i.e.* in the same program and version of the spacecraft vendor or in a neutral file format such as IGES or STEP.

14.3.1.2 *Unit of Measure*
Model units should be in SI.

14.4 **Mass Model**

14.4.1 <u>Instrument Mass Model</u>
The Instrument Provider should provide all physical mass models required for spacecraft mechanical testing.

15.0 ACRONYMS AND SCIENTIFIC UNITS

15.1 Acronyms

AC	Alternating Current
AFT	Allowable Flight Temperatures
ANSI	American National Standards Institute
AO	Atomic Oxygen
APID	Application Process IDentifer
ARC	Ames Research Center
ASD	Acceleration Spectral Density
AWG	American Wire Gauge
C&DH	Command and Data Handling
CAD	Computer Aided Design
CCSDS	Consultative Committee for Space Data Systems
CDR	Critical Design Review
CERES	Clouds and Earth's Radiant Energy System
CII	Common Instrument Interface
CRC	Cyclic Redundancy Check
CUC	CCSDS Unsegmented (Time) Code
DA	Double Amplitude
DC	Direct Current
ECSS	European Cooperation for Space Standardization
EED	Electro-explosive Device
EEP	Error End of Packet
EHA	Engineering, Health, and Accountability
EIA	Electronic Industries Association
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOL	End of Life
EOP	End Of Packet
EOS	Earth Observing System
EPS	Electrical Power Subsystem
ESA	European Space Agency
ESD	Earth Science Division
ESSP	Earth System Science Pathfinder
EVI	Earth Venture Instrument
ExP	Either EEP (Error End-of-Packet) or EOP (End-Of-Packet)
FDIR	Fault Detection, Isolation, and Recovery
FEM	Finite Element Model
FGS	Fine Guidance Sensor (JWST instrument)
FOV	Field of View
G	Gauss
GEO	Geostationary Earth Orbit
GEVS	General Environmental Verification Standard

GIRD	General Instrument Requirements Document
GIRD	General Interface Requirements Document
GOCE	Gravity field and steady-state Ocean Circulation Explorer
GOES-R	Geostationary Operational Environmental Satellite – R Series
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
H×W×L	Height × Width ×Length
HPO	Hosted Payload Opportunity
I&T	Integration and Test
IAC	Interface Alignment Cube
ICD	Interface Control Document
IDD	Instrument Description Document
JAXA	Japan Aerospace Exploration Agency
JPL	Jet Propulsion Laboratory
JWST	James Webb Space Telescope
KDP	Key Decision Point
LaRC	Langley Research Center
LEO	Low Earth Orbit
LEOP	Launch and Early Orbit Operation
LET	Linear Energy Transfer
LRO	Lunar Reconnaissance Orbiter
LV	Launch Vehicle
LVLH	Local Vertical/Local Horizontal
mA	Milliamp
MAC	Mass Acceleration Curve
MeV	Mega electron Volts
MHz	Megahertz
MICD	Mechanical Interface Control Document
MIL-STD	Military Standard
MIRI	Mid-Infrared Instrument (JWST instrument)
MLI	Multi-Layer Insulation
mm	Millimeter
MMO	Mercury Magnetospheric Orbiter (BepiColombo)
MPO	Mercury Planetary Orbiter (BepiColombo)
MSFC	Marshall Space Flight Center
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NDE	Non-Destructive Evaluation
NeXT	Non-thermal Energy eXploration Telescope
NICM	NASA Instrument Cost Model
NIRCam	Near-Infrared Camera (JWST instrument)
NIRSpec	Near-Infrared Spectrograph (JWST instrument)
NSI	NASA Standard Initiator
NTE	Not to Exceed

OAP	Orbital Average Power
PDR	Preliminary Design Review
PF	Payload Fairing
PM	EOS Afternoon (PM) Mission Project
PPL	Preferred Parts List
PPS	Pulse Per Second
RDM	Radiation Design Margin
RSDO	Rapid Spacecraft Development Office
SI	Système international d’unités
SMD	Science Mission Directorate
SpW	SpaceWire
SRS	Shock Response Spectrum
STP-SIV	Space Test Program Standard Interface Vehicle
TBD	To Be Determined
TBR	To Be Resolved
TID	Total Ionizing Dose
TIRS	Thermal Infrared Sensor
TSIS	Total and Spectral Solar Irradiance Sensor
UIID	Unique Instrument Interface Document
VDC	Volts direct current
W	Watts
WFC	Wide Field Camera

15.2 **Units of Measure**

Abbreviation	Unit
A	Ampere
B	Bel
bps	Bits per second
eV	Electron-Volt
F	Farad
g	Gram
Hz	Hertz
J	Joule
m	Meter
N	Newton
s	Second
T	Tesla
V	Volt
Ω	Ohm

15.3 Metric Prefixes

Prefix	Meaning
M	mega (10^6)
k	kilo (10^3)
d	deci (10^{-1})
c	centi (10^{-2})
m	milli (10^{-3})
µ	micro (10^{-6})
n	nano (10^{-9})
p	pico (10^{-12})

16.0 REFERENCE MATERIAL / BEST PRACTICES

16.1 Data Interface Reference Material / Best Practices

16.1.1 SpaceWire

16.1.2 SpaceWire Exclusivity

16.1.2.1 *Command and Time Message Inputs*

The Instrument should process and receive all commands and time messages exclusively using SpaceWire (ECSS-E-ST-50-12C).

16.1.2.2 *Telemetry and Science Data Outputs*

The Instrument should send all telemetry and science data exclusively using SpaceWire (ECSS-E-ST-50-12C).

Rationale: SpaceWire provides low-power, high data-rate, asynchronous serial communications.

SpaceWire is comprised of physical, link, and logical data layers. The physical layer specification defines detailed requirements for flight-qualified cabling and a micro-miniature D-type 9-pin connector. The link layer provides for handshake and low-level time synchronization. The data layer defines character codes, control codes, and a basic packet structure.

SpaceWire allows for the combination of many common data functions, often provided by disparate instrument-to-spacecraft data lines, into a single cable. For example, in this specification, traditional 1 PPS time signals are subsumed by SpaceWire time codes Section 16.1.4). Also, analog data read backs (e.g. thermistors) are replaced by digital housekeeping data sent from instrument to spacecraft as CII SpaceWire telemetry packets (see Section 16.1.13).

16.1.3 SpaceWire Clock and Data Rate

The SpaceWire link between the instrument and spacecraft should be clocked at 10 MHz in both directions. This provides a total data rate of 10 Mbps and, accounting for SpaceWire overhead, a useable data rate of approximately 8 Mbps.

Rationale: Based on a review of the NICM database, the majority of hosted payload class instruments do not exceed an expected data rate of 1.5 Mbps. While SpaceWire defines a minimum data signaling rate of 2 Mbps [ECSS-E-ST-50-12C, Section 6.6.1], the initial (i.e.

default at power-on) data signaling rate is 10 Mbps [ECSS-E-ST-50-12C, Section 6.6.5]. Reconciling these numbers, the CII default SpaceWire data rate is set to 10 Mbps. This comfortably exceeds expected hosted payload instrument capability and provides margin for possible outlier higher data rate instruments or those instruments that may need to exceed the expected maximum for short bursts. Further, adopting the initial SpaceWire 10 Mbps data rate eliminates the need for both instrument and spacecraft hardware and software to negotiate a lower data rate (e.g. 2 Mbps), which simplifies the interface.

Note: Data rates higher than 10 Mbps *may* be negotiated directly between a specific instrument and spacecraft provider once an HPO has been identified.

16.1.4 SpaceWire Time Codes

The Instrument should receive a low-level SpaceWire time code from the Spacecraft once per second.

Rationale: The SpaceWire time (control) code supplants the traditional 1 pulse-per-second (PPS) dedicated control line from spacecraft to instrument. This reduces the total number of wires between the instrument and spacecraft and simplifies the interface. Instruments are assumed to maintain their own internal clock, that coupled with Spacecraft Status Messages (see Section 16.1.10) further obviates the need for a 1 PPS signal. Low-level instrument electronics that still require a 1 PPS signal will use the SpaceWire time code.

Note: The 1 PPS time code signal is distinct from the high-level, periodic time message sent by the spacecraft to the instrument (see Sections 16.1.9.1 and 16.1.10).

16.1.5 SpaceWire Redundancy (optional)

16.1.5.1 *Multilink Configuration*

The instrument may employ two or four SpaceWire links for communication with the spacecraft in a simple parallel or cross-strapped configuration.

16.1.5.2 *Active Links*

Only one link should be considered active at any given time.

16.1.5.3 *Secondary Links*

Secondary link(s) should be used in the event the primary active link becomes inoperative or the instrument swaps to second string components.

Rationale: Hosted payload instruments are expected to be fall into NASA risk classifications C where reduced mission assurance standards are permitted. Furthermore, instrument providers will likely not be able to afford SpaceWire link-redundancy and all of its associated hardware and software complexity. Single-string instrument providers are strongly encouraged to minimize risk using other measures (e.g. design, analysis, parts selection).

16.1.6 <u>CII Messages, Packet Formats, and Protocol</u>

16.1.7 <u>CII Messages</u>

There are five types of CII messages. Each message type is assigned a function code and data path (either instrument to spacecraft or vice-versa). Table 16-1 summarizes the message types, which subsequent subsections detail.

Table 16-1: CII Message Types

Type Code	Message Type	Data Path
`0x02`	Spacecraft Status Message	Spacecraft to Instrument
`0x04`	Command	Spacecraft to Instrument
`0x08`	Command Acknowledgement	Instrument to Spacecraft
`0x10`	Instrument Telemetry	Instrument to Spacecraft
`0x20`	Instrument Science Data	Instrument to Spacecraft

16.1.7.1 *Unidirectional Message Flow*

Each CII message type should flow only in the single direction indicated in Table 16-1.

16.1.8 <u>CII Basic Packet Format</u>

The SpaceWire – CCSDS Packet Transfer Protocol Specification (ECSS-E-ST-50-53C) forms the basis of the CII data packet format. The SpaceWire – CCSDS specification builds upon the SpaceWire (ECSS-E-ST-50-12C), SpaceWire Protocol Identification (ECSS-E-ST-50-12C) specifications, and CCSDS Space Packet Protocol Blue Book (CCSDS 133.0-B-1).

16.1.8.1 *Packet Structure*

The structure of a complete CII SpaceWire packet should conform to that depicted in Figure 16-1.

<table>
<tr><th colspan="8">CII SpaceWire Packet</th></tr>
<tr><td rowspan="3">SpW Logical Address</td><td colspan="6">Cargo</td><td rowspan="3">ExP</td></tr>
<tr><td rowspan="2">Protocol ID (0x02)</td><td colspan="5">CCSDS</td></tr>
<tr><td>Reserved (0x00)</td><td>Type Code</td><td>Header</td><td>Data</td><td>CRC</td></tr>
<tr><td rowspan="2">1 byte</td><td>1 byte</td><td>1 byte</td><td>1 byte</td><td>15 bytes</td><td>0–236 bytes</td><td>1 byte</td><td rowspan="2">4 bits</td></tr>
<tr><td colspan="6">19–255 bytes</td></tr>
</table>

<u>Specification Legend</u>

- SpaceWire (ECSS-E-ST-50-12C)
- SpaceWire Protocol Identification (ECSS-E-ST-50-51C)
- SpaceWire – CCSDS Packet Transfer Protocol (ECSS-E-ST-50-53C)
- GOES-R Reliable Data Delivery Protocol (417-R-RPT-0050; uses CRC only)
- Packet specific data (e.g. time, position, commands, telemetry, science data)

Figure 16-1: CII SpaceWire Packet

The SpaceWire Specification defines a basic data packet format with one or more optional SpaceWire target path addresses and/or a target logical address. Following the SpaceWire path address are arbitrary packet data cargo and then an end-of-packet (EOP) marker. In the case of a SpaceWire link error, where a packet transmission ended prematurely, an error end-of-packet (EEP) marker replaces the EOP.

16.1.8.2 *Address Assignment*
The Spacecraft Provider may assign a path or logical SpaceWire address to the Instrument at the time of integration.

Rationale: Since path addresses are removed by SpaceWire hardware (e.g. when a packet passes through a SpaceWire router) and thus not encountered by SpaceWire applications (e.g. the instrument), their prescription is largely outside the scope of this document.

16.1.8.3 *Cargo Packet Size*
CII SpaceWire packet cargo should be no larger than 256 bytes in size.

Rationale: Northrop Grumman personnel with technical experience regarding SpaceWire's use on JWST strongly recommend a 256-byte size cap on SpaceWire packet data.

16.1.8.4 *Single Byte Logical Address*
A CII SpaceWire packet should start with a single byte logical address when it arrives at either the instrument or spacecraft.

16.1.8.5 *Default Logical Address*
If no specific logical address is assigned, the default logical address of 254 (`0xFE`) should be used.

Rationale: Required by the SpaceWire Protocol Identification specification [ECSS-E-ST-50-51C, 5.2.1 (a) and (b)].

16.1.8.6 *Protocol ID Byte*
The CII packet Protocol ID byte should be set to `0x02`.

Rationale: Required by the SpaceWire – CCSDS Packet Transfer Protocol specification [ECSS-E-ST-50-53C, 5.3.3].

16.1.8.7 *Reserved Byte*
The CII packet Reserved byte should be set to `0x00`.

Rationale: Required by the SpaceWire – CCSDS Packet Transfer Protocol specification [ECSS-E-ST-50-53C, 5.3.4].

The CII packet Type Code byte indicates the specific type of CII message that follows in the CCSDS packet data portions of the CII packet.

16.1.8.8 *Type Code*

The CII packet Type Code should be set to one of the values listed in Table 16-1.

A CRC byte located at the end of the CII packet CCSDS section allows packet recipients to check for packet corruption.

16.1.8.9 *CRC Computation Method*

The CRC computation method is defined in the GOES-R Reliable Data Delivery Protocol specification [417-R-RPT-0050].

16.1.8.10 *CRC Computation Leading and Trailing Bytes*

The CRC computation should include bytes starting with the CII packet Protocol ID byte and ending with the final CII packet Data byte.

Rationale: The CRC allows errors to be detected occurring in either the packet control data (e.g. Type Code, CCSDS packet length fields, etc.) or instrument telemetry and science data.

16.1.9 <u>CCSDS Packet Data Structure</u>

The structure of a complete CII CCSDS packet data header should conform to that depicted in Figure 16-2 through Figure 16-4.

Rationale: CCSDS headers are the established industry standard for space packet communication. The CCSDS primary header provides basic packet control data (e.g. packet sequence count, packet length, virtual channels, etc.). The CCSDS secondary header provides time information.

CCSDS Packet Data			
Primary Header	Secondary Header (Time)	Data	CRC
6 bytes	9 bytes	0–236 bytes	1 byte

Figure 16-2: CII CCSDS Packet Data

CCSDS Primary Header						
	Packet Identification			Packet Sequence Control		
Packet Version Number	Type	Secondary Header Flag	APID	Sequence Flags	Sequence Count	Packet Length
3 bits	1 bit	1 bit	11 bits	2 bits	14 bits	16 bits
2 bytes				2 bytes		2 bytes

Figure 16-3: CCSDS Primary Header

CCSDS Secondary Header								
CCSDS Unsegmented Time Code (CUC) Field								
P-Field		T-Field						
Default	Extended	Coarse Time				Fine Time		
1 byte	1 byte	1 byte	1 byte	1 byte	1 byte	1 byte	1 byte	1 byte

Figure 16-4 CCSDS Secondary Header (Time)

16.1.9.1 *Elapsed Time since Default CCSDS Epoch*

The CII CCSDS secondary header should contain the elapsed time (ET) in coarse (seconds) and fine (sub-second) units since the default CCSDS epoch of January 1, 1958.

Rationale: CCSDS headers are the established industry standard for space packet communication. The CCSDS secondary header provides time information. CCSDS Time is a common way to represent time in format that is straightforward for computers to store and manipulate.

Four bytes of coarse time allows approximately 136 years (year 2094) of time resolution. Three bytes of fine time allows up to 60 ns of sub-second resolution.

For example, the second draft revision of this document was created on Monday, February 7, 2011. To represent this date and time as a CII time, determine the number of seconds between January 1, 1958 and February 7, 2011: 1,675,728,000 seconds. This corresponds to the four byte coarse time: `0x63 0xE1 0x94 0x80` and three byte fine time: `0x00 0x00 0x00`. If the next draft were issued 125ms later, the fine time would increment by $2^{24} \div 8 = 2,097,152$ sub-seconds, for a three byte fine time: `0x02 0x00 0x00`.

16.1.9.2 *Extended Time P-Field*

The next SpaceWire 6-bit time code should be embedded in the CUC Extended time P-Field, as defined in the SpaceWire – CCSDS Unsegmented Code Transfer Protocol (proposed).

Rationale: Synchronizing the next SpaceWire time code with the current time packet provides a convenient mechanism to indicate when the time should take effect. This also allows the SpaceWire time code to replace the 1 PPS time signal (see Section 6.3.1). Although three bytes of CCSDS CUC fine time allow 60 ns resolution, the delivery of SpaceWire time codes has an approximate resolution of only one microsecond.

Table 16-2 provides additional detail on the CII CCSDS primary header including valid values and value ranges for each field.

Table 16-2: CCSDS Primary Header Fields

Field	Length (bits)	Value	Comments
Packet Version Number	3	`0b001`	CII Packet Version 1
Packet Type	1	`0b0`	CII SpaceWire-CCSDS packet
Secondary Header Flag	1	`0b1`	Indicates the CII-CCSDS secondary header is present
Application Process Identifier	11	0–2048	Reserved for instrument use (e.g. may be used to create virtual science data channels).
Sequence Flags	2	`0b11`	Indicates packet is unsegmented
Sequence Count	14	0–16383	Specifies the packet sequence count. The packet count begins at zero, is incremented by one for each successive packet sent, and resumes at zero after 16384 packets have been sent.
Packet Length	16	0–236	Specifies the length of the packet data field and CRC byte in bytes.

16.1.9.3 *Packet Version Number*
The CII CCSDS Packet Version Number bits should be set to `0b001`.

16.1.9.4 *Packet Type*
The CII CCSDS Packet Type bit should be set to `0b0`.

16.1.9.5 *Secondary Header Flag*
The CII CCSDS Secondary Header Flag bit should be set to `0b1`.

16.1.9.6 *Sequence Flags*
The CII CCSDS Sequence Flags bits should be set `0b11`.

16.1.9.7 *Distinct Sequence Counts*
The Instrument and spacecraft should maintain distinct CII CCSDS packet sequence counts.

16.1.9.8 *Sequence Count Initial Value*
The CII CCSDS Sequence Count should begin at zero.

16.1.9.9 *Sequence Count Incrementing*
The CII CCSDS Sequence Count should increment by one for each successive packet sent.

16.1.9.10 *Sequence Count Rollover*
The CII CCSDS Sequence Count should resume at zero after 16384 packets have been sent.

16.1.9.11 *Packet Length Field*

The CII CCSDS Packet Length should specify the length of the packet data field and CRC byte in bytes.

The CII CCSDS Application Processes Identifier (APID) is reserved for Instrument use. For example, it may be used to create virtual science data channels.

16.1.10 Spacecraft Status Message

16.1.10.1 *Receipt and Processing*

The Instrument should receive and process CII Spacecraft Status Messages containing the current spacecraft time (via the CII CCSDS secondary header) and position.

16.1.10.2 *Updates to Instrument Internal Clock*

The Instrument should update its internal clock to the current Spacecraft time upon receipt of the SpaceWire time code contained in the P-Field of the CCSDS Secondary header (see Section 16.1.9.1).

16.1.10.3 *Spacecraft Status Message Intervals*

The Spacecraft should send status messages at one second intervals.

16.1.10.4 *Spacecraft Status Message Packet Structure*

The structure of a complete CII Spacecraft Status Message packet should conform to that depicted in Figure 16-5 and Figure 16-6.

CII SpaceWire Spacecraft Status Message Packet							
SpW Logical Address	Cargo						ExP
	Protocol ID (`0x02`)	CCSDS					
		Reserved (`0x00`)	Type (`0x02`)	Header	Data	CRC	
1 byte	1 byte	1 byte	1 byte	15 bytes	80 bytes	1 byte	4 bits
	99 bytes						

Figure 16-5: CII Spacecraft Status Message Packet

Data		
Ephemeris *X*	Ephemeris *V*	Quaternion
24 bytes	24 bytes	32 bytes

Figure 16-6: Data portion of CII Spacecraft Status Message Packet

The Ephemeris Message represents the current spacecraft position with three vectors, a position (X), a velocity (V), and a quaternion (Q).

16.1.10.5 *Position Vector*

The three-element position vector (X) should represent the spacecraft position with respect to the center of the Earth, in true-of-date inertial frame, in units of meters.

16.1.10.6 *Velocity Vector*

The three-element velocity vector (*V*) should represent the spacecraft inertial velocity, in true-of-date inertial frame in units of meters/second.

16.1.10.7 *Quaternion*

The four-element, non-dimensional quaternion (*Q*) should define the transformation from the local-vertical, local-horizontal frame (LVLH) to the spacecraft body frame. Q_0 is the scalar term and Q_1, Q_2, Q_3 are the vector terms.

Note: Guidelines 16.1.10.5-16.1.10.7 follow *Space Test Program – Standard Interface Vehicle (STP-SIV) Payload User's Guide* for representing spacecraft position.

16.1.10.8 *Floating Point Encoding*

Each element of the position vector (*X*), velocity vector (*V*), and quaternion (*Q*) should be encoded as a big-endian IEEE-754 floating-point value.

16.1.11 <u>Instrument Command</u>

16.1.11.1 *Spacecraft-Originated Command Messages*

The Instrument should receive and process CII Command Messages from the Spacecraft.

16.1.11.2 *Ground-Originated Command Messages*

The Instrument should receive and process CII Command Messages from the ground (via the Spacecraft).

16.1.11.3 *Command Processing Rates*

The Instrument should receive and process no more than 10 spacecraft or ground commands per second.

16.1.11.4 *Command Execution Priority*

With the exception of the Enter SAFE Mode command message, the instrument should process all CII Command Messages in the order received.

16.1.11.5 *Command Message Packet Structure*

The structure of a complete CII Command Message packet should conform to that depicted in Figure 16-7 and Figure 16-8.

<table>
<tr><th colspan="8">CII SpaceWire Command Message Packet</th></tr>
<tr><td rowspan="3">SpW Logical Address</td><td colspan="6">Cargo</td><td rowspan="3">ExP</td></tr>
<tr><td rowspan="2">Protocol ID (0x02)</td><td colspan="5">CCSDS</td></tr>
<tr><td>Reserved (0x00)</td><td>Type (0x04)</td><td>Header</td><td>Data</td><td>CRC</td></tr>
<tr><td rowspan="2">1 byte</td><td>1 byte</td><td>1 byte</td><td>1 byte</td><td>15 bytes</td><td>3–236 bytes</td><td>1 byte</td><td rowspan="2">4 bits</td></tr>
<tr><td colspan="6">22–255 bytes</td></tr>
</table>

Figure 16-7: CII Command Message Packet

Data		
Command Source	Command ID	Command Data
1 byte	2 bytes	0–233 bytes

Figure 16-8: Data portion of CII Command Message Packet

The Command Source field indicates whether the command message originated from the spacecraft or the ground (operations team) (see Table 16-3).

Table 16-3: CII Command Source

Command Source	Command Origination Point
`0x01`	Spacecraft
`0x02`	Ground

16.1.11.6 *Spacecraft-Originated Command Source Byte*

For command messages that originate on the spacecraft, the CII Command Message Command Source byte should be set to `0x01`.

16.1.11.7 *Ground-Originated Command Source Byte*

For command messages that originate on the ground (e.g. uplinked by the operations team), the CII Command Message Command Source byte should be set to 0x02.

16.1.11.8 *Command Message Command Identifier Bytes*

The CII Command Message Command Identifier bytes should uniquely identify the command to be executed.

Most instrument command messages will be defined by the instrument provider and are largely opaque to the spacecraft provider. Therefore, the use and contents of the CII Command Message Command Data field are at the discretion of the instrument provider. CII defines only four basic commands (see Table 16-4).

Table 16-4: CII Command ID's

Command ID	Command Message
`0x0000`	No Operation (no-op)
`0x0011`	Start Sending Telemetry
`0x00FF`	Stop Sending (Telemetry or Science) Data
`0x5AFE`	Enter SAFE mode
`0x00FF–0xFFFF`	Instrument specific commands (excluding 0x5AFE)

16.1.11.9 *No Operational Command Message*

The instrument should acknowledge a No Operation (no-op) command message as received and successfully processed.

Rationale: A no-operation command is common practice. It allows the spacecraft and ground operations team to determine whether or not the instrument's basic command subsystem is operating nominally.

16.1.11.10 *Command Message Validation with CRC Byte*
The Instrument should reject any CII Command Message with an invalid CRC byte.

Rationale: Command verification is a best practice. It allows the spacecraft and ground operations team to devise and operate FDIR procedures.

16.1.11.11 *Resumption of Telemetry Transmission*
When data transmission is suspended, the Instrument should resume telemetry transmission only upon receipt of a Start Sending Telemetry command.

16.1.11.12 *Suspension of Telemetry Transmission*
When telemetry and/or science data transmission is in progress, the Instrument should immediately suspend both telemetry and science data transmission upon receipt of the Stop Sending Data command.

Rationale: Data flow control is common practice. It allows the Spacecraft and ground operations team to devise and operate FDIR procedures.

16.1.11.13 *Command Definition for Science Data Transmission Control*
Instrument providers should define commands to disable, enable, and/or resume the transmission of only science data.

Rationale: Data flow control is common practice. However, commands for disabling, enabling, and/or resuming the transmission of science data may require instrument specific parameters (science data sets, locations in memory or on disk, begin and end observation times, etc.) and are therefore outside the scope of this document.

16.1.11.14 *SAFE Mode Command Message*
The Instrument should enter SAFE mode when it receives an Enter SAFE Mode command message command from the Spacecraft or ground.

16.1.11.15 *SAFE Mode Command Priority*
The Instrument should immediately process the Enter SAFE Mode command message ahead of any queued, but not yet processed, commands.

Rationale: Transition to SAFE mode is a high priority event. For the Instrument's, and possibly Spacecraft's, own health and safety, SAFE mode entry should be immediately accommodated.

16.1.12 Instrument Command Acknowledgement

16.1.12.1 *Command Acknowledge Message*
The Instrument should reply to every command received with a CII Command Acknowledgment Message.

16.1.12.2 *Command Acknowledgement Rate*

The instrument should acknowledge no more than 10 Spacecraft or ground commands per second.

16.1.12.3 *Command Acknowledgement Message Packet Structure*

The structure of a complete CII Command Acknowledgement Message packet should conform to that depicted in Figure 16-9 and 16-10.

CII SpaceWire Command Acknowledgement Message Packet							
SpW Logical Address	Cargo						ExP
	Protocol ID (0x02)	CCSDS					
		Reserved (0x00)	Type (0x08)	Header	Data	CRC	
1 byte	1 byte	1 byte	1 byte	15 bytes	9 bytes	1 byte	4 bits
	28 bytes						

Figure 16-9: CII Command Acknowledgement Packet

Data				
Instrument Mode	Command Source	Command ID	Command Sequence Count	Command Status
2 bytes	1 byte	2 bytes	2 bytes	2 bytes

Figure 16-10: Data portion of CII Command Acknowledgement Packet

16.1.12.4 *Instrument Major Mode*

The first two bits of the CII Command Acknowledgement Message Instrument Mode bytes should denote the instrument major mode: INITIALIZATION (`0b00`), OPERATION (`0b01`), and SAFE (`0b10`) (see Section 5.1).

16.1.12.5 *Invalid Major Mode Values*

The first two bits of the CII Command Acknowledgement Message Instrument Mode bytes should never be set to the invalid value of `0b11`.

16.1.12.6 *Instrument Submodes*

The remaining 14 bits of the CII Command Acknowledgement Instrument Mode may be used at the instrument's discretion to indicate sub-modes within any major mode (see Section 5.1).

16.1.12.7 *Fields to Echo*

The CII Command Acknowledgement Command Source, Command ID, and CCSDS Primary Header Sequence Count fields should be echoed from the CII Command Message being acknowledged.

16.1.12.8 *Status of Last Command*

The CII Command Acknowledgement Command Status field should indicate the status of the last command (Table 16-4).

Table 16-4: Command Status

Command Status	Description
`0x0000`	Command Received and Processed Nominally
`0x0002`	Command Packet CRC Error
`0x0004`	Invalid Flow Control Operation
`0x0008`	Instrument Already in SAFE mode
`0x00FF–0xFFFF`	Instrument Specific Command Status Codes

16.1.12.9 *Acknowledgement of Messages with Invalid CRC*
The Instrument should acknowledge any CII Command Message with an invalid CRC byte with a Command Packet CRC error Command Status code (`0x00002`).

Rationale: Notification of potentially invalid commands is a common practice.

16.1.12.10 *Start Sending Telemetry Acknowledgement with Telemetry in Progress*
When telemetry transmission is in progress, the Instrument should acknowledge a Start Sending Telemetry command message with an Invalid Flow Control Operation Command Status code (`0x0004`).

16.1.12.11 *Stop Sending Telemetry Acknowledgement with Telemetry Suspended*
When data transmission is suspended, the Instrument should acknowledge a Stop Sending Data command message with an Invalid Flow Control Operation Command Status code (`0x0004`).

16.1.12.12 *Instrument Already in SAFE Mode Acknowledgement*
When in SAFE mode, the Instrument should acknowledge an Enter SAFE Mode command message with an Instrument Already in SAFE Mode Command Status code (`0x0008`).

16.1.12.13 *Off-Nominal or Erroneous Condition Acknowledgement*
If the processing of any CII Command Message fails an internal command or argument sanity check or results in an off-nominal or erroneous condition, the Instrument should acknowledge with a specific and precise instrument specific Command Status code `(0x00FF–0xFFFF)`.

16.1.12.14 *Command Received and Processed Nominally Acknowledgement*
If the processing of any CII Command Message was successful and nominal in all respects, the Instrument should acknowledge with a Command Received and Processed Nominally Command Status code `(0x0000)`.

Rationale: Explicit, specific status codes are common practice.

16.1.13 Instrument Telemetry
CII Instrument Telemetry is sent without acknowledgement or retries. An invalid CRC may be used to identify and flag corrupt packet data.

16.1.13.1 *EHA Telemetry Transmission Rate*
The Instrument should transmit engineering, health, and accountability (EHA) telemetry data at a rate of up to 10 Hz.

16.1.13.2 *Instrument Telemetry Message Packet Structure*
The structure of a complete CII Instrument Telemetry Message packet should conform to that depicted in Figure 16-11 and Figure 16-12.

CII SpaceWire Instrument Telemetry Message Packet							
SpW Logical Address	Cargo						ExP
	Protocol ID (`0x02`)	CCSDS					
		Reserved (`0x00`)	Type (`0x10`)	Header	Data	CRC	
1 byte	1 byte	1 byte	1 byte	15 bytes	236 bytes	1 byte	4 bits
	255 bytes						

Figure 16-11: CII Telemetry Packet

Data				
Instrument Mode	Telemetry Bank	Telemetry Data		
2 bytes	2 bytes	4 bytes	…	4 bytes

Figure 16-12: Data portion of CII Telemetry Packet

16.1.13.3 *Instrument Major Mode*
The first two bits of the CII Telemetry Message Instrument Mode bytes should denote the instrument major mode: INITIALIZATION (`0b00`), OPERATION (`0b01`), and SAFE (`0b10`) (see Section 5.1).

16.1.13.4 *Instrument Major Mode Invalid Value*
The first two bits of the CII Telemetry Message Instrument Mode bytes should never be set to the invalid value of `0b11`.

16.1.13.5 *Instrument Sub-Modes*
The remaining 14 bits of the CII Command Acknowledgement Instrument Mode may be used at the Instrument's discretion to indicate sub-modes within any major mode (see Section 5.1).

CII Telemetry Messages contain up to 58 telemetry values per packet. A group of 58 values comprises a single telemetry bank. A telemetry bank is simply a numerical identifier used to logically identify a set of telemetry values. The rotation order and frequency of telemetry banks is at the discretion of the Instrument.

16.1.13.6 *Unused Value Slots*
Instrument telemetry banks containing less than 58 telemetry values, should set unused value slots to zero (`0x00000000`).

16.1.13.7 *Telemetry Value Encoding*
Each Instrument telemetry value should be encoded big-endian as either a standard two's complement 32-bit signed integer, a 32-bit unsigned integer, or an IEEE-754 floating-point values.

16.1.13.8 *Telemetry Dictionary*
All Instrument telemetry should be documented in a telemetry dictionary, with a complete listing of telemetry banks, telemetry value indexes within each bank, and telemetry value types to the spacecraft provider.

Rationale: A common format to communicate telemetry values will help facilitate straightforward monitoring of certain safety-critical telemetry values by the spacecraft (e.g. to command the instrument to SAFE mode if a thermal limit is exceeded) and/or ground operations team.

16.1.13.9 *Diagnostic Data*
In addition to sensor read backs, Instrument Providers may include software versions, status codes, internal counters, and other useful diagnostic data in their telemetry definitions.

The CII Data Interface guidelines for instrument telemetry are intended to replace discrete read backs (e.g. thermistors, relay status, etc.). While discrete line interfaces are best practice, CII does not define a specific cable level discrete interface. Furthermore, it is extremely advantageous for Instrument Providers to minimize their use of and dependence on discrete line interfaces. Doing so will maximize HPO's. Many spacecraft providers have communicated to the CII Team that they may not be able to provide many or any discrete line interfaces to hosted payloads.

16.1.14 <u>Instrument Science Data</u>
CII Instrument Science Data is sent without acknowledgement or retries. An invalid CRC may be used to identify and flag corrupt data.

16.1.14.1 *Maximum Number of Science Data Packets*
The instrument should limit its maximum number of science data packets to match the SpaceWire date rate specified in Section 16.1.3.

16.1.14.2 *Science Data Message Packet Structure*
The structure of a complete CII Science Data Message packet should conform to that depicted in Figure 16-13 and Figure 16-14.

CII SpaceWire Instrument Science Data Message Packet							
SpW Logical Address	Cargo						ExP
	Protocol ID (0x02)	CCSDS					
		Reserved (0x00)	Type (0x20)	Header	Data	CRC	
1 byte	1 byte	1 byte	1 byte	15 bytes	236 bytes	1 byte	4 bits
	255 bytes						

Figure 16-13: CII Science Data Packet

Data
Science Data
236 bytes

Figure 16-14: Data portion of CII Science Data Packet

Protocol Overhead: According to data found in Appendix B of the JWST SpaceWire Specification, SpaceWire science packets for NIRCam, NIRSpec, MIRI, FGS each have a 23% total (SpaceWire plus protocol) overhead. The proposed CII protocol has 26% total overhead. (Since SpaceWire encodes each byte using 10 bits, it is not possible to achieve better than 20% total overhead.) For science data, JWST has half the protocol-only overhead as the proposed CII protocol.

For comparison, on a 200 Mbps link, JWST can achieve a maximum of 154 Mbps of science data while CII can achieve 148 Mbps. JWST uses a custom SpaceWire – CCSDS protocol (likely because the SpaceWire – CCSDS Standard was not available when JWST developed their protocol) for all SpaceWire communications *except* science data.

16.2 Electrical Power Interface Reference Material / Best Practices

16.2.1 <u>Electrical Interface Definitions</u>

16.2.1.1 *Power Interface*

There should be three redundant power buses feeding each Instrument. They are designated as the Power Feed #1, Power Feed #2, and the Survival Heater Power Feed.

16.2.1.2 *Power Interface Returns*

Each power bus should have separate ground returns.

16.2.1.3 *Power Bus Redundancy*

Each power bus should be redundant with independent sides designated as Power Bus A and Power Bus B.

16.2.1.4 *Power Bus Current Rate of Change*

For power bus loads with current change greater than 2 A, the rate of change of current should not exceed 500 mA/µs.

16.2.1.5 *Power Bus*

Each power bus is redundant with primary and redundant power designated as Power Bus A and Power Bus B.

16.2.1.6 *Survival Heater Power Bus*

The Survival Heater Power Bus is a redundant power bus. SURVIVAL is a Spacecraft-initiated mode in which operational power to the Instruments is interrupted. Survival power is used only for heaters and associated passive control circuitry which maintain the Instrument at minimum turn-on temperature. Sides A and B are designated as Survival Heater Bus A and Survival Heater Bus B.

16.2.1.7 *Survival Heater Redundancy*

The survival heaters should be redundant.

16.2.1.8 *Survival Heater Power Bus Isolation*

The survival heater power bus primary and redundant power should be electrically isolated from each other and from chassis.

16.2.1.9 *Survival Heater Power Bus Returns*

The survival heater power bus should have independent power returns. The Spacecraft can switch off survival power buses, which will normally be continuously powered during flight.

16.2.1.10 *Survival Heater Power Bus Power Limit*

Instrument survival heater power should not exceed 60W end-of-life (EOL) at worst-case low bus voltage. This power covers all of the heater power requirements required to maintain the Instrument at or above the non-operational survival temperature when the spacecraft is either in the Launch and Early orbit Operation (LEOP) portion of the mission, in SAFE Mode, or performing an orbit correction maneuver. The spacecraft provides the switched power to the Instrument when required by the mission.

16.2.1.11 *Survival Heater Power Bus Circuit Failure*

The Instrument design should prevent a stuck on condition of the survival heaters due to internal failures.

16.2.1.12 *Survival Heater Power Bus Heater Type*

The Instrument should use survival power only for resistive heaters (and associated thermal control device) which maintain the Instrument at minimum turn-on temperature when the main power bus is disconnected from the Instrument.

16.2.1.13 *Survival Heater Power Bus Design*

The system design should be such that having both primary and redundant survival heater circuits enabled does not violate any thermal or power requirement.

16.2.1.14 *Survival Heater Power Bus Isolation*

Survival heater power buses should be electrically isolated from each other, from other Instrument thermal control, from chassis, and have independent power returns.

16.2.1.15 *Survival Heater Power Bus Emergency*

The Spacecraft should ensure that both the primary and redundant survival heater circuits are normally enabled on-orbit when an Instrument is off. However, even the survival heaters may be turned off in the event of an emergency where the survival of the spacecraft is in jeopardy.

16.2.1.16 *Spacecraft Operational Power Switching*

The Spacecraft should provide switched Primary and Redundant Operational Power to the Instrument.

16.2.1.17 *Spacecraft Survival Heater Power Bus Power Application*

The Spacecraft should sequentially apply power to each survival heater power feed.

16.2.1.18 *Spacecraft Survival Heater Power Bus Switching*

The Spacecraft should provide switched Primary and Redundant Survival Power to the Instrument.

16.2.1.19 *Energy Recovery Mode*

In the event that the Spacecraft battery state-of-charge falls below 50%, the Spacecraft will power off the Instrument after appropriately placing the Instrument in SAFE mode, and Instrument operation will not resume until the ground operators have determined it is safe to return to OPERATION mode.

16.2.1.20 *Low Voltage Detection*

A voltage excursion that causes the spacecraft Primary Power Bus to drop below 22VDC in excess of four seconds constitutes an under-voltage condition. In the event of an under-voltage condition, the spacecraft will shed various loads without delay, including the Instrument. A ground command should be required to re-power the load.

16.2.1.21 *Bus Undervoltage and Overvoltage Transients*

The Instrument should be able to survive without damage a power bus undervoltage or overvoltage condition occurring.

16.2.1.22 *Bus Undervoltage and Overvoltage Transients*

Derating factors should take into account the stresses that components are subjected to during periods of undervoltage or overvoltage, including conditions which arise during ground testing, while the bus voltage is slowly brought up to its nominal value.

16.2.1.23 *Bus Undervoltage and Overvoltage Transients*

The Instrument should not generate a spurious response that can cause equipment damage or otherwise be detrimental to the spacecraft operation during bus voltage variation, either up or down, at ramp rates below the limits specified in the sections below, and over the full range from zero to maximum bus voltage.

16.2.1.24 *Normal Transients from Load Changes*

The bus voltage should not vary by more than 1.0 V during a 5 A in 1 ms step load change (+1 V during a -5 A step, -1 V during a +5A step).

16.2.1.25 *Normal Transients from Eclipse Entry*
At entry into eclipse, the bus voltage should not vary by more than –0.2 V/s, with a maximum voltage change of –6 V, not to go below 22 VDC at the Instrument input.

16.2.1.26 *Normal Transients from Eclipse Exit*
At exit from eclipse, the bus voltage should not change by more than 0.2 V/s, with a maximum change in sunlight of +6 V, not to exceed 34 VDC at the Instrument input.

16.2.1.27 *Abnormal Transients Undervoltage*
An abnormal undervoltage event should be defined as a transient decrease in voltage on the +28 VDC bus to no less than +10 VDC, maintaining the decreased voltage for no more than 10 ms, and returning to its previous voltage in less than 200 ms.

16.2.1.28 *Abnormal Transients Undervoltage Tolerance*
All Spacecraft components should ensure that overstress does not occur to the unit during the undervoltage.

16.2.1.29 *Abnormal Transients Recovery*
Units which shut-off during an undervoltage should return to a nominal power-up state at the end of the transient.

16.2.1.30 *Abnormal Transients Overvoltage*
An overvoltage event should be defined as a transient increase in voltage on the +28 VDC bus to no greater than +40 VDC, maintaining the increased voltage for no more than 10 ms, and returning to its previous voltage in less than 200 ms.

16.2.1.31 *Abnormal Transients Overvoltage Tolerance*
All spacecraft components should operate through the overvoltage transient, with no degradation in performance, and no overstress of electrical components.

16.2.1.32 *Power Bus Ripple Voltage*
The ripple voltage on the power bus should be less than 5% peak-to-peak of line voltage (28 V) over all frequency ranges from 1 Hz to 10 MHz. The output ripple of the power subsystem alone should be less than 250 mV peak-to-peak over the frequency range of 1 Hz to 10 MHz.

16.2.1.33 *Power Bus Ripple Limit*
The output ripple of the power subsystem including the effect of all loads should not exceed 500 mV peak-to-peak over the frequency range of 1 Hz to 10 MHz.

16.2.1.34 *Instrument Turn-on Transients*
During initial turn-on of the Instrument, or as a result of a change in operating mode, the operational inrush current drawn by the DC power input should meet the guidelines in the following paragraphs (see Figure 16-15).

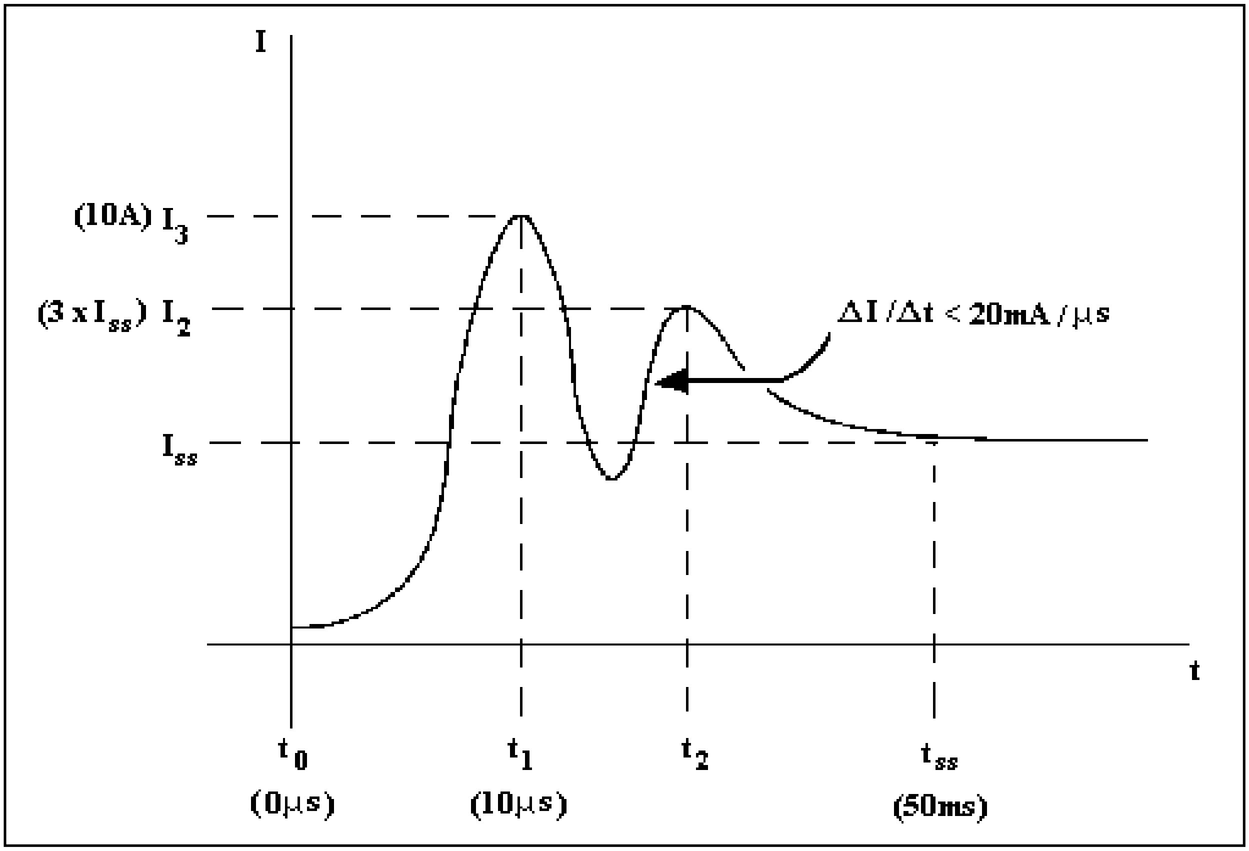


Figure 16-15: Maximum Inrush Current

16.2.1.35 *Instrument Initial In-rush Current*
After application of +28 VDC power at t_0, the initial inrush (charging) current due to distributed capacitance, EMI filters, etc., should be completed in 10 µs with its peak no greater than 10 A.

16.2.1.36 *Instrument Initial In-rush Current Rate of Change*
The rate of change of inrush current after the initial application of +28V power should not exceed 20 mA/µs.

16.2.1.37 *Instrument In-rush Current after 10 µs*
After 10µs, the transient current peak should not exceed three times the maximum steady state current.

16.2.1.38 *Instrument Steady State Operation*
Steady state operation should be attained within 50 ms from turn-on or transition to OPERATION mode, except for motors.

16.2.1.39 *Instrument Turn-off Peak Voltage Transients*
The peak voltage of transients generated on the Instrument side of the power relay caused by inductive effects of the load should fall within the -2 VDC to +40 VDC range.

16.2.1.40 *Instrument Turn-off Transient Suppression*
The Instruments should use suppression devices, such as diodes, across all filter inductors, relay coils, or other energy sources that could induce transients on the power lines during turn-off.

16.2.1.41 *Instrument Turn-off Transient Suppression Device Location*
The suppression devices should be located at the source of the inductive transients.

16.2.2 Reflected Ripple Current – Steady State
The peak-to-peak amplitude of steady state load current generated by the Instrument should not exceed 2% of the maximum average steady state current drawn by the Instrument.

16.2.3 Reflected Ripple Current – Mode Changes
The load current ripple due to rpm mode changes should not exceed 2 times the steady state current during the period of the motor spin-up or spin-down.

16.2.3.1 *Instrument Operational Transients Current Limit*
Operational transients that occur after initial turn-on should not exceed 125% of the peak operational current drawn during normal operation.

16.2.3.2 *Instrument Operational Transients Duration*
The maximum duration of the transients should not exceed 50 ms.

16.2.3.3 *Instrument Reflected Ripple Current*
The peak-to-peak load current ripple generated by the Instrument should not exceed 25% of the average current on any Power Feed bus.

16.2.3.4 *Overcurrent Protection Definition*
The analysis defining the overcurrent protection device specification(s) should consider turn-on, operational, and turn-off transients.

16.2.3.5 *Overcurrent Protection – Harness Compatibility*
Harness wire sizes should be consistent with overcurrent protection device sizes and de-rating factors.

16.2.3.6 *Overcurrent Protection – Power Supply Compatibility*
Power supply output capability should be adequate to clear a fuse or other overcurrent protection device.

16.2.3.7 *Overcurrent Protection Device Size Documentation*
The agreed-upon type, size, and characteristics of the overcurrent protection device(s) should be documented in the Spacecraft to Instrument ICD.

16.2.3.8 *Instrument Internal Overcurrent Protection*
All Instrument internal overcurrent protection devices should be accessible at the Spacecraft integration level without the disassembly of the Instrument.

16.2.3.9 *Instrument Internal Fuses*

The Instrument should not have internal fuses.

16.2.3.10 *Instrument Fault Propagation Protection*

The Instrument and spacecraft should not propagate a single fault occurring on either the “A” or “B” power interface circuit, on either side of the interface, to the redundant interface or Instrument.

16.2.3.11 *Instrument High-Voltage Ambient Operation*

To allow ambient testing, Instrument high-voltage supplies should be capable of being operated at ambient pressure.

16.2.3.12 *Instrument High-Voltage Current Limiting*

The output of each Instrument's high-voltage supply should be current limited to prevent the supply's discharge from damaging the Spacecraft and other Instruments.

16.2.3.13 *Instrument High-Voltage Disabling*

If an Instrument high-voltage supply cannot be operated at ambient pressure, it should be appropriately disabled by manual means to allow ambient testing of the instrument.

16.2.3.14 *Isolation*

Isolation requirements from primary power to chassis and primary power to secondary power should be adhered to at each primary power service.

16.2.3.15 *Power Input Isolation*

At the Instrument interface, isolation should exist between the six power buses shown in Figure 7-1.

16.2.3.16 *Primary Power Isolation*

The 28 VDC primary power leads and returns should be isolated from signal and signal return, and chassis ground by greater than 1 MΩ when measured at the Instrument input.

16.2.3.17 *Secondary Power Isolation*

Secondary power should be isolated from 28 VDC primary power by greater than 1 MΩ.

16.2.3.18 *Secondary Power Return*

Secondary power circuits should provide current return leads from each Instrument component which utilizes secondary power.

16.2.3.19 *Secondary Power Return*

These secondary power returns should be connected at a single point, referred to as the Secondary Power Reference.

16.2.3.20 *Secondary Power Reference*

The Secondary Power Reference should bond to the chassis ground.

16.2.3.21 *Isolated Secondary Referencing*
If isolated secondary power is used within the Instrument, the isolated returns should be referenced to chassis ground through a resistive impedance whose value is selected by the Instrument provider and not left floating.

16.2.3.22 *Signal Reference*
The Instrument's Secondary Power Reference and the signal reference for Spacecraft interface circuits should be the same point electrically.

16.2.3.23 *Signal Reference Connectivity*
All Instrument signal references for the Spacecraft interface circuits should electrically connect within the Instrument.

16.2.3.24 *Signal Reference Constraints*
Neither signal nor chassis ground reference points should be used as power conductors.

16.2.4 <u>Chassis Ground</u>
16.2.4.1 *Instrument Ground Plane*
The Spacecraft Provider should furnish a common electrically conductive ground plane to which all Instrument chassis should be electrically connected.

16.2.4.2 *Chassis Ground Current*
Instruments should not use chassis ground to conduct power and signal currents under normal conditions. Only fault and leakage currents should be conducted through chassis grounds.

16.2.5 <u>Component Grounding</u>
16.2.5.1 *Component Ground Location*
The Instrument should provide a designated chassis ground terminal on each component, as close to all electrical connectors as possible, for connection to the ground plane.

16.2.6 <u>External Grounding Tie Point</u>
16.2.6.1 *External Ground Tie Point*
Each Instrument should identify in the Electrical Interface Control Drawing an external chassis ground tie point to be used for external connections while the Instrument is being moved.

16.2.7 <u>Signal Reference Plane</u>
16.2.7.1 *Instrument Ground Plane Connection*
The Instrument Ground Plane is the electrically conductive surface to which all Instrument components are electrically connected (bonded). The Spacecraft Contractor should electrically connect the Instrument ground plane to the Signal Reference Plane.

16.2.8 <u>Thermal Blanket Grounding</u>
16.2.8.1 *Thermal Blanket Layer Interconnection*
All thermal insulation blankets should be designed with metallized and conductive layers electrically interconnected such that the resistance between layers is less than 10 Ω (DC).

16.2.8.2 *Thermal Blanket Chassis Grounding*

Thermal insulation blankets should be connected to chassis ground with a resistance of less than 10 Ω (DC).

16.2.9 Grounding of External Surfaces

To prevent electrostatic charging, external insulating materials and surface finishes should have a surface resistivity of less than or equal to 1×10^9 ohms per Square.

16.2.10 Grounding of External Surfaces

The following parameters in Table 16-5 should apply to surface coatings (e.g., conductive paints, thermal protection):

Table 16-5: Surface Resistivity

Description	Maximum Surface Resistivity (ohms per Square)
Surface Coating over Dielectric	1×10^9
Surface Coating over Metal or Conductive Composite	$\frac{1 \times 10^9}{\text{coating thickness (cm)}}$
Thin Film Material	1×10^{12}

16.2.11 Grounding of External Surfaces

The resistance of the connection between the conductive paint and the basic structure should be less than 1×10^5 Ω when dry.

16.2.12 Power Return

Power return conductors should be provided for all current drawn from the +28 VDC Power Bus.

16.2.13 Power Return Isolation

Power return-to-chassis isolation should be at least 1 MΩ.

16.2.14 Signal Ground DC/DC Converter Reference

Signal ground should be the zero reference voltage for the secondary side of all instrument DC/DC converters.

16.2.15 Signal Ground Telemetry Reference

Signal ground should be the ground reference for all telemetry circuits.

16.2.16 Signal Ground

A signal ground wire should be connected from each component to the spacecraft Secondary Signal Reference Point.

16.2.17 Signal Return Continuity

Signal ground-to-chassis continuity should be less than 10 mΩ.

16.2.18 Signal Return Separation

Each connector should have at least two separate pins for analog and digital signal return.

16.2.19 Component Grounding
In general, Instrument housings should be grounded to the satellite structure via the component mounting feet or via the baseplate together with an electrically and thermally conductive adhesive.

16.2.20 Component Grounding
Any electrically nonconductive anticorrosion finish should be removed from the joint faces before grounding. The instrument may be grounded to the satellite structure via an external ground wire for the following reasons:

1) The component was not designed for grounding via the mounting feet.
2) The component requires thermal isolation from its mounting surface.
3) The component requires special, non-conductive, shock absorbing mounts.
4) The component employs electrically non-conductive adhesive between baseplate and panel.

16.2.21 Component Grounding
All chassis-to-structure electrical bonds should comply with NASA-STD-4003, with a maximum DC resistance of 10 mΩ across the interface.

16.2.22 Component Grounding
The total DC resistance between Instrument housings and single point ground should not exceed 10 mΩ multiplied by the number of mechanical interfaces between the housing and spacecraft single-point ground.

16.2.23 Component Grounding
Interfaces between all isolated conducting items (except antenna) which are external to the vehicle and have any linear dimension greater than 3 inches, or otherwise are subject to frictional charging, should have secure connections to the spacecraft structure.

16.2.24 Component Grounding
The resistance of the connection should be less than 1 Ω as measured between the isolated conducting item and the nearest conductive spacecraft structural element.

16.2.25 Component Grounding
All metallic subchassis, chassis, and enclosures of the Instrument, including all component shields, lids, connector shells and other fittings, should be considered as electrical extensions of the ground plane.

16.2.25.1 *SpaceWire Harness Wiring Requirements*
The instrument SpaceWire harness should conform to ECSS-E-ST-50-12C.

16.2.26 Tie Points

16.2.26.1 *Tie Point Locations and Provider*

The locations of harness tie points should be based on an agreement between the Instrument Provider and the Spacecraft Provider.

16.2.26.2 *Tie Point Documentation*

The locations of the tie points should be documented in the appropriate Electrical Interface Control Drawing.

16.2.27 Connectors

The following should be applied to the selection and use of all interface connectors.

16.2.27.1 *Connector Selection*

Connector selection should consider materials outgassing, EMI, and reliability.

16.2.27.2 *Connectors – Magnetic Content*

Connectors should be constructed of low magnetic content materials with a residual magnetic field of < N×0.1nT where N represent the number of contacts.

16.2.27.3 *Connectors – Circular Features*

Circular connectors should feature scoop proof design and include grounding fingers.

16.2.27.4 *Connector Savers*

Throughout all hardware development, test, and integration phases, connector savers should be used to preserve the mating life of component flight connectors.

16.2.27.5 *Connector Saver Adapters*

The instrument provider should deliver connector saver adapters with high reliability connectors to minimize mating and demating operations with the flight connectors during integration and test.

16.2.27.6 *Connector Separation*

Separate harness interface connectors should be provided on all components for each of the following functions:

1) +28 VDC bus power and return
2) Telemetry and command signals with returns
3) Deployment actuation power and return (where applicable)

16.2.27.7 *Electrical Connector Constraints*

The connector half that has the primary power source should use socket contacts.

16.2.27.8 *Command and Telemetry Returns*

Telemetry return and relay driver return pins should be assigned on the same connector(s) as the command and telemetry signals.

16.2.27.9 *Connector Usage and Pin Assignments*
Harness side power connectors and all box/bracket mounted connectors supplying power to other components should have female contacts (connector does not have exposed contacts).

16.2.27.10 *Connector Contact Assignments*
The connector contact assignment should consider effects of electromagnetic coupling.

16.2.27.11 *Connector Function Separation*
Incompatible functions should be physically separated.

16.2.27.12 *Triaxial Connector Shields*
If triaxial cables are used for signals, the inner shield signal ground should be assigned to the pin adjacent to the signal.

16.2.27.13 *Connector Derating*
Instrument and Spacecraft should derate electrical connectors using *Electronic Parts, Materials, and Processes for Space and Launch Vehicles* (MIL-HDBK-1547A) as a guide.

16.2.27.14 *Connector Access*
At least 50 mm of clearance should exist around the outside of mated connectors.

16.2.27.15 *Connector Clearance*
The mated connectors should be accessible on the Spacecraft without removal of adjacent Instruments.

16.2.27.16 *Connector Location*
The Spacecraft Provider should define any areas in which connectors are not allowed as a result of satellite configuration constraints.

16.2.27.17 *Connector Engagement*
Connectors should be mounted to ensure straight and free engagement of the contacts.

16.2.27.18 *Connector Spacing*
Connectors should be spaced far enough apart so that the device can be held firmly either by hand or plug removal tool for connecting and disconnecting.

16.2.27.19 *Connector Location and Type Documentation*
Connector locations, orientations, and keyway locations should be identified in the appropriate Electrical Interface Control Drawing.

16.2.27.20 *Power Connector Type*
The Instrument power connectors should be space-flight qualified *General Specification for Connectors, Electric, Rectangular, Nonenvironmental, Miniature, Polarized Shell, Rack and Panel* (MIL-DTL-24308), Class M, Subminiature Rectangular connectors with standard density size 20 crimp contacts and conform to *Connectors, Electrical, Polarized Shell, Rack and Panel, for Space Use* (GSFC S-311-P-4/09).

16.2.27.21 *Power Connector Size and Conductor Gauge*
The Instrument power connectors should have 20 AWG, 9 conductor (shell size 1) or 15 conductor (shell size 2) connectors.

16.2.27.22 *Power Connector Pin Out*
The Instrument power connectors should utilize the supply and return pin outs defined in Table 16-6 and identified in Figure 16-16 thru Figure 16-18. *Note that the connectors are depicted with the instrument side of the connector (pins) shown while the spacecraft side of the connector (sockets) is the mirror image.*

Table 16-6: Instrument Power Connector Pin Out Definition

Power feed	Bus	Supply Conductor Position	Return Conductor Position
Power Feed #1	A	6	1
Power Feed #1	B	9	5
Power Feed #2	A	9	1
Power Feed #2	B	15	8
Survival Heater	A	2	7
Survival Heater	B	4	8

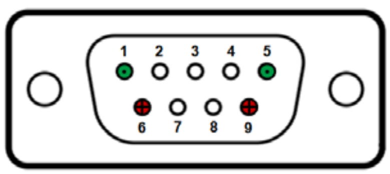


Figure 16-16: Instrument Side Power Feed #1 Bus A & Bus B

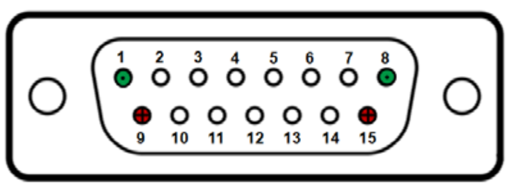


Figure 16-17: Instrument Side Power Feed #2 Bus A & Bus B

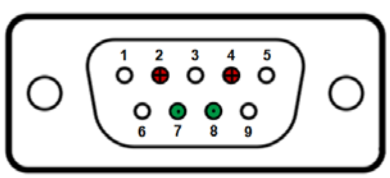


Figure 16-18: Instrument Side Survival Heater Feed Bus A & Bus B

16.2.27.23 *SpaceWire Connector Type*
The Instrument SpaceWire connectors should be space-flight qualified *General Specification for Connectors, Electrical, Rectangular, Microminiature, Polarized Shell* (MIL-DTL-83513), Class M, Microminiature Rectangular Polarized Shell connectors with size 24 crimp or solder contacts and conform to ECSS-E-ST-50-12C.

16.2.27.24 *SpaceWire Connector Pin Out*
The Instrument SpaceWire connectors should and conform to ECSS-E-ST-50-12C.

16.2.27.25 *Power Connector Provision*
The Instrument Provider should furnish all instrument power mating connectors (Socket Side) to the Spacecraft Provider for interface harness fabrication.

16.2.27.26 *Power Connector Conductor Size and Type*
The Instrument should have size 20 socket crimp contacts on the Instrument side power connectors and size 20 pin crimp contacts on the Spacecraft side power connectors.

16.2.28 Prevention of Mismating
Where adjacent signal connectors are the same shell sizes, alternate gender insert/contacts should be utilized.

16.2.29 Connector Orientation
Adjacent rectangular “D" shell connectors of the same size and gender should be rotated 180^{o} from each other.

16.2.30 Connector Marking
When receptacles of similar configuration are in close proximity, the mating plugs and receptacles should be suitably marked or coded to indicate clearly the mating connections.

16.2.31 Connector Configuration
All connectors should be different sizes, different types, or uniquely keyed to prevent incorrect connections with other accessible connector plugs or receptacles.

16.2.32 Prevention of Mismating
Alternative insert position or shell keyway polarizing should be specified to prevent incorrect connections with other adjacent connector plugs or receptacles of the same shell size. In most cases, normal (N) insert or shell key polarized items are to be chosen for ease of procurement and logistic support.

16.2.33 Prevention of Mismating
Physically adjacent shells should be different sizes to avoid the mismating of connectors.

16.2.34 Power Connector Keying
The instrument power connectors should be keyed as defined in Figure 16-19.

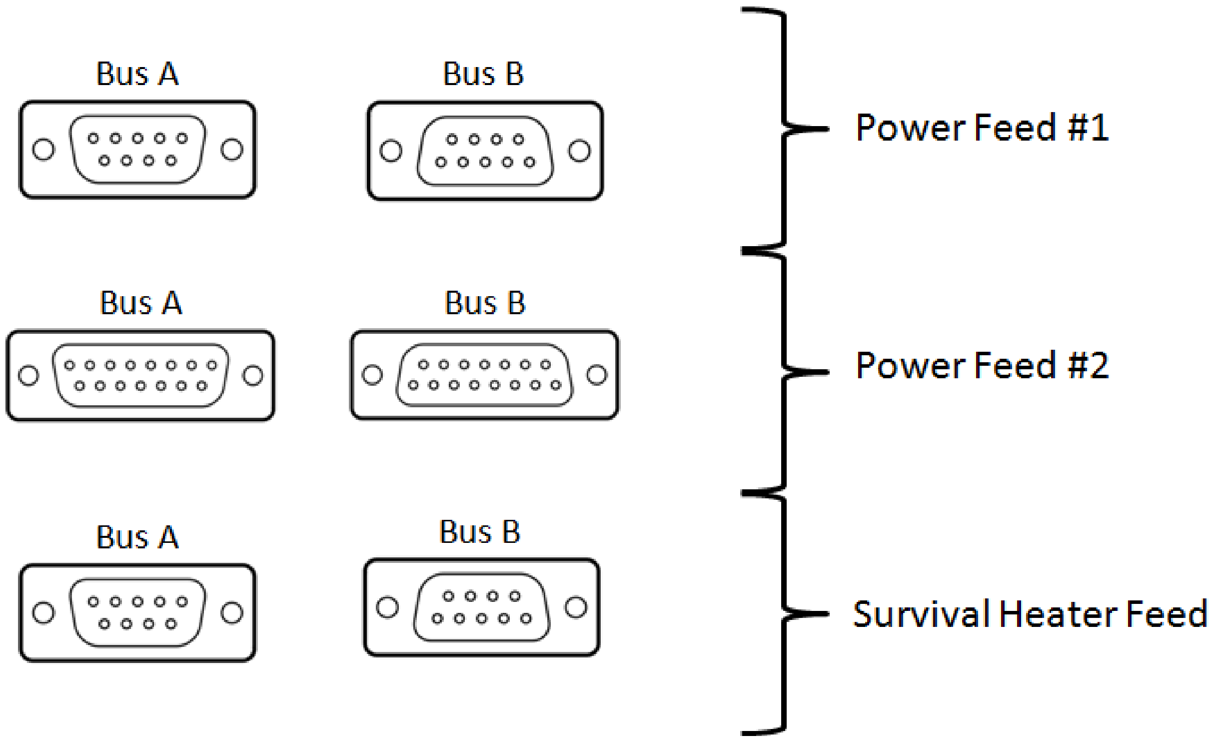


Figure 16-19: Power Connector Keying

16.2.35 Interface Connector Provider

16.2.35.1 *Harness Connectors*

All connectors attached to Spacecraft harnesses should be furnished by the Spacecraft Provider except as noted at Instrument interfaces.

16.2.35.2 *Harness Connectors*

The Spacecraft Provider should be responsible for verifying that the Instrument connectors and Spacecraft connectors are compatible.

16.2.35.3 *Connector Type Selection*

All connectors to be used by the Instrument should be selected from the GSFC Preferred Parts List (PPL).

16.2.35.4 *Connector Type Documentation*

The appropriate Electrical Interface Control Drawing should document the connector types used by the Instrument.

16.2.36 Flight Plugs

16.2.36.1 *Flight Plug Installation*

Flight plugs requiring installation prior to launch should be capable of being installed at the Spacecraft level.

16.2.36.2 *Flight Plug Responsibility*

The Instrument Provider should furnish flight plugs.

16.2.36.3 *Flight Plug Documentation*
The appropriate Electrical Interface Control Drawing should document flight plugs and their locations.

16.2.36.4 *Connector Protective Covers*
The Instrument Provider should furnish captive covers for all instrument connectors which are not mated to harnesses or flight plugs.

16.2.36.5 *Connector Protective Cover Supply*
The Instrument Provider should furnish protective covers for all Instrument-provided connectors.

16.2.36.6 *Non-flight Connector Protective Covers*
Non-flight covers should be marked in red.

16.2.36.7 *Flight Connector Protective Covers*
Covers remaining on the satellite for flight should be flight quality.

16.2.36.8 *Protective Covers for Launch Site Serviceable Connectors*
Connectors requiring servicing at the launch site should have captive covers and hardware.

16.2.36.9 *Connector Protective Cover Flexible Coupling*
Circular connector captive covers should be connected to the component with a flexible coupling to prevent separation of the cover from the component during cover removal.

16.2.36.10 *Connector Protective Cover Screws*
All screws used to secure connector covers to “D" type connectors should be designed to be held captive to the cover.

16.2.37 <u>Test Connectors</u>
16.2.37.1 *Test Connector Location and Types*
The location of test connectors and test coupler ports should maximize unrestricted access, so that connections can be mated and demated without affecting the qualification status of the spacecraft and with minimum risk to the surrounding flight hardware.

16.2.37.2 *Test Connector Location and Types*
Test connector and coupler port test access should be maintained throughout the buildup of the spacecraft and instrument, up to and including the final launch configuration.

16.2.37.3 *Test Connector Accessibility*
Test connectors should be accessible at the integrated Spacecraft level without disassembly.

16.2.37.4 *Test Connector Documentation*
The appropriate Electrical Interface Control Drawing should document test connectors and their locations.

16.2.37.5 *Breakout Boxes*

The Instrument Provider should furnish test tees and breakout boxes.

16.2.38 Buffer Connectors and Connector Savers

16.2.38.1 *Connector Saver Utilization*

Instrument buffer connectors and connector savers should be utilized prior to Spacecraft-level system tests.

16.3 Mechanical Interface Reference Material / Best Practices

16.3.1 Documentation

The MICD should document the Instrument component envelope (including kinematic mounts and MLI) as a "not to exceed" dimension.

16.3.2 Dynamic Envelope or Surfaces

For an instrument with mechanisms that cause a change in the external envelope or external surfaces of the Instrument, the initial and final configurations, as well as the swept volumes, should be contained within the envelope and documented in the MICD.

16.3.3 Instrument Mass

16.3.3.1 *Mass Documentation and Tolerance*

The MICD should document the mass of the Instrument, which should be measured to ±1%.

16.3.4 Center of Mass & Moments of Inertia

16.3.4.1 *Measurement Precision*

The launch and on-orbit center of mass of each Instrument should be measured and reported to ±5 mm, referenced to the Instrument coordinate axes as documented in the MICD.

16.3.4.2 *Measurement Accuracy*

Moments of inertia values should be accurate to within ±10%

16.3.5 Documentation of Dynamic Mechanical Elements

The MICD should document whether the Instrument contains movable masses, expendable masses, or deployables, and their respective inertia variations.

16.3.6 Instrument Mounting

16.3.6.1 *Documentation of Mounting*

The MICD should document the mounting interface, method, and geometry, including dimensions of the holes for mounting hardware.

16.3.6.2 *Instrument Mounting Fasteners*

The Spacecraft Provider should furnish all instrument mounting fasteners.

16.3.6.3 *Kinematic Mounts*

The Instrument Provider should provide all kinematic mounts.

16.3.6.4 *Documentation of Instrument Mounting Hardware*
The MICD should define and document Instrument mounting hardware.

16.3.6.5 *Documentation of Finish and Flatness Guidelines*
The Spacecraft Contractor should specify finish and flatness guidelines for the mounting surfaces and document them in the MICD.

16.3.6.6 *Metric Units*
Mounting fasteners should be metric, in conformance with ISO standards, and meet the guidelines of *Metric Design Parameters for Male Threaded Fasteners* (AIA/NAS ANA0156) and *Federal Standard: Screw-Thread Standards for Federal Services* (FED-STD-H28A).

16.3.7 <u>Mounting Location</u>
16.3.7.1 *Documentation of Instrument Mounting Location*
The MICD should document the mounting location of the Instrument on the Spacecraft.

16.3.8 <u>Drill Template</u>
16.3.8.1 *Drill Template Usage*
Instrument, spacecraft, and test fixture interfaces should be drilled using templates to correctly establish the mounting hole patterns. The MICD should document drill template details and serialization.

16.3.9 <u>Optical Alignment Cubes</u>
The MICD should document the location of all optical alignment cubes on the Instrument.

16.3.10 <u>Access, Handling, and Servicing Accessibility</u>
16.3.10.1 *Installation/Removal*
The Instrument should be capable of being installed or removed in its launch configuration without disturbing the primary payload.

16.3.10.2 *Mechanical Attachment Points*
The Instrument should provide mechanical attachment points that will be used by a handling fixture during integration of the instrument. The MICD should document details of the mechanical attachment points.

16.3.10.3 *Accessibility of Red Tag Items*
All "red tag" items that are planned to be removed or installed from the instrument should be accessible without disassembly of another instrument component.

16.3.11 <u>Test points and Test guidelines</u>
16.3.11.1 *Marking and Documentation*
All test points and I&T interfaces should be clearly marked and designated on the MICD.

16.3.12 <u>Documentation of Orientation Constraints</u>
The MICD should document instrument mechanisms, thermal control, or any exclusions to testing and operations related to orientations.

16.3.13 Identification
All items to be installed prior to or removed following test, and all items to be installed or removed prior to flight, *etc*. should be documented and clearly identified.

16.3.14 Temporary Sensors
The Instrument should accommodate temporarily installed sensors (e.g. acceleration sensors, thermal monitors, etc.) and supporting hardware to support environmental testing.

16.3.15 Venting and Purge
16.3.15.1 *Venting Documentation*
The MICD should document the number, location, size, vent path, and operation time of Instrument vents.

16.3.16 Purge Documentation
Instrument purge guidelines, including type of purge gas, flow rate, gas purity specifications, filter pore size, type of desiccant (if any), and what (if any) interruptions in the purge (and their duration) are tolerable, should be documented upon delivery.

16.3.17 Fields of View Allocation
16.3.17.1 *Documentation of Coordinate System*
The MICD should document the coordinate system.

16.3.18 Mechanisms
16.3.18.1 *Caging During Test and Launch Site Operations*
Instrument mechanisms which require caging and/or uncaging during test and launch site operations should be capable of being caged and uncaged by command and/or by manual actuation of accessible locking/unlocking devices.

16.3.19 Captive Hardware
All items planned to be installed, removed, or replaced during integration should use captive hardware except Instrument mounting hardware and MLI.

16.3.20 Combined Structural Dynamics Analysis Results
The Spacecraft Provider should furnish the combined structural dynamics analysis results to the respective Instrument Provider.

16.3.21 Allowable Angular Momentum
Steady state angular momentum reacted from the Instrument to the Spacecraft should be less than 0.5 N·m·s per axis.

16.3.22 Handling Fixtures
16.3.22.1 *Responsibility for Providing Handling Fixtures*
The Instrument Provider should provide proof tested handling fixtures for each component with mass in excess of 16 kg.

16.3.23 Load Margins
Handling and lifting fixtures should be designed to five (5) times limit load for ultimate and three (3) times limit load for yield. Handling fixtures should be tested to two (2) times working load.

16.3.24 Interface Alignment Cube (IAC) and Alignment Parameters
16.3.24.1 *Instrument Interface Alignment Cube*
The Instrument should contain an optical cube, to be referred to as the Interface Alignment Cube (IAC), nominally aligned with the Instrument Reference Coordinate Frame defined in the MICD.

16.3.24.2 *Interface Alignment Cube Location*
The Instrument Provider should locate the IAC in close proximity to the Instrument mounting interface and should make it visible from at least two orthogonal directions.

16.3.24.3 *Pointing Accuracy, Knowledge, and Stability*
The Instrument Provider should specify the pointing accuracy, knowledge, and stability guidelines in the MICD.

16.3.25 Instrument Boresight
The Instrument Provider is responsible for measuring the alignment angles between the IAC and the Instrument boresight and it should be documented in the MICD.

16.3.26 Kinematic Mount Fracture Guidelines
16.3.26.1 *Fracture Critical Components*
Kinematic mounts should be analyzed, designed, fabricated, and inspected as fracture critical components.

16.3.27 Non-Destructive Evaluation
Kinematic mount flight hardware should be proof loaded and inspected for micro cracks using Non-Destructive Evaluation (NDE) techniques.

16.4 Thermal Interface Reference Material / Best Practices

16.4.1 Heat Transfer Hardware
The use of heat transfer hardware such as heat pipes and high thermal conductivity straps should be considered to make the Instrument design flexible for easier accommodation on the Spacecraft.

16.4.2 Survivability at Very Low Temperature
The Instrument should use components that can survive at very low allowable flight temperature levels to minimize the survival power demands on the Spacecraft.

16.4.3 Implementation of Cooling Function
The Instrument Designer should consider the use of thermoelectric coolers or mechanical coolers if cryogenic temperatures are required for the instrument to ease the restrictions on Instrument radiator orientations.

16.4.4 Implementation of High Thermal Stability

The use of high thermal capacity hardware such as phase change material thermal storage should be considered to meet the high thermal stability requirement.

16.4.5 Survival Power and Temperature

16.4.5.1 *Minimum Turn-On Temperatures*

The Instrument should maintain minimum turn-on temperatures via survival power and heaters controlled by mechanical thermostats.

16.4.5.2 *Surviving Arbitrary Pointing Orientations*

The Instrument should be capable of surviving arbitrary pointing orientations without permanent degradation of performance for a minimum of three (3) orbits with survival power only.

16.4.6 Temperature Maintenance

16.4.6.1 *Documentation of Temperature Limits*

The MICD should document temperature limits for Instrument components during ground test and on-orbit scenarios.

16.4.7 Temperature Monitoring

16.4.7.1 *Documentation of Monitoring Location*

The MICD should document the location of all Instrument temperature sensors.

16.4.7.2 *Temperature Monitoring During OFF Mode*

The Instrument Designer should assume that Spacecraft may not be able to provide temperature monitoring of the payload when it is OFF.

16.4.8 Thermal Hardware

16.4.8.1 *Thermal Control Hardware Documentation*

The MICD should document Instrument thermal control hardware.

16.4.8.2 *Closeout Thermal Hardware Responsibility*

The Spacecraft Provider should be responsible for thermal hardware used to close-out the interfaces between the Instrument and Spacecraft, such as close-out MLI.

16.4.8.3 *Survival Heater Responsibility*

The Instrument Provider should provide and install Instrument survival heaters.

16.4.8.4 *Mechanical Thermostats*

The Instrument should control Instrument survival heaters via mechanical thermostats.

16.4.8.5 *Survival Heater Power*

Instrument survival heaters should obtain survival heater power from the Spacecraft survival heater bus and should be limited to no more than 20% of the average Instrument power.

16.4.8.6 *Survival Heater Documentation*

The MICD should document survival heater characteristics, as well as, mounting locations and attachment details.

16.4.9 Thermal Analytical Models

16.4.10 Comparison of Detailed Thermal Analysis and Reduce Node Model

The Instrument Provider should furnish the Spacecraft Provider with a written report documenting the results of the detailed thermal analysis and the comparison of results to the reduced node model, including a high-level energy balance and heat flow map.

16.4.11 Launch Thermal Environment

The Instrument should be designed for the launch heating profile as specified in the MICD.

16.5 Environmental Reference Material / Best Practices

16.5.1 Temporary Loss of Function or Loss of Data

Temporary loss of function or loss of data is permitted, provided that the loss does not compromise instrument health and full performance can be recovered rapidly.

16.5.2 Restoration of Normal Operation and Function

To minimize loss of data, normal operation and function should be restored via internal correction methods without external intervention.

16.5.3 Irreversible Actions

Irreversible actions should not be permitted. The hardware design should have no parts which experience radiation induced latch-up to an effective LET of 75 MeV/mg/cm^2 and a fluence of 107 ions/cm^2.

16.5.4 Electromagnetic Interference & Compatibility Environment

16.5.4.1 *Magnetic Field Emissions*

The instrument should be designed with the magnetic field emissions in accordance with Table 16-7.

Table 16-7 : Magnetic Field Emissions (RE04)

Magnetic Field Emissions
AC Magnetic Field Emissions: The radiated AC magnetic field levels from the instrument should be limited to 60 dB above 1 pT between 20 Hz and 50 kHz using the RE04 test method of MIL-STD-462. The measurement bandwidth should be 10 Hz between 20 Hz and 200 Hz; 100 Hz between 200 Hz and 20kHz; and 1 kHz between 20 kHz and 50 kHz.
DC Magnetic Field Emissions: The residual magnetic dipole moment of the instrument should be less than 0.5 A·m^2.

16.5.4.2 *Magnetic Field Susceptibility*

The instrument should be designed in accordance with the magnetic field susceptibility constraints in Table 16-8.

Table 16-8 : Magnetic Field Susceptibility (RS01)

Magnetic Field Susceptibility
<u>AC Magnetic Field Susceptibility</u>: The instrument should operate within specification when subjected to an AC magnetic field level of 124 dB above 1 pT between 30 Hz and 200 kHz. The RS01 test method of MIL-STD-462 should be used for measurement.
<u>DC Magnetic Field Susceptibility</u>: The instrument should operate within specification when subjected to an ambient magnetic field consisting of the Earth's field (15 to 50 µT), the fields generated by neighboring instruments (3 µT maximum), and the field produced by the spacecraft magnetic torquers (1000 µT maximum).

16.5.4.3 *Conducted Emissions (CE102)*

The electromagnetic emissions from the instrument, conducted on the power lead(s) and/or the power return lead(s), should not exceed the levels specified in requirement CE102 of MIL-STD-461F based upon the instrument source voltage (see Figure 16-20.)

Verification testing of requirement CE102, as defined in MIL-STD-461F, should be performed on all instrument power leads and power return leads that obtain power from sources that are not a part of the instrument.

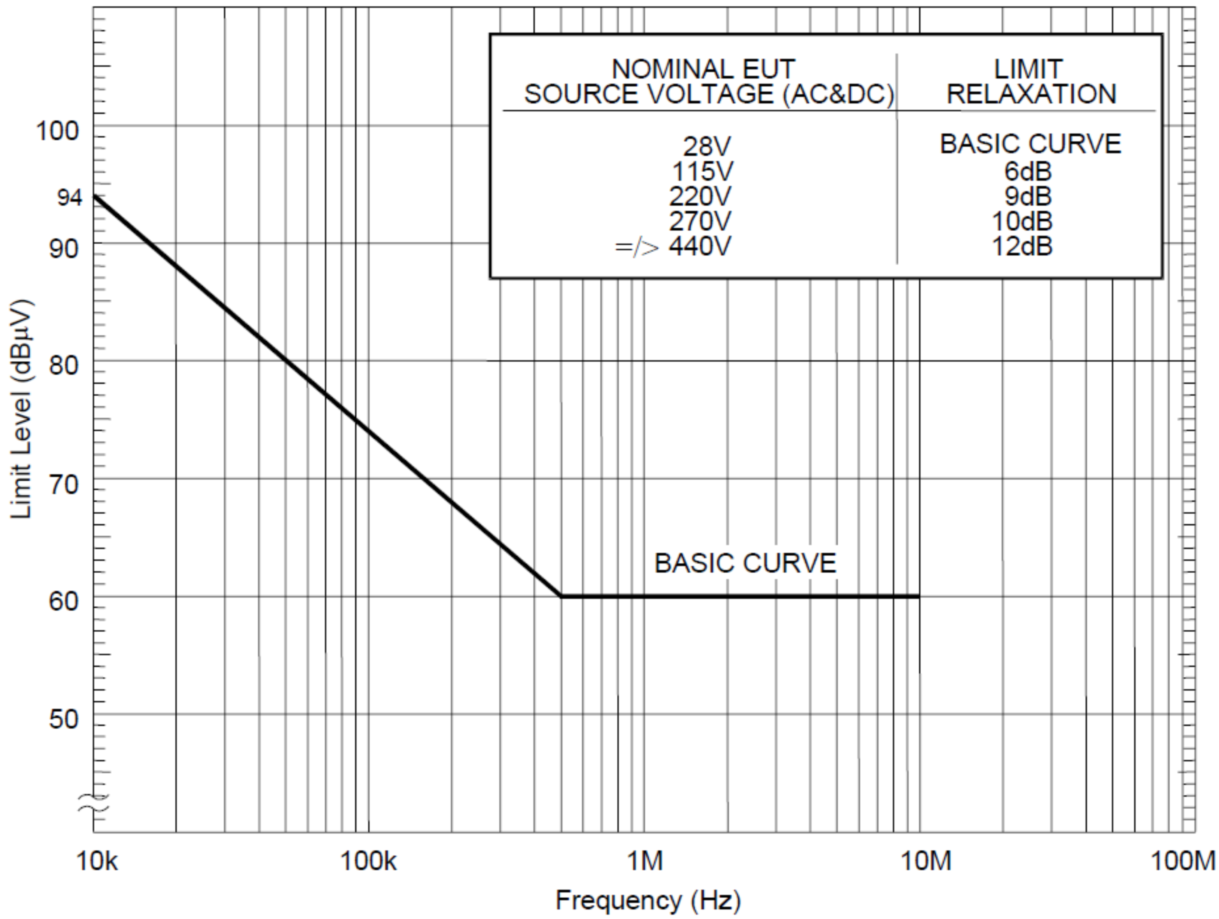


Figure 16-20: Conducted Emissions Environment (Reference)

16.5.4.4 *Radiated Emissions (RE102)*

The electric field emissions radiated from the instrument should not exceed the levels specified in the portion of requirement RE102 of MIL-STD-461F defined for spacecraft (see Figure 16-21).

MIL-STD-461F defines verification testing of requirement RE102, which is applicable to all instrument enclosures, interconnecting cables and antennas designed to be permanently mounted to the instrument (receivers and transmitters in standby mode). RE102 does not apply at the transmitter fundamental frequencies and the necessary occupied bandwidth of the signal. Requirement RE102 is applicable from 10 kHz to 18 GHz[2]. Above 30 MHz, the limits should be met for both horizontally and vertically polarized fields.

[2] Testing is required up to the greater of 1 GHz or 10 times the highest intentionally generated frequency within the instrument. Measurements beyond 18 GHz are not required.

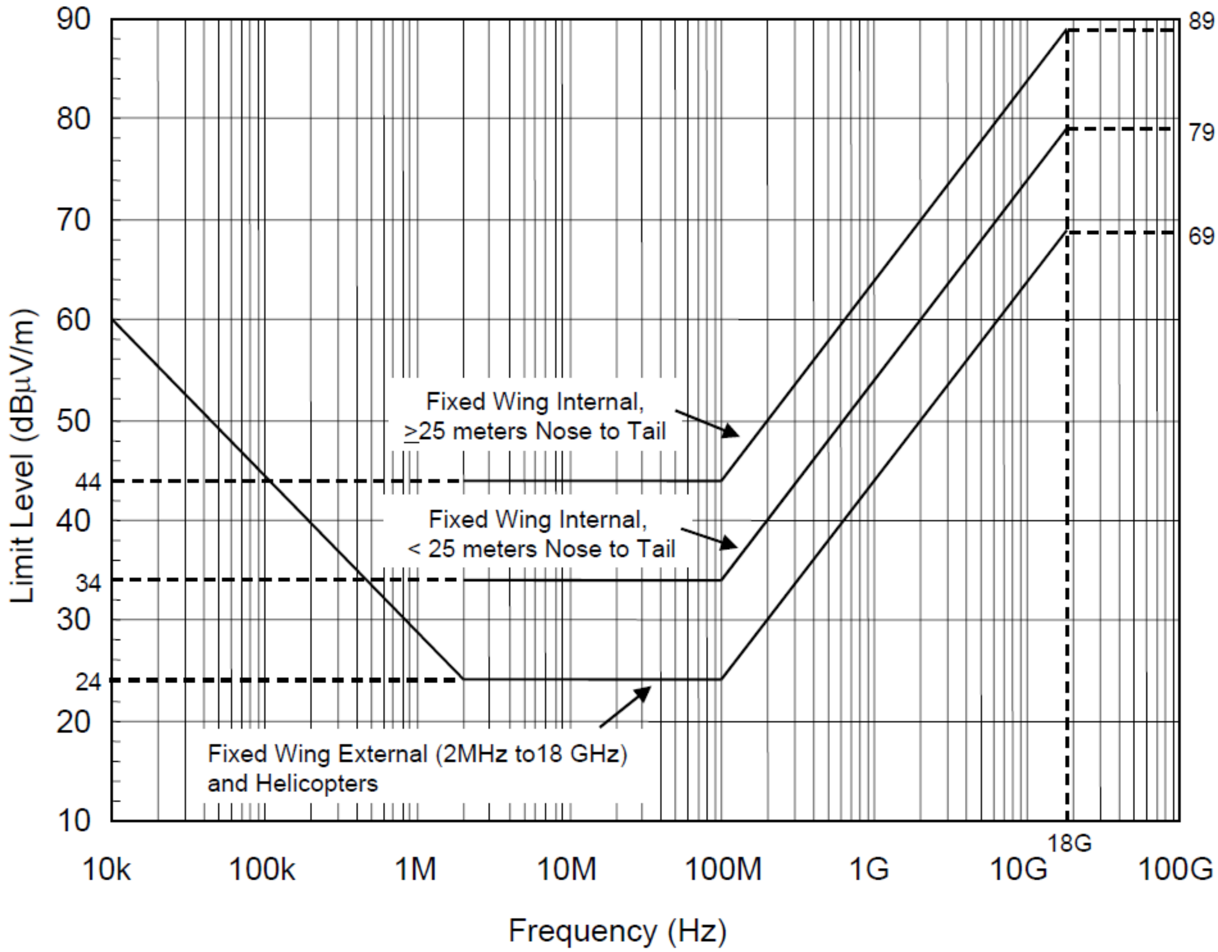


Figure 16-21: Radiated Emissions Environment (Reference)

16.5.4.5 *Conducted Susceptibility (CS101)*

The instrument should not exhibit any malfunction, degradation of performance, or deviation from specified indications, beyond the tolerances indicated in the instrument specification, when subjected to requirement CS101 of MIL-STD-461F (see Figure 16-22 and Figure 16-23).

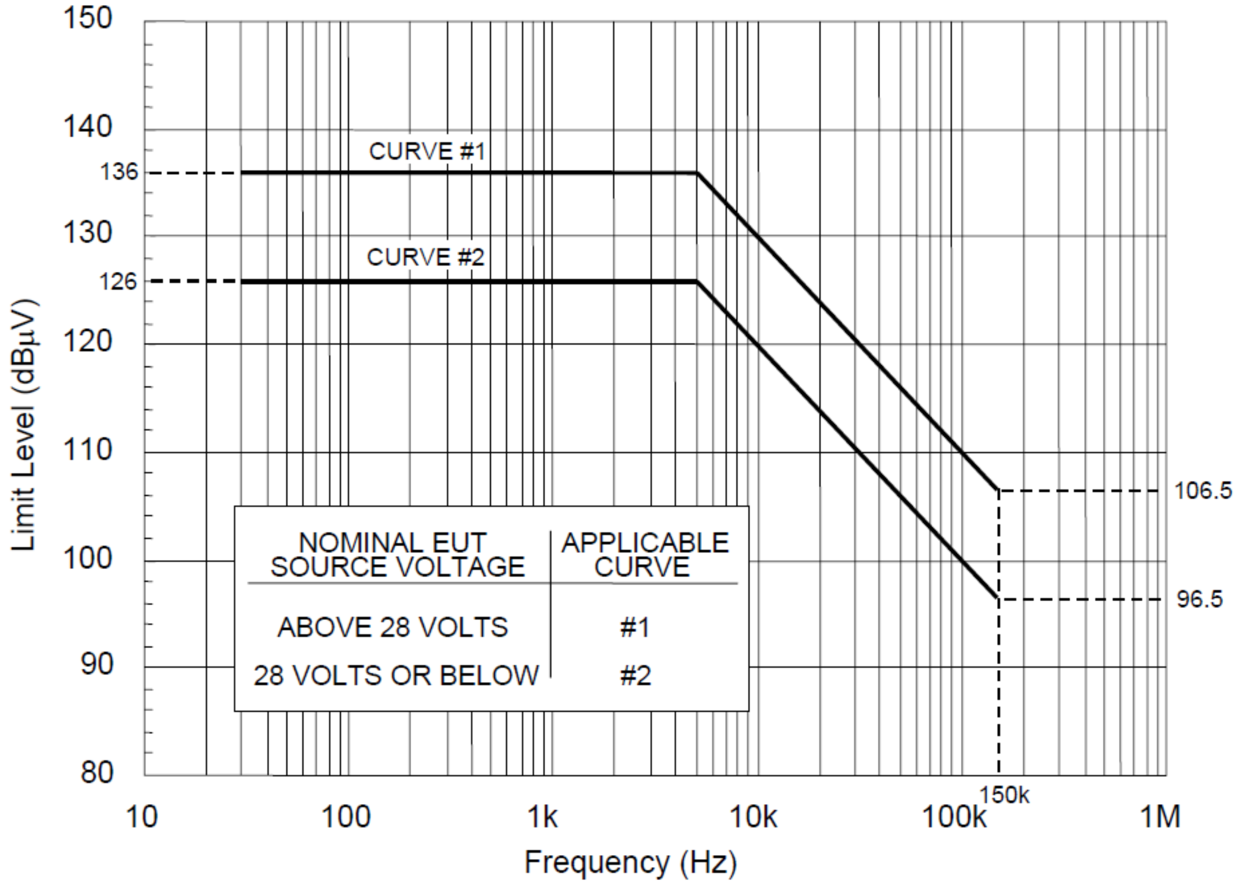


Figure 16-22: Conducted Susceptibility Voltage Limits (Reference)

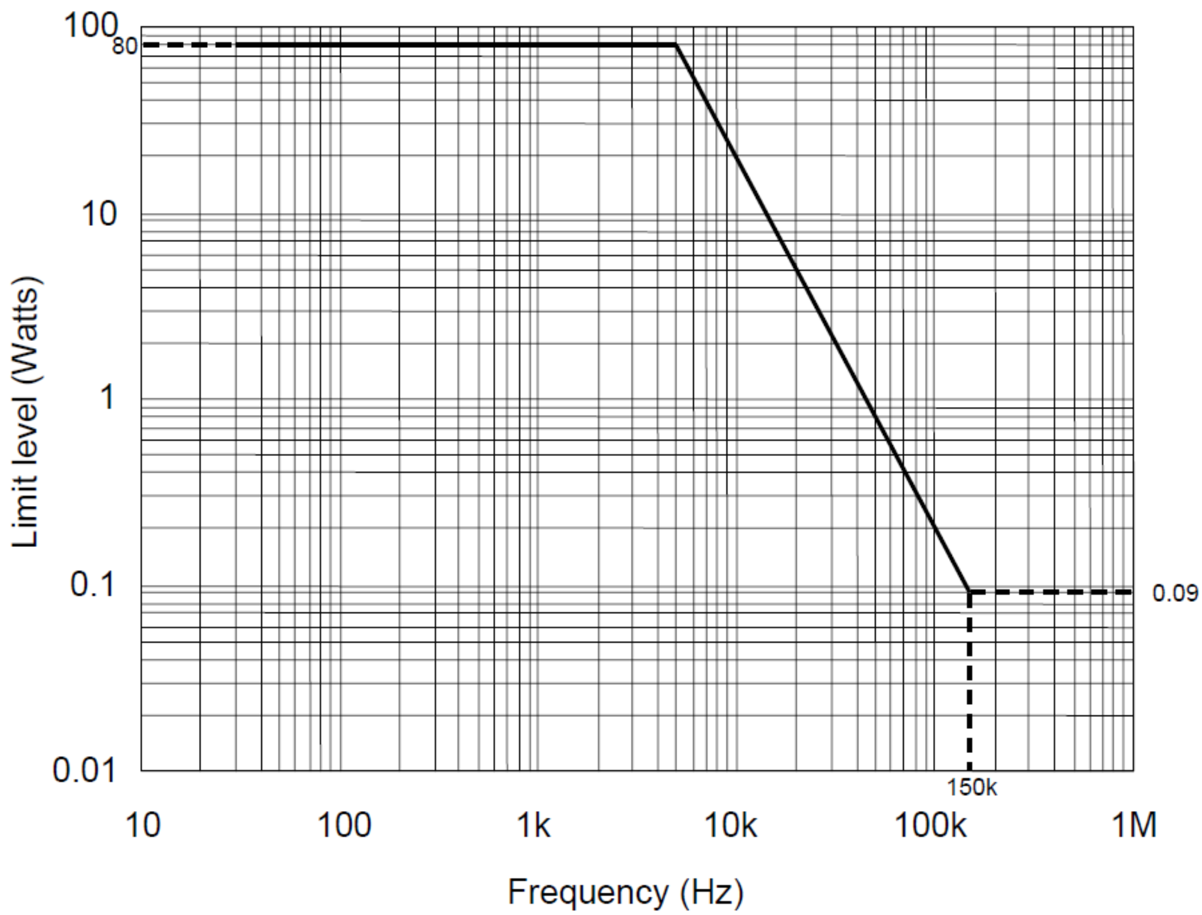


Figure 16-23: Conducted Susceptibility Power Limits (Reference)

MIL-STD-461F defines verification testing of requirement CS101, which is applicable to instrument AC, limited to current draws less than or equal to 100 A per phase, and DC input power leads, not including returns. If the instrument is DC operated, this requirement is applicable over the frequency range of 30 Hz to 150 kHz. If the instrument is AC operated, this requirement is applicable starting from the second harmonic of the instrument power frequency and extending to 150 kHz.

16.5.4.6 *Conducted Susceptibility (CS114)*

The instrument should not exhibit any malfunction, degradation of performance, or deviation from specified indications beyond the tolerances indicated in the instrument specification, when subjected to requirement CS114 of MIL-STD-461F. The injection probe drive level must be pre-calibrated to the current limits defined in curve #3 of Figure 16-24 and modulated as specified in requirement CS114.

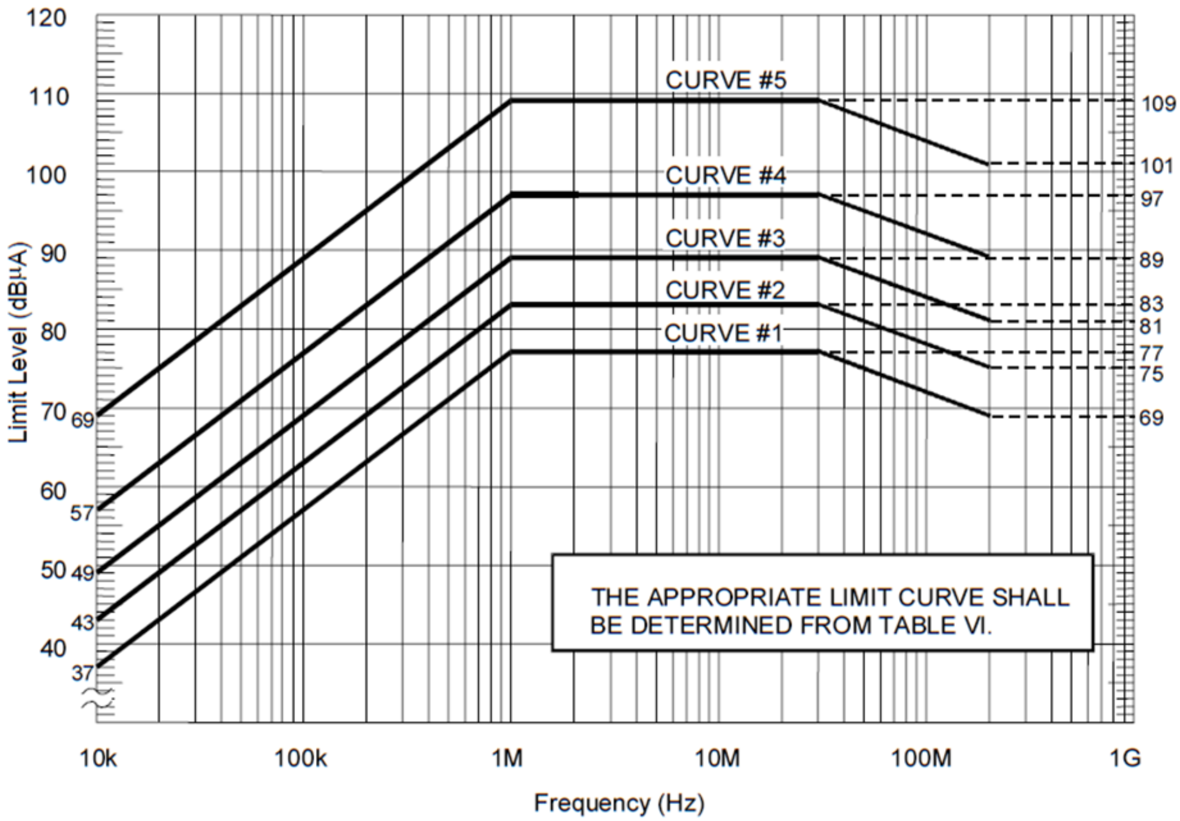


Figure 16-24: Conducted Susceptibility Current Limits (Reference)

Requirement CS114 may also be satisfied if the instrument is not susceptible at forward power levels sensed by the directional coupler that are below those determined during calibration provided that the actual current induced in the cable under test is 6 dB or greater than the calibration limit.

MIL-STD-461F defines verification testing of requirement CS114, which is applicable for all interconnecting cables, including power cables but is not applicable for coaxial cables to antenna ports of antenna-connected receivers.

16.5.4.7 *Conducted Susceptibility (CS115)*

The instrument should not exhibit any malfunction, degradation of performance, or deviation from specified indications, beyond the tolerances indicated in the instrument specification, when subjected to a pre-calibrated signal having rise and fall times, pulse width, and amplitude as specified in requirement CS115 of MIL-STD-461F (see Figure 16-25).

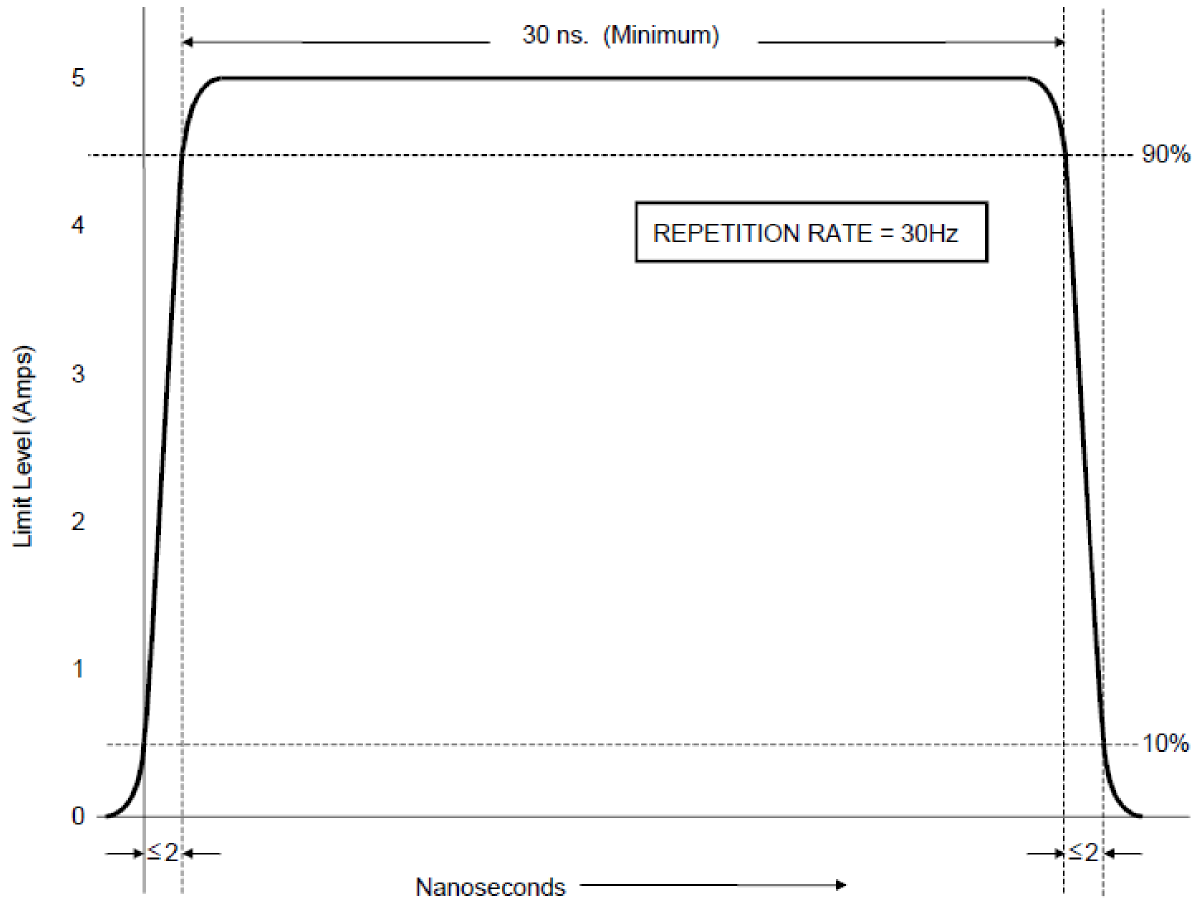


Figure 16-25: Conducted Susceptibility Current Limits (Reference)

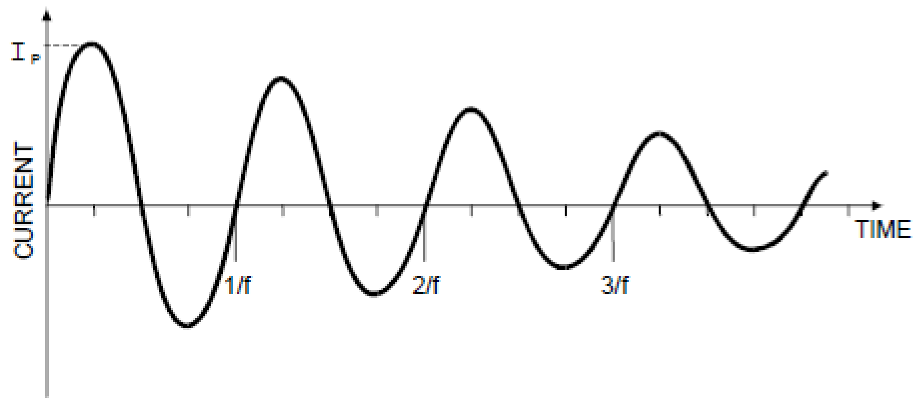
MIL-STD-461F defines verification testing of requirement CS115, which is applicable to all space and ground system interconnecting cables, including power cables.

16.5.4.8 *Conducted Susceptibility (CS116)*

The instrument should not exhibit any malfunction, degradation of performance, or deviation from specified indications, beyond the tolerances indicated in the instrument specification, when subjected to requirement CS116 of MIL-STD-461F. Figure 16-26 depicts the test signal waveform, and Figure 16-27 specifies the maximum current. The current limit is applicable across the entire specified frequency range.

MIL-STD-461F defines Verification testing of requirement CS116, which is applicable to all interconnecting cables, including power cables, and individual high side power leads. Power returns and neutrals need not be tested individually.

As a minimum, compliance should be demonstrated at 0.01, 0.1, 1, 10, 30, and 100 MHz. Compliance should also be demonstrated at other frequencies known to be critical to the equipment installation, such as platform resonances. The test signal repetition rate should be no greater than one pulse per second and no less than one pulse every two seconds. The pulses should be applied for a period of five minutes.



NOTES: 1. Normalized waveform: $e^{-(\pi f t)/Q} \sin(2\pi f t)$

Where:
f = Frequency (Hz)
t = Time (sec)
Q = Damping factor, 15 ± 5

2. Damping factor (Q) shall be determined as follows:

$$Q = \frac{\pi(N - 1)}{\ln(I_P/I_N)}$$

Where:
Q = Damping factor
N = Cycle number (i.e. N = 2, 3, 4, 5,...)
I_P = Peak current at 1st cycle
I_N = Peak current at cycle closest to 50% decay
ln = Natural log

3. I_P as specified in Figure CS116-2

Figure 16-26: Conducted Susceptibility Waveform (Reference)

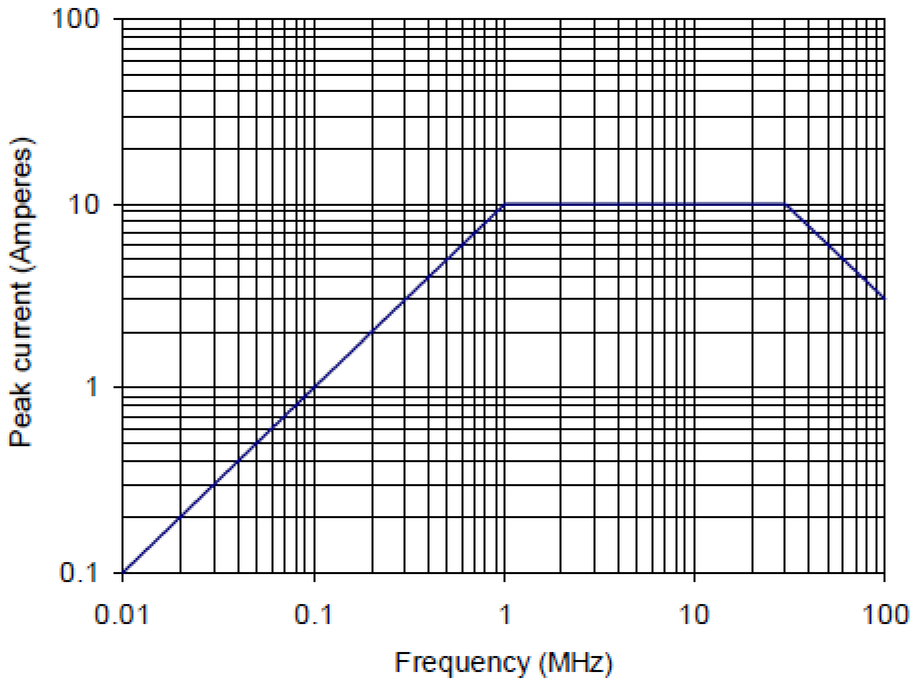


Figure 16-27: Conducted Susceptibility Current Limits (Reference)

16.5.4.9 *Radiated Susceptibility (RS103)*

The instrument should not exhibit any malfunction, degradation of performance, or deviation from specified indications, beyond the tolerances indicated in the instrument specification, when subjected to requirement RS103 of MIL-STD-461F (a radiated electric field of 20 Volts/meter modulated between 2 MHz and 40 GHz).

MIL-STD-461F defines verification testing of requirement RS103, which is applicable to instrument enclosures and all interconnecting cables. For modulations ≤ 30 MHz, the requirement should be met for vertically polarized fields. For modulation > 30 MHz, the requirement should be met for both horizontally and vertically polarized fields. Circular polarized fields are not acceptable. There is no requirement at the tuned frequency of antenna-connected receivers.