National Aeronautics and
Space Administration

Document Number: CII-CI-XXXX
Version: Rev A Draft

Common Instrument Interface Project

Hosted Payload Guidelines Document

Earth System Science Pathfinder Program Office
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3 **Change Log**

Version	Date	Section Affected	Description
Rev A	11-26-2012	All	Incorporated GEO guidelines
Rev A	11-26-2012	All	Reorganized document to delineate Level 1 guidelines, Level 2 guidelines, and Best Practices
Rev A	11-26-2012	All	Added Hosted Payload Concept of Operations
Rev A	02-15-2013	All	Updated document based upon stakeholder feedback from December 2012 CII Working Group Meeting

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1.0 OVERVIEW

1.1 Introduction

These Common Instrument Interface (CII) Hosted Payloads Guidelines provide a prospective Instrument Developer with technical recommendations to help them design an Instrument that may be flown as a hosted payload either in Low Earth Orbit (LEO) or Geostationary Earth Orbit (GEO).

NASA Earth Science has implemented its Earth Venture Instrument (EVI) line of missions using a hosted payload model. Therefore, these guidelines primarily support stakeholders involved in NASA's EVI suite of investigations.

NASA competitively selects Principal Investigator (PI)-led EVI investigations via solicitations that "call for developing instruments for participation on a NASA-arranged spaceflight mission of opportunity to conduct innovative, integrated, hypothesis or scientific question-driven approaches to pressing Earth system science issues." The deliverables of a selected investigation includes "a flight qualified spaceflight instrument or instrument package ready for integration to a spacecraft, technical support for integration onto a NASA-determined spacecraft, and on-orbit operation of the instrument and delivery of science quality data." Prospective PI's propose their Instrument "without a firm identification of the spacecraft to accommodate it," and NASA deploys the selected Instrument on an existing planned spacecraft (Host Spacecraft). [1]

This guideline document focuses on the technical aspects of flying an Earth Science Instrument on a Hosted Payload Opportunity (HPO). Because of the nature of the EVI acquisition strategy, Instrument Developers and Spacecraft Manufacturers proceed along the early stages of their respective product lifecycles independently. By vetting these technical parameters with space industry stakeholders, the CII team hopes to ensure maximum compatibility with the Earth-orbiting spacecraft market, leading to an increased likelihood of a successful Instrument to HPO pairing.

Instrument Developers are not required to comply with these guidelines. These guidelines are not met to replace Instrument Developer collaboration with Spacecraft Manufacturers, rather to provide familiarity of Spacecraft interfaces and accommodations in order to assist with such collaboration. Instrument parameters that exceed values specified in this guidance may very well be accommodated with additional resources that offset the impact to existing HPO designs (*e.g.,* investments enhancing Instrument capability) or that propose to enable compatibility after minor alterations to spacecraft performance (*e.g.,* investments enhancing Spacecraft capability). It is ultimately the responsibility of the Instrument Provider to investigate such cost-benefit considerations during proposal development.

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[1] "Earth Venture Instrument-1," from Program Element Appendix (PEA) J of the Second Stand Alone Missions of Opportunity Notice (SALMON-2), 2012.

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1.2 Nomenclature and Definitions

The verb "should" denotes a recommendation. "Shall" denotes a requirement. "Will" denotes an expected future event.

Hosted Payload: a payload manifested on a spacecraft bus flying on a primary space mission.

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

Instrument: the hosted payload of record that these guidelines describe.

Instrument Developer: the organization responsible for developing and building the Instrument itself.

Host Spacecraft: the Hosted Payload Opportunity spacecraft bus of record that these guidelines describe.

Host Spacecraft Manufacturer: the organization responsible for manufacturing the Host Spacecraft and the primary commercial payloads.

Satellite Operator: the organization responsible for on-orbit and ground operations throughout the Host Spacecraft's lifetime.

Systems Integrator: the organization responsible for integrating the Instrument and Host Spacecraft.

1.3 Methodology

The content of this document is aggregated from several sources. The CII team used personal engineering experience, publicly available information, and privately held information shared by industry to define the primary technical components structuring this document and to establish its content. The CII team leveraged stakeholder feedback and numerous peer review workshops to guide efforts seeking to establish appropriate breadth and depth of the source material for the guidelines document. In order to increase the likelihood that a guideline-compliant Instrument design would technically fit within the accommodation space of an HPO, the CII team used an "all-satisfy" strategy. Specifically, for each technical performance measure, guidance is generally prescribed by the most restrictive value from the set of likely spacecraft known to operate in both the LEO and GEO domains. This strategy was again generally utilized to characterize environments, whereby the most strenuous environment expected in both the LEO and GEO domains inform best practices. Where considered necessary, the CII team based environmental guidance on independent modeling of particular low Earth orbits that are commonly considered advantageous in supporting Earth science measurements.

This methodology also allows for the sanitization of industry proprietary data. The set of expected LEO spacecraft is based upon the Rapid Spacecraft Development Office Catalog (http://rsdo.gsfc.nasa.gov/catalog.html), tempered by CII analyses of NASA databases and Communities of Practice. Smaller spacecraft (including microsatellites or secondary platforms) are not precluded from host consideration. The set of expected GEO spacecraft is based upon industry responses to the *Request for Information for Geostationary Earth Orbit Hosted Payload Opportunities.*

One limitation of the "all-satisfy" strategy is that it constrains all instrument accommodation parameters to a greater degree than should be expected once the Instrument is paired with a Host Spacecraft. One size does not fit all in Hosted Payloads, especially in the GEO domain where the bus sizes vary among and within Spacecraft Manufacturers. Additionally, because a Spacecraft Manufacturer tailors its bus design to each Satellite Operator's requirements, Instrument Developers may be able to negotiate an agreement for the Spacecraft Manufacturer to supply or for the Satellite Operator to require a larger bus or upgraded spacecraft performance than originally specified for the Satellite Operator. This enables the Host Spacecraft to accommodate more demanding Instrument requirements, given the application of enough resources. Because the Instrument to Host Spacecraft pairing occurs in the vicinity of Key Decision Point (KDP) C, certain knowledge of these available accommodation resources will be delayed well into the Instrument's development timeline.

1.4 **Interpretation**

The content of this document represents recommendations, not requirements. These recommendations aid Instrument Developers proposing to EVI AO's by documenting the CII team's analysis of the interfaces and resource demands most likely to be accommodated on LEO and GEO HPO's. While the EVI–1 AO references the CII guidelines and HPO database as "activities that document … the types of opportunities that exist and the current interfaces and constraints that exist for each potential platform," it does not state that compliance with the CII guidelines is mandatory.[2] The CII Team's expects that future EVI AO's will use the same model. The CII Team has limited the depth of guidelines to strike a balance between providing enough technical information to add value to a Pre-Phase A (Concept Studies) project and not overly constraining the Instrument design. This allows for a design sufficiently fungible to adapt to expected HPO's and limits any (incorrectly inferred) compliance burdens.

While this document focuses on the technical aspects of hosted payloads, it is noteworthy that programmatic and market-based factors are likely more critical to the success of a hosted payload project than technical factors. When paired with commercial satellites, NASA can take advantage of the commercial space industries best practices and profit incentives to fully realize the benefits of hosted payloads. Because an EVI Instrument would neither be the primary payload or financial contributor on the Host Spacecraft, NASA may relinquish some of the oversight and decision rights it traditionally exerts in a dedicated mission. This leads to the "Do No Harm" concept explained in the Level 1 Design Guidelines. With this exception, programmatic and business aspects of hosted payloads are outside the scope of this document.

[2] "Earth Venture Instrument-1," from Program Element Appendix (PEA) J of the Second Stand Alone Missions of Opportunity Notice (SALMON-2), 2012.

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1.5 **Scope**

This document's scope comprises five primary technical components of the Instrument to Host Spacecraft pairing: interface, accommodations, best practices, assumptions, and negotiated parameters. Figure 1-1 uses color to identify the scope: colored components are in scope; black components are out of scope.

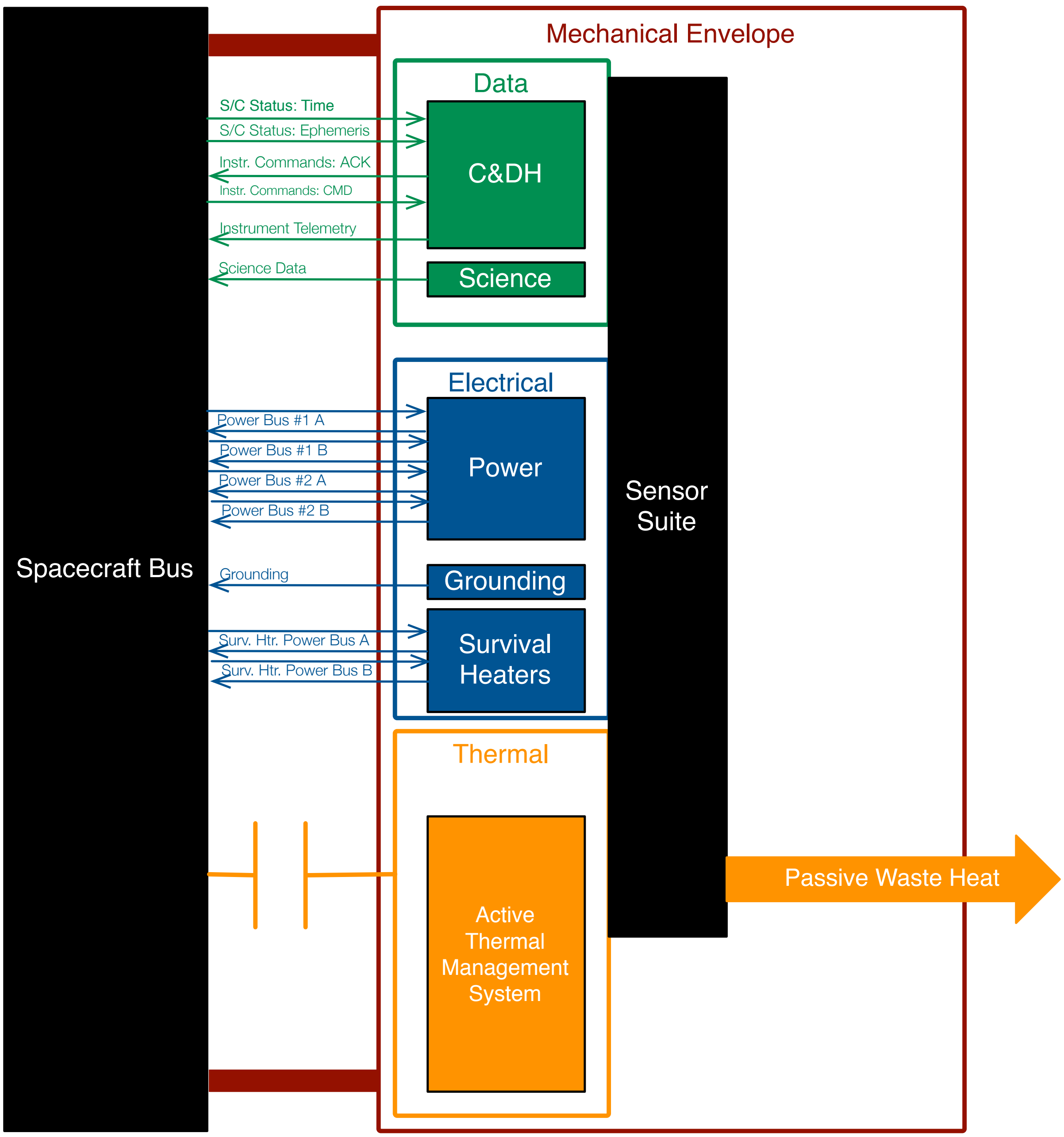

Figure 1-1: CII Scope

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Interface guidelines describe the direct interactions between the Instrument and Host Spacecraft, such as physical connections and transfer protocols. Accommodation guidelines describe the constraints on the resources and services the Instrument is expected to draw upon from the Host Spacecraft, including size, mass, power, and transmission rates. While guidelines are not requirements—using the verb "should" instead of "shall"—they try to follow the rules of writing proper requirements, including providing rationale and maintaining traceability to higher level guidelines. This document provides two hierarchical levels of guidelines, because suggesting more specific technical details in the context of a hosted payload with not yet identified stakeholders is not credible.

The best practices capture additional technical information, less prescriptive than the guidelines, which an Instrument Developer might still find useful.

Assumptions are generally expectations of the characteristics and behavior of the Host Spacecraft and/or Host Spacecraft Manufacturer. Since Instrument requirement definition and design will likely happen prior to identification of the Host Spacecraft, these assumptions help bind the trade space.

Negotiated parameters reflect the effect of the Host Spacecraft and Instrument beginning development simultaneously and independently—some parameters will not be resolved prior to the Host Spacecraft to Instrument pairing. This document uses an Interface Control Document (ICD) construct as the means to record agreements reached among the Instrument Developer, Host Spacecraft Manufacturer, Launch Vehicle Provider, and Satellite Owner.

This document's recommendations cover both the LEO and GEO domains. If a guideline or best practice is specific to one of the domains, it begins with either the [LEO] or [GEO] prefix. Guidelines or best practices without prefixes apply to both domains.

1.6 Revisioning

The Earth System Science Pathfinder (ESSP) Program Office released the Baseline version of CII guideline document, which only addressed interfaces for LEO platforms, in November 2011 in preparation for the EVI–1 AO. This Revision A version precedes the EVI–2 AO, providing more explicit definition of guideline scope and technical components, incorporating technical content for the GEO domain, and reducing design constraints on the Instrument Developer.

The CII Team plans to release updated guidelines preceding each future EVI AO release. This forward approach will ensure this document's guidance reflects current technical interface capabilities of commercial spacecraft manufacturers and maintains cognizance of industry-wide design practices resulting from technological advances (*e.g.* xenon ion propulsion).

1.7 Interaction with Other Agencies Involved with Hosted Payloads

A measure of success for these guidelines is that they will have a broad acceptance among different communities and agencies. The Air Force Space and Missile Systems Center recently stood up a Hosted Payload Office (SMC/XRFH) to evaluate HPO's as a distributed, resilient option within operational architectures. The European Space Agency's (ESA) Future Missions

Division of their Earth Observation Program Directorate is also formulating a hosted payload concept for their future missions. Both organizations are currently developing hosted payload standards, although one important note is that the SMC and ESA elements will be prescriptive requirements as opposed to the CII recommended guidelines.

The CII team has been working very closely with ESA over the past couple of years on a unified set of guidelines for electrical power and data interfaces in the LEO domain. Due to different sets of common practices in the American and European space industries, a small number of technical differences exist between the CII and ESA guidelines.

Similarly, the CII team has been collaborating with SMC/XRFH to minimize the differences in CII top-level guidelines and SMC requirements. The SMC requirements are currently in development. Future versions of the CII document will summarize the differences with the SMC requirements once they are finalized.

Appendix H summarizes these differences in tabular form.

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2.0 LEVEL 1 DESIGN GUIDELINES

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

2.1 **Assumptions**

2.1.1 Hosted Payload: The Host Spacecraft will have a primary mission different than that of the Instrument.

2.1.2 [GEO] Nominal Orbit: The Host Spacecraft will operate in GEO with an altitude of approximately 35786 kilometers and eccentricity and inclination of approximately zero.

2.1.3 [LEO] Nominal Orbit: The Host Spacecraft will operate in LEO with an altitude between 350 and 2000 kilometers with eccentricity less than 1 and inclination between zero and 180°, inclusive.

2.1.4 Responsibility for Integration: The Host Spacecraft Manufacturer will integrate the Instrument onto the Host Spacecraft with support from the Instrument Developer.

2.2 **Guidelines**

2.2.1 Hosted Payload Worldview

The Instrument should prevent itself or any of its components from damaging or otherwise degrading the mission performance of the Host Spacecraft or any other payloads.

Rationale: The most important constraint on a hosted payload is to "do no harm" to the Host Spacecraft or other payloads. The Satellite Operator will have the authority to remove power or otherwise terminate the Instrument should either the Host Spacecraft's available services degrade or the Instrument pose a threat to the rest of the Spacecraft. This guideline applies over the period beginning at the initiation of Instrument integration to the Host Spacecraft and ending at the completion of the disposal of the Host Spacecraft. It is important to note that most GEO communications satellites have a nominal mission lifetime in excess of 15 years or more while a hosted payload Instrument nominal lifetime is on the order of five years.

2.2.2 Data Interface

[LEO] The Instrument-to-Host Spacecraft data interfaces should use RS-422, SpaceWire, or MIL-STD-1553.

Rationale: RSS-422, SpaceWire, and MIL-STD-1553 are commonly accepted spacecraft data interfaces.

[GEO] The Instrument should use MIL-STD-1553 as the command and telemetry data interface with the Host spacecraft.

Rationale: The use of MIL-STD-1553 for command and telemetry is nearly universal across GEO spacecraft buses.

[GEO] The Instrument should send science data directly to its transponder via an RS-422, LVDS, or SpaceWire interface.

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Rationale: The use of RS-422, LVDS, or SpaceWire directly to a transponder for high-volume payload data is a common practice on GEO spacecraft buses.

2.2.3 Data Accommodation

[LEO] The Instrument should transmit less than 10 Mbps of data on average to the Host Spacecraft. Data may be transmitted periodically in bursts of up to 100 Mbps.

Rationale: CII analysis of the NICM Database (see Appendix E) shows 10 Mbps to be the upper bound for instruments likely to find rides as LEO hosted payloads. Many spacecraft data buses are run at signaling rates than can accommodate more than 10 Mbps. While this additional capacity is often used to share bandwidth among multiple payloads, it may also be used for periodic burst transmission when negotiated with the Host Spacecraft Providers and/or Operators. When sizing Instrument data volume, two considerations are key: 1) The Instrument should not assume the Host Spacecraft will provide any data storage (see guideline 4.3.1), and 2) LEO downlink data rates vary considerably depending upon the antenna frequencies employed (*e.g.* S-Band is limited to 2 Mbps while X-Band and Ka-Band may accommodate 100 Mbps or more).

[GEO] The Instrument should utilize less than 500 bps of MIL-STD-1553 bus bandwidth when communicating with the Host Spacecraft.

Rationale: The MIL-STD-1553 maximum 1 Mbps data rate is a shared resource. Most spacecraft buses provide between 250 bps and 2 kbps for commanding and up to 4 kbps for telemetry for all instruments and components on the spacecraft bus. Telemetry that is not critical to the health and safety of either the Instrument or Host Spacecraft does not need to be monitored by the Satellite Operator and therefore may be multiplexed with Instrument science data.

[GEO] The Instrument should transmit less than 60 Mbps of science data to its transponder.

Rationale: Transponder bandwidth is a function of lease cost and hardware capability. Data rates in the range of 60-80 Mbps for a single transponder are common. Higher data rates can be achieved with multiple transponders (at an increased cost).

2.2.4 Electrical Power System Interface

The Instrument should electrically ground to a single point on the Host Spacecraft.

Rationale: The Instrument Electrical Power System (EPS) should ground in a way that reduces the potential to introduce stray currents or ground loop currents into the Instrument, Host Spacecraft, or other payloads.

2.2.5 Electrical Power System Accommodation

[LEO] The Instrument EPS should draw less than or equal to 100W, averaged over the orbit, from the Host Spacecraft.

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Rationale: CII analysis of the NICM Database (see Appendix E) shows 100 W to be the upper bound for instruments likely to find rides as LEO hosted payloads.

[LEO] The Instrument EPS should accept an unregulated input voltage of 28 ± 6 VDC.

Rationale: The EPS architecture is consistent across LEO spacecraft bus manufacturers with the available nominal voltage being 28 Volts Direct Current (VDC) in an unregulated (sun regulated) configuration.

[GEO] The Instrument should draw less than or equal to 300W of electrical power from the Host Spacecraft.

Rationale: The Host Spacecraft available electrical power varies significantly both by manufacturer and by spacecraft bus configuration. 300 Watts represents a power level that all of the Primary Manufacturers'[3] buses can accommodate, and requiring a power level less than this increases the likelihood of funding a suitable Host Spacecraft.

[GEO] The Instrument EPS should accept a regulated input voltage of 28 ± 3 VDC.

Rationale: Host Spacecraft bus voltages vary by manufacturer, who design electrical systems with the following nominal voltages: 28, 36, 50, 70, and 100 VDC. To maximize both voltage conversion efficiency and available hosting opportunities, the Instrument should accept the lowest nominal voltage provided, which is 28 VDC.

Note: this guideline may be superseded by Instruments that have payload-specific voltage or power requirements or by "resistance only" power circuits (see below).

[GEO] The Instrument payload primary heater circuit(s), survival heater circuit(s) and other "resistance only" power circuits that are separable subsystems of the Instrument payload EPS should accommodate the Host Spacecraft bus nominal regulated voltage and voltage tolerance.

Rationale: Host Spacecraft bus voltages vary by manufacturer, who design electrical systems with the following nominal voltages: 28, 36, 50, 70, and 100 VDC. To minimize the amount of power required to be converted to an input voltage of 28 ± 3 VDC and to maximize the available hosting opportunities, an Instrument Developer should design "resistance only" power loads to accept the spacecraft bus nominal voltage.

2.2.6 Thermal Interface

The Instrument should be thermally isolated from the Host Spacecraft.

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[3] In the context of this guideline, the Primary Manufacturers are the spacecraft manufacturers who responded to the CII RFI for GEO Hosted Payload Opportunities. They comprise more than 90% of the GEO commercial satellite market, based upon spacecraft either on-orbit or with publically-announced satellite operator contracts.

Rationale: As a hosted payload, the Instrument should manage its own heat transfer needs without depending on the Host Spacecraft. The common practice in the industry is to thermally isolate the payload from the spacecraft.

2.2.7 Mechanical Interface

The Instrument should be capable of fully acquiring science data when directly mounted to the Host Spacecraft nadir deck.

Rationale: Assessments of potential LEO Host Spacecraft and the responses to the *CII RFI for GEO Hosted Payload Opportunities* indicate nadir-deck mounting of hosted payloads can be accommodated. Alternative mechanical interface locations or kinematic mounts are not prohibited by this guidance but may increase interface complexity.

2.2.8 Mechanical Accommodation

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

Rationale: Analysis of the NICM database indicates that a 100kg allocation represents the upper bound for potential hosted payloads.

[GEO] The Instrument mass should be less than or equal to 150 kg.

Rationale: Analysis of the responses to the *CII RFI for GEO Hosted Payload Opportunities* indicate an instrument of up to 150 kg can be accommodated with minimal impact to existing spacecraft design and function. Instruments exceeding 150 kg can be accommodated but may require additional resources to address growing impacts to existing designs.

2.2.9 [GEO] Attitude Control System Pointing Accommodation

The Instrument 3σ pointing accuracy required should exceed 1440 seconds of arc (0.4 degrees) in each of the Host Spacecraft roll, pitch, and yaw axes.

Rationale: The Host Spacecraft bus pointing accuracy varies significantly both by manufacturer and by spacecraft bus configuration. 1440 arc-seconds represents a pointing accuracy that all of the Primary Manufacturers' buses can achieve. If an Instrument requires a pointing accuracy that is equivalent to or less stringent than this value, then the likelihood of finding a suitable Host Spacecraft increases significantly.

2.2.10 [GEO] Attitude Determination System Pointing Knowledge Accommodation

The Instrument 3σ pointing knowledge required should exceed 450 seconds of arc (0.125 degrees) in the Host Spacecraft roll and pitch axes and 900 seconds of arc (0.25 degrees) in the yaw axis.

Rationale: The Host Spacecraft bus pointing knowledge varies significantly both by manufacturer and by spacecraft bus configuration. 450 arc-seconds (roll/pitch) and 900 arc-seconds (yaw) represent a pointing knowledge that all of the Primary Manufacturers' buses can achieve. If an Instrument requires a pointing knowledge that is equivalent to or less stringent than this value, then the likelihood of finding a suitable Host Spacecraft increases significantly.

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2.2.11 [GEO] <u>Payload Pointing Stability Accommodation</u>

The Instrument should require a short term (≥ 0.1 Hz) 3σ pointing stability that is greater than or equal to 110 seconds of arc/second (0.03 degrees/second) in each spacecraft axis and a long term (Diurnal) 3σ pointing stability that is greater than or equal to 440 seconds of arc (0.12 degrees/second) in each spacecraft axis.

Rationale: Host Spacecraft pointing stability varies significantly both by manufacturer and by bus configuration. In order to maximize the probability of matching an available hosting opportunity, an instrument should be compatible with the maximum pointing stability defined for all responding Host Spacecraft Manufacturers' buses and configurations. According to information provided by industry, the level of short term (≥ 0.1 Hz) pointing stability available for secondary hosted payloads is ≤ 110 seconds of arc/second (0.03 degrees/second) in each of the spacecraft axes. The level of long term (Diurnal) pointing stability available for secondary hosted payloads is ≤ 440 seconds of arc/second (0.12 degrees/second) in each of the spacecraft axes. Therefore, an Instrument pointing stability requirement greater than these values will ensure that any prospective Host Spacecraft bus can accommodate the Instrument.

2.2.12 <u>Environmental Interface</u>

The Instrument should be compatible with and function according to its operational specifications in those environments encountered during Shipping/Storage, Integration and Test, Launch, and Operations as defined in Section 8.0.

Rationale: From the time the Instrument departs the facility in which it was constructed through on-orbit operations and decommissioning, it will encounter disparate environments with which it needs to be compatible with and function reliably and predictably.

2.2.13 <u>Instrument Models</u>

The Instrument Developer should submit finite element, thermal math, mechanical computer aided design, and mass models of the instrument to the Host Spacecraft manufacturer/integrator.

Rationale: The Host Spacecraft manufacturer/integrator requires models of all spacecraft components in order to complete the design portion of the spacecraft lifecycle.

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3.0 HOSTED PAYLOAD WORLDVIEW LEVEL 2 GUIDELINES

3.1 Mission Risk

The Instrument should comply with Mission Risk Class C safety and mission assurance requirements, in accordance with NPR 8705.4.

Rationale: NPR 8705.4 assigns Class C to medium priority, medium risk payloads, with medium to low complexity, short mission lifetime, and medium to low cost. The EVI-1 Announcement of Opportunity solicited "… proposals for science investigations requiring the development and operation of space-based instruments, designated as Class C on a platform to be identified by NASA at a later date."[4]

3.2 Instrument End of Life

The Instrument should place itself into a "safe" configuration upon reaching its end of life to prevent damage to the Host Spacecraft or any other payloads.

Rationale: The Instrument may have potential energy remaining in components such as pressure vessels, mechanisms, batteries, and capacitors, from which a post-retirement failure might cause damage to the Spacecraft Host or its payloads. The Instrument Developer should develop, in concert with the Host Spacecraft and the Satellite Operator, an End of Mission Plan that specifies the actions that the Instrument payload and Host Spacecraft will take to "safe" the Instrument payload by reduction of potential energy once either party declares the Instrument's mission "Complete."

3.3 Prevention of Failure Back-Propagation

The Instrument and all of its components should prevent anomalous conditions, including failures, from propagating to the Host Spacecraft or other payloads.

Rationale: The Instrument design should isolate the effects of Instrument anomalies and failures, such as power spikes, momentum transients, and electromagnetic interference so that they are contained within the boundaries of the Instrument system.

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[4] "Earth Venture Instrument-1," from Program Element Appendix (PEA) J of the Second Stand Alone Missions of Opportunity Notice (SALMON-2), 2012.

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4.0 DATA LEVEL 2 GUIDELINES

4.1 Assumptions

The CII data guidelines assume the following regarding the Host Spacecraft:

1) During the matching process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the data interface. The Data Interface Control Document (DICD) will record those parameters and decisions.

4.2 Data Interface Guidelines

4.2.1 <u>Command Dictionary</u>

The Instrument Provider should provide a command dictionary to the Host Spacecraft Manufacturer, the format and detail of which will be negotiated with the Host Spacecraft Manufacturer.

Rationale: Best practice and consistent with DICD. A command dictionary defines all instrument commands in detail, by describing the command, including purpose, preconditions, possible restrictions on use, command arguments and data types (including units of measure, if applicable), and expected results (*e.g.* hardware actuation and/or responses in telemetry) in both nominal and off-nominal cases. Depending on the level of detail required, a command dictionary may also cover binary formats (*e.g.* packets, opcodes, *etc*.).

4.2.2 <u>Telemetry Dictionary</u>

The Instrument Provider should provide a telemetry dictionary to the Host Spacecraft Manufacturer, the format and detail of which will be negotiated with the Host Spacecraft Manufacturer.

Rationale: Best practice and consistent with DICD. A telemetry dictionary defines all information reported by the instrument in detail, by describing the data type, units of measure, and expected frequency of each measured or derived value. If telemetry is multiplexed or otherwise encoded (*e.g.* into virtual channels), the telemetry dictionary will also describe decommutation procedures which may include software or algorithms. By their nature, telemetry dictionaries often detail binary packet formats.

4.2.3 <u>SAFE mode</u>

The Instrument should provide a 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).

4.2.4 <u>Command (SAFE mode)</u>

The Instrument should enter SAFE mode when commanded either directly by the Host Spacecraft or via ground operator command.

Rationale: The ability to put the Instrument into SAFE mode protects and preserves both the Instrument and the Host Spacecraft under anomalous and resource constrained conditions.

4.2.5 Command (Data Flow Control)

The Instrument should respond to commands to suspend and resume the transmission of Instrument telemetry and Instrument science data.

Rationale: Data flow control allows the Spacecraft, Satellite Operator, and ground operations team to devise and operate Fault Detection Isolation, and Recovery (FDIR) procedures, crucial for on-orbit operations.

4.2.6 Command (Acknowledgement)

The Instrument should acknowledge the receipt of all commands, in its telemetry.

Rationale: Command acknowledgement allows the Spacecraft, Satellite Operator, and ground operations team to devise and operate FDIR procedures, crucial for on-orbit operations.

4.3 Data Accommodation Guidelines

4.3.1 Onboard Science Data Storage

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

Rationale: Buffering all data on the Instrument imposes no storage capacity requirements on the Spacecraft. This is consistent with the direct-to-transponder science data interface [GEO]. A spacecraft need only enough buffer capacity to relay Instrument telemetry. Fewer resource impacts on the Spacecraft maximize Instrument hosting opportunities.

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5.0 ELECTRICAL POWER SYSTEM LEVEL 2 GUIDELINES

5.1 Assumptions

The CII electrical power guidelines assume the following regarding the Host Spacecraft:

1) During the matching process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the electrical power interface. The Electrical Power Interface Control Document (EICD) will record those parameters and decisions.
2) [LEO] The Host Spacecraft will supply to the Instrument EPS unregulated (sun regulated) electrical power 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.
3) [GEO] The Host Spacecraft will supply to the Instrument EPS regulated electrical power within the range of 28 ±3 VDC, including ripple and normal transients as defined below, and power distribution losses due to switching, fusing, harness and connectors.
4) [LEO] The Host Spacecraft will provide connections to two 50W (Orbital Average Power: OAP) power buses as well as a dedicated bus to power the Instrument's survival heaters. Each bus will have a primary and redundant circuit. For the purpose of illustration, this document labels these buses as Power Bus #1, Power Bus #2, and Survival Heater Power Bus. This document also labels the primary and redundant circuits as A and B, respectively. Figure 5-1 shows a pictorial representation of this architecture.
5) [GEO] The Host Spacecraft will provide connections to two 150W (Average Power: AP) power buses as well as a dedicated bus to power the Instrument's survival heaters. Each power bus will be capable of supporting both primary and redundant power circuits. For the purpose of illustration, this document labels these buses as Power Bus #1, Power Bus #2, and Survival Heater Power Bus. This document also labels the primary and redundant circuits as A and B, respectively. Figure 5-1 shows a pictorial representation of this architecture.
6) The Host Spacecraft will energize the Survival Heater Power Bus at 30% of the OAP [LEO]/AP [GEO] in accordance with the mission timeline documented in the EICD.
7) The Host Spacecraft Manufacturer will supply a definition of the maximum source impedance by frequency band. Table 5-1 provides an example of this definition.
8) The Host Spacecraft Manufacturer will furnish all Spacecraft and Spacecraft-to-Instrument harnessing.
9) The Host Spacecraft will deliver Instrument power via twisted conductor (pair, quad, etc.) cables with both power and return leads enclosed by an electrical overshield.

10) The Host Spacecraft will protect its own electrical power system via overcurrent protection devices on its side of the interface.

11) The Host Spacecraft will utilize the same type of overcurrent protection device, such as latching current limiters or fuses, for all connections to the Instrument.

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

13) The Host Spacecraft will deliver a maximum transient current on any Power Feed bus of 100 percent (that is, two times the steady state current) of the maximum steady-state current for no longer than 50 ms.

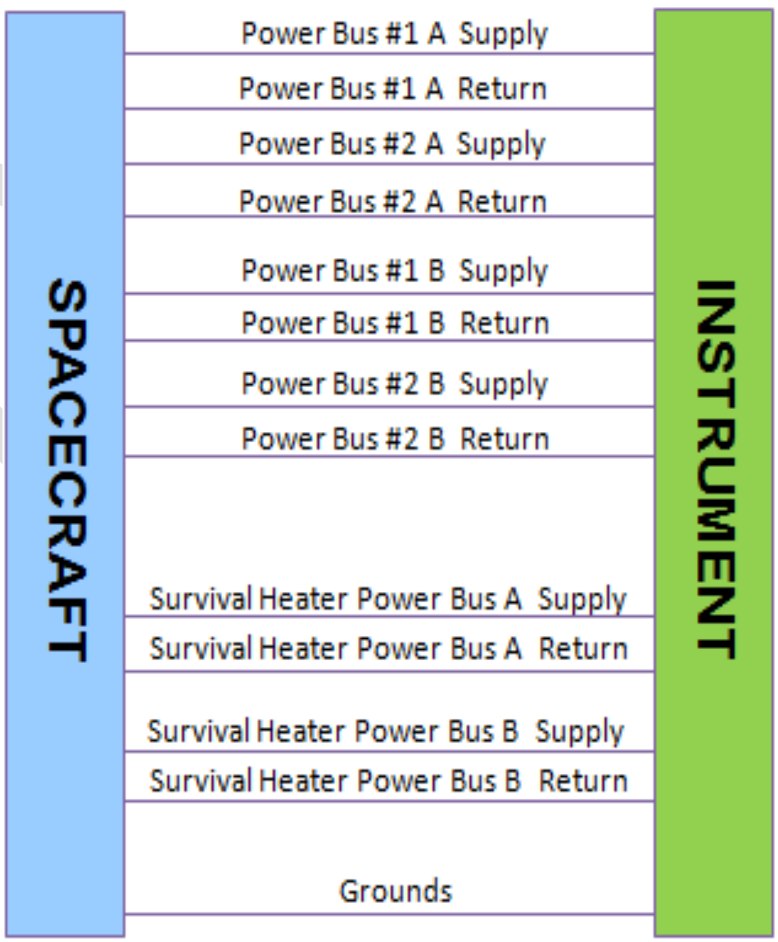

Figure 5-1: Spacecraft-Instrument Electrical Interface (Depicted with the optional Instrument side redundant Power Bus B interface)

Table 5-1: Example of Power Source Impedance Function

Frequency	Maximum Source Impedance [Ω]
1 Hz to 1 kHz	0.2
1 kHz to 20 kHz	1.0
20 kHz to 100 kHz	2.0
100 kHz to 10 MHz	20.0

5.2 EPS Interface

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

5.2.1 Power Bus Interface

The EPS should provide nominal power to each Instrument component via one or both of the Power Buses.

Rationale: The Power Buses supply the electrical power for the Instrument to conduct normal operations. Depending on the load, a component may connect to one or both of the power buses.

Note: The utilization of the redundant power circuits (Power Circuits B) by the Instrument is optional based upon instrument mission classification, reliability, and redundancy requirements.

5.2.2 Survival Heater Bus Interface

The EPS should provide power to the survival heaters via the Survival Heater Power Bus.

Rationale: The Survival Heaters, which are a member of the Thermal subsystem, require power to heat certain instrument components during off-nominal scenarios when the Power Buses are not fully energized. See Best Practices sections 9.2.3 and 9.4.2 for more discussion about survival heaters.

5.2.3 Grounding

The Instrument grounding architecture should comply with NASA-HDBK-4001.

Rationale: The Instrument grounding architecture must be established at the earliest point in the design process. The implementation of the subject level 1 guidance in conjunction with the consistent and proven design principles described in the ascribed reference will support a successful instrument development and integration to a Host Spacecraft and mission.

5.2.4 Grounding Documentation

The EICD will document how the Instrument will ground to the Host Spacecraft.

Rationale: It is necessary to define and document the Instrument to Host Spacecraft grounding interface architecture.

5.2.5 Bonding

The Instrument bonding should comply with NASA-STD-4003.

Rationale: The instrument bonding practices must be defined to support the instrument design and development process. The implementation of the subject reference will provide consistent

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and proven design principles and support a successful instrument development, integration to a Host Spacecraft and mission.

5.2.6 Mitigation of In-Space Charging Effects

The Instrument should comply with NASA-HDBK-4002 to mitigate in-space charging effects.

Rationale: The application of the defined reference to the Instrument grounding architecture and bonding practices will address issues and concerns with the in-flight buildup of charge on internal spacecraft components and external surfaces related to space plasmas and high-energy electrons and the consequences of that charge buildup.

5.2.7 Instrument Harnessing

The Instrument Developer should furnish all Instrument harnessing.

Rationale: The Instrument Developer is responsible for all harnesses that are constrained by the boundaries of the Instrument as a single and unique system. This refers only to those harnesses that are interconnections between components (internal and external) of the Instrument system and excludes any harnesses interfacing with the Host Spacecraft or components that are not part of the Instrument system.

5.2.8 Harness Documentation

The EICD will document all harnesses, harness construction, pin-to-pin wiring, cable type, connectors, ground straps, and associated service loops.

Rationale: The EICD documents agreements made between the Host Spacecraft Manufacturer and Instrument Developer regarding harness hardware and construction.

5.3 EPS Accommodation

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.

5.3.1 Definitions

Average Power Consumption: the total power consumed averaged over any 180-minute period.

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

5.3.2 Instrument Power Harness

Instrument power harnesses should be sized to support the peak Instrument power level and both Host Spacecraft and Instrument overcurrent protection devices.

Rationale: Sizing all components of the Instrument power harness, such as the wires, connectors, sockets, and pins to the peak power level required by the Instrument and Host Spacecraft prevents damage to the power harnessing.

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5.3.3 Allocation of Instrument Power

The EPS should draw no more power from the Host Spacecraft in each Instrument mode than defined in Table 5-2.

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.

Note: Instrument and Instrument survival heater power should not exceed the defined power allocation at end-of-life at worst-case low bus voltage.

Note: The instrument modes are notional and based upon an example provided in Appendix G.

Table 5-2: Instrument Power Allocation

Mode	LEO		GEO
	Peak (W)	Average (W)	Average (W)
OFF/ SURVIVAL	0/60	0/30	0/90
ACTIVATION	200	100	300
SAFE	100	50	150
OPERATION	200	100	300

5.3.4 Unannounced Removal of Power

The Instrument should function according to its operational specifications when nominal power is restored following an unannounced removal of power.

Rationale: In the event of a Host Spacecraft electrical malfunction, the instrument would likely be one of the first electrical loads to be shed either in a controlled or uncontrolled manner.

5.3.5 Reversal of Power

The Instrument should function according to its operational specifications when proper polarity is restored following a reversal of power (positive) and ground (negative).

Rationale: This defines the ability of an instrument to survive a power reversal anomaly which periodically occurs during assembly, integration, and test (AI&T).

5.3.6 Power-Up and Power-Down

The Instrument should function according to its operational specifications when the Host Spacecraft changes the voltage across the Operational Bus from +28 to 0 VDC or from 0 to +28 VDC as a step function.

Rationale: A necessary practice to preclude instrument damage/degradation.

5.3.7 Abnormal Operation Steady-State Voltage Limits

The Instrument should function according to its operational specifications when the Host Spacecraft restores nominal power following exposure to steady-state voltages from 0 to 50 VDC.

Rationale: Defines a verifiable (testable) limit for off-nominal input voltage testing of an instrument.

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6.0 MECHANICAL LEVEL 2 GUIDELINES

6.1 Assumptions

The CII mechanical guidelines assume the following regarding the Host Spacecraft:

1) During the matching process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the mechanical interface. The Mechanical Interface Control Document (MICD) will record those parameters and decisions.
2) The Host Spacecraft will accommodate fields-of-view (FOV) that equal or exceed the Instrument science and radiator requirements.
3) The Host Spacecraft Contractor will furnish all instrument mounting fasteners.

6.2 Mechanical Interface Guidelines

6.2.1 Functionality in 1 g Environment

The Instrument should function according to its operational specifications in any orientation while in the integration and test environment.

Rationale: As a hosted payload, the Instrument will attach to one of multiple decks on the Host Spacecraft. Its orientation with respect to the Earth's gravitational field during integration and test will not be known during the instrument design process. The function of the instrument and accommodation of loads should not depend on being in a particular orientation.

6.2.2 Stationary Instrument Mechanisms

The Instrument should cage any mechanisms that require restraint, without requiring Host Spacecraft power to maintain the caged condition, throughout the launch environment.

Rationale: As a hosted payload, the Instrument should not assume that the Host Spacecraft will provide any power during launch.

6.3 Mechanical Accommodation Guidelines

6.3.1 Dimensions

[LEO] The Instrument and all of its components should remain within the detailed instrument envelope of 400mm × 500mm× 850mm (H×W×L) during all phases of flight.

Rationale: Engineering analysis determined guideline payload volume based on mass guidelines and comparisons to spacecraft envelopes in the NASA Rapid Spacecraft Development Office (RSDO) catalog.

[GEO] The Instrument and all of its components should remain within the detailed instrument envelope of 1000mm ×1000mm×1000mm (H×W×L) during all phases of flight.

Rationale: Engineering analysis determined guideline payload volume based on mass guidelines and comparisons to spacecraft envelopes in responses to the *CII RFI for GEO Hosted Payload Opportunities and Accommodations.*

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7.0 THERMAL LEVEL 2 GUIDELINES

7.1 Assumptions

The CII thermal guidelines assume the following regarding the Host Spacecraft:

1) During the matching process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the thermal power interface. The Thermal Interface Control Document (TICD) will record those parameters and decisions.
2) The Host Spacecraft will maintain a temperature range of between -40° C and 70°C on the Spacecraft side of the interface from the Integration through Disposal portions of its lifecycle.
3) The Spacecraft Manufacturer will be responsible for thermal hardware used to close out the interfaces between the Instrument and Spacecraft, such as closeout Multi-layer Insulation (MLI).

7.2 Thermal Interface

7.2.1 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.

Rationale: As a hosted payload, the instrument should not interfere with the Host Spacecraft's functions. The common practice in the industry is to thermally isolate the payload from the spacecraft.

7.2.2 Conductive Heat Transfer

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

Rationale: A conductive heat transfer of 15 W/m^2 or 4 W is considered small enough to meet the intent of being thermally isolated.

7.2.3 Radiative Heat Transfer

The TICD will document the allowable radiative heat transfer from the Instrument to the Host Spacecraft.

Rationale: There is a limit to how much heat the Instrument should transmit to the Host Spacecraft via radiation, but that limit will be unknown prior to the thermal analysis conducted following Instrument-to-Host Spacecraft pairing. The TICD will document that future negotiated value.

7.2.4 Temperature Maintenance Responsibility

The Instrument should maintain its own instrument temperature requirements.

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Rationale: As a thermally isolated payload, the Instrument has to manage its own thermal properties without support from the Host Spacecraft.

7.2.5 Instrument Allowable Temperatures

The TICD will document the allowable temperature ranges that the Instrument will maintain in each operational mode/state.

Rationale: Defining the instrument allowable temperatures drives the performance requirements for the thermal management systems for both the Instrument as well as the Host Spacecraft.

7.2.6 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.

Rationale: This responsibility naturally follows the responsibility for the instrument thermal design and maintaining the temperature requirements of the instrument.

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8.0 ENVIRONMENTAL LEVEL 2 GUIDELINES

8.1 Assumptions

The CII environmental guidelines assume the following regarding the Host Spacecraft, launch vehicle, and/or integration and test facilities:

1) During the matching process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the environmental interface. The Environmental Interface Control Document (EICD) will record those parameters and decisions.

Note: the design of the Instrument modes of operation are the responsibility of the Instrument Developer. For purposes of illustration, the operational modes in this section are equivalent to the Instrument modes and states as defined in Appendix G.

8.2 Shipping/Storage Environment

The Shipping/Storage Environment represents the time in the Instrument's lifecycle between when it departs the Instrument Developer's facility and arrives at the facility of the Spacecraft Manufacturer/Systems Integrator. The Instrument is dormant and attached mechanically to its container (see Figure 8-1).

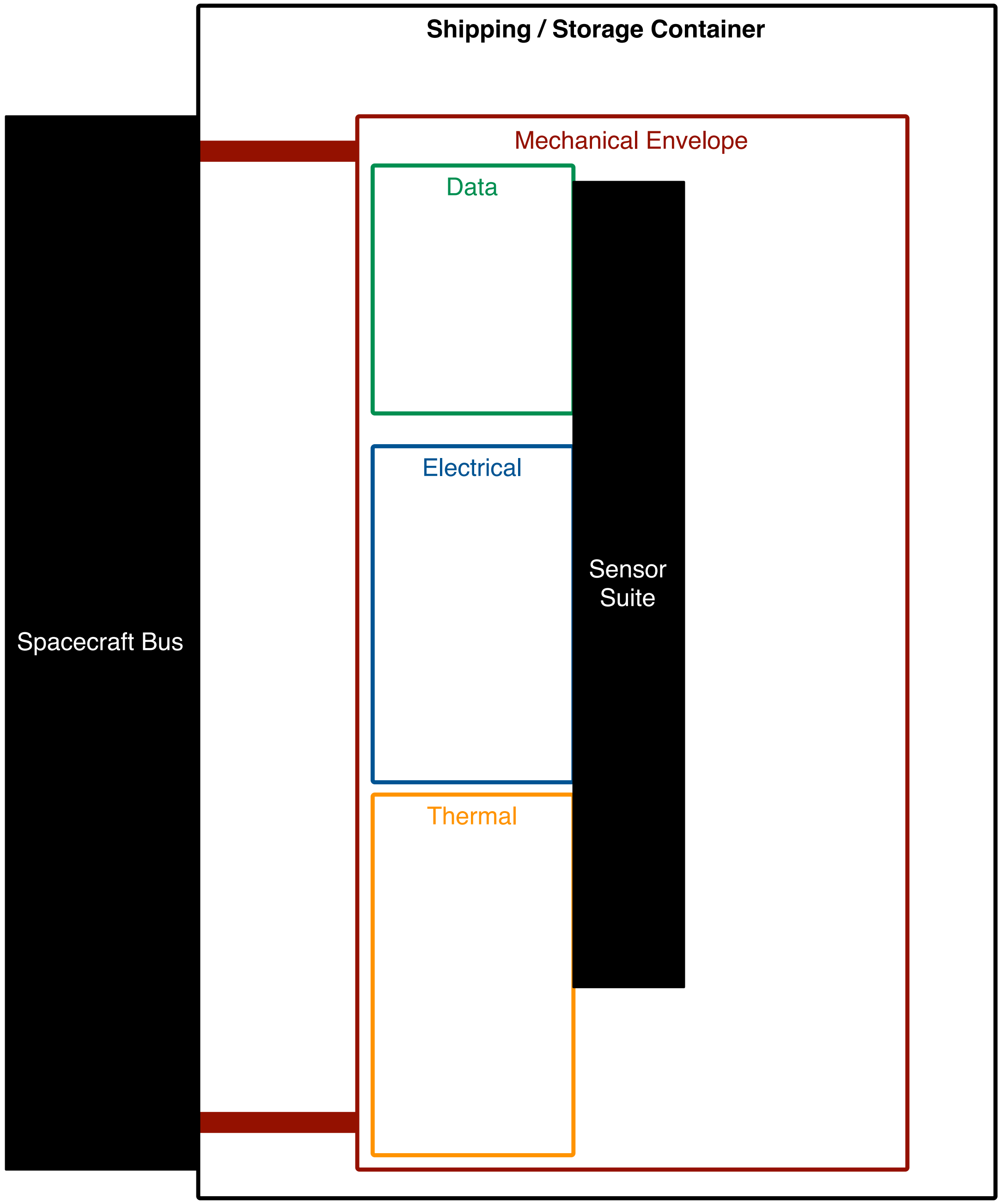

Figure 8-1: Shipping / Storage Environment

8.2.1 <u>Documentation</u>

The EICD will document the expected environment the Instrument will experience between the departure from the Instrument assembly facility and arrival at the Host Spacecraft integration facility.

Rationale: The nature of the Shipping/Storage Environment depends upon the point at which physical custody of the Instrument transfers from Instrument Developer to the Satellite Contractor/Systems Integrator as well as negotiated agreements on shipping/storage procedures.

The interfaces associated with the shipping/storage environment include the allowable temperatures and the characteristics of the associated atmosphere.

8.2.2 Instrument Configuration

The EICD will document the configuration and operational state of the Instrument during the Shipping/Storage phase.

Rationale: Specifying the configuration of the Instrument during shipping/storage drives the volume requirements for the container as well as any associated support equipment and required services.

The Instrument will likely be in the OFF/SURVIVAL mode while in this environment.

8.3 Integration and Test Environment

The Integration and Test Environment represents the time in the Instrument’s lifecycle between when it arrives at the facility of the Spacecraft Manufacturer/Systems Integrator through payload encapsulation at the launch facility. During this phase, the Host Spacecraft Manufacturer/Systems Integration will attach the Instrument to the Host Spacecraft Bus and verify that system performs as designed throughout various environmental and dynamics regimes. The Instrument may be attached to the Host Spacecraft Bus or to various ground support equipment that transmits power, thermal conditioning, and diagnostic data (see Figure 8-2).

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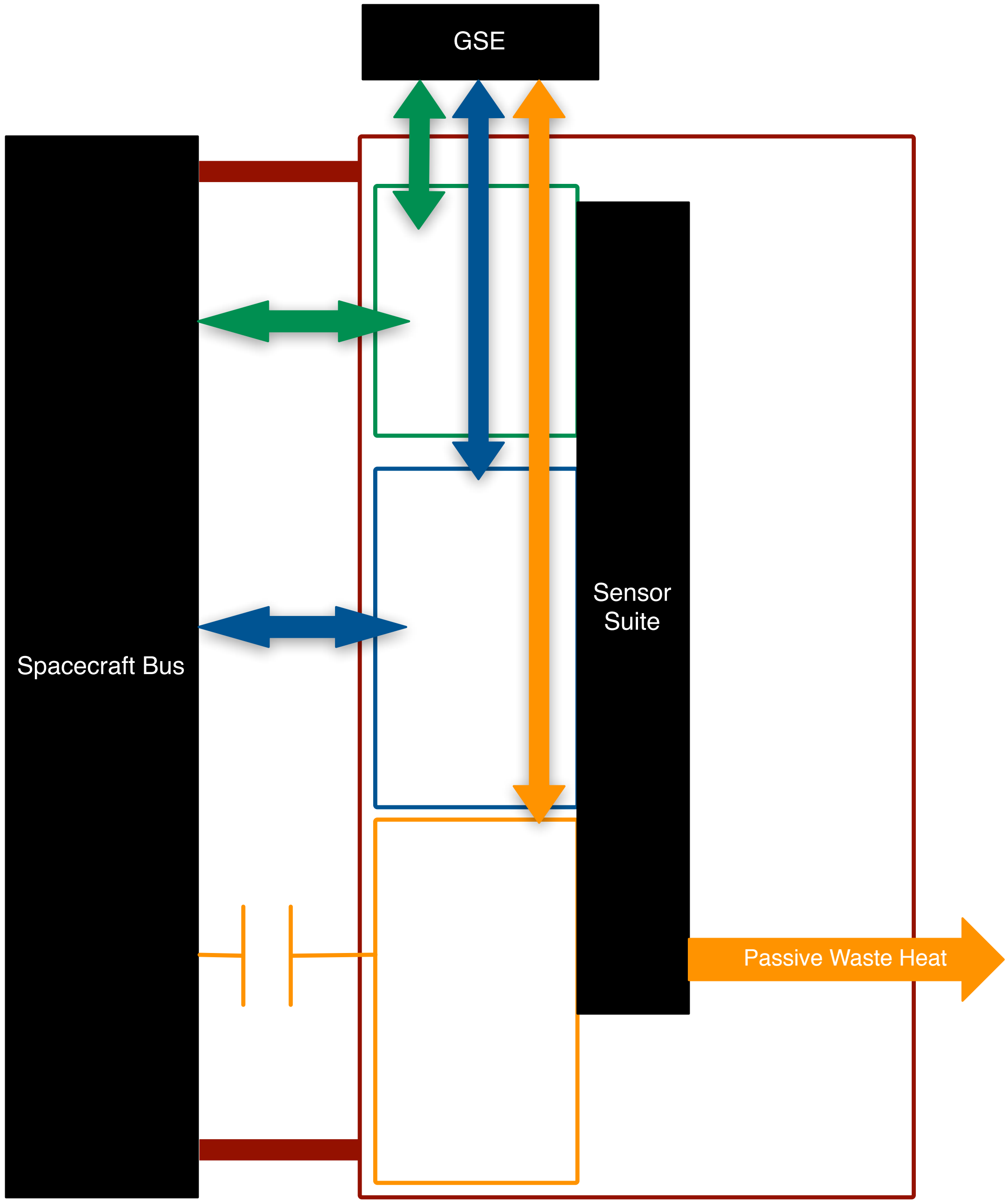

Figure 8-2: Integration and Test Environment

8.3.1 <u>Documentation</u>

The EICD will document the expected environments the Instrument will experience between arrival at the Host Spacecraft integration facility and Launch.

Rationale: The nature of the Integration and Test Environment depends upon the choice of Host Spacecraft and Launch Vehicle as well as the negotiated workflows at the Systems Integration and Launch facilities.

Example environmental properties include the thermal, dynamic, atmospheric, electromagnetic, radiation characteristics of each procedure in the Integration and Test process. The EICD may either record these data explicitly or refer to a negotiated Test and Evaluation Master Plan (TEMP).

8.3.2 Instrument Configuration

The EICD will document the configuration and operational mode of the Instrument during the Integration and Test phase.

Rationale: Proper configuration of the Instrument during the various Integration and Test procedures ensures the validity of the process.

8.4 Launch Environment

The Launch Environment represents that time in the Instrument's lifecycle when it is attached to the launch vehicle via the Host Spacecraft, from payload encapsulation at the Launch facility through the completion of the launch vehicle's final injection burn (see Figure 8-3).

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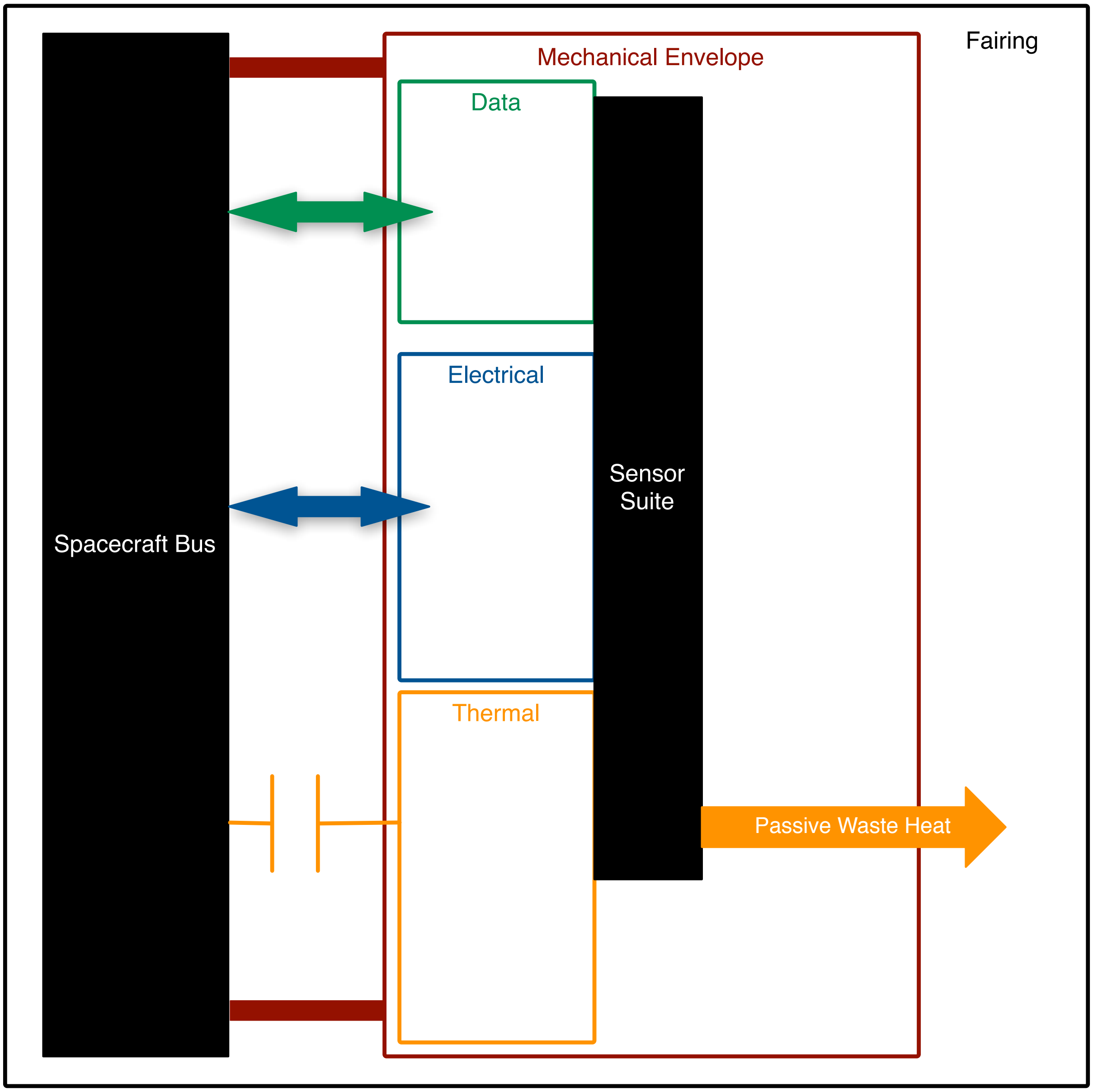

Figure 8-3: Launch Environment

8.4.1 Documentation

The EICD will document the expected environments the Instrument will experience between Launch and Host Spacecraft / Launch Vehicle separation.

Rationale: The nature of the Launch Environment depends upon the choice of Host Spacecraft and Launch Vehicle. Significant parameters related to the launch environment include temperature, pressure, and acceleration profiles.

8.4.2 Instrument Configuration

The EICD will document the configuration and operational state of the Instrument during the Launch phase.

Rationale: The Launch phase is the most dynamic portion of the mission, and the Instrument configuration and operational mode are chosen to minimize damage to either the Instrument or Host Spacecraft. The Instrument will likely be in the OFF/SURVIVAL mode while in this environment.

The following guidelines are representative of a typical launch environment but may be tailored on a case-by-case basis.

8.4.3 Launch Pressure Profile

The Instrument should function according to its operational specifications after being subjected to an atmospheric pressure decay rate of 7 kPa/s (53 Torr/s).

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environments without suffering degraded performance or being damaged or inducing degraded performance of, or damage to, the Host Spacecraft or other payloads. This guidance represents the maximum expected pressure decay rate during launch ascent and applies to LEO and GEO launch vehicles. The GEO guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses, the *General Environmental Verification Specification for STS & ELV Payloads, Subsystems, and Components (GEVS-SE)*, and *Geostationary Operational Environmental Satellite GOES-R Series General Interface Requirements Document (GOES-R GIRD).*

8.4.4 Quasi-Static Acceleration

[GEO] The Instrument should function according to its operational specifications after being subjected to a launch vehicle-induced quasi-static acceleration environment represented by the MAC defined in Table 8-1.

Table 8-1: [GEO] Mass Acceleration Curve

Mass [kg]	Acceleration [g]
0 to 2.5	± 55
2.5 to 30	= ± (-1.273 × Mass + 58.182)
>30	± 20

Rationale: The Instrument must able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. This guidance represents the need to be compatible with the quasi-static loads that will be experienced during launch ascent. The GEO guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses, the *GEVS-SE,* and *GOES-R GIRD*.

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The "Mass" is the mass of the entire instrument or any component of the instrument. The MAC applies to the worst-case 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 and also to the two remaining orthogonal directions.

8.4.5 Sinusoidal Vibration

The Instrument should function according to its operational specifications after being subjected to a launch vehicle-induced transient environment represented by the sinusoidal vibration environment defined in Table 8-2.

Table 8-2: Sinusoidal Vibration Environment

Frequency (Hz)	Acceleration Amplitudes	
	Acceptance	Qualification
2 – 5	1.0 g peak	1.4 g peak
5 – 18	1.4 g peak	2.0 g peak
18 – 30	1.5 g peak	2.1 g peak
30 – 40	1.0 g peak	1.4 g peak
40 – 55	3.0 g peak	4.2 g peak
55 – 100	1.0 g peak	1.4 g peak
Acceptance Sweep Rate: From 5 to 100 Hz at 1.0 octaves/minute except from 40 to 55 Hz at 12 Hz/min Qualification Sweep Rate: From 5 to 100 Hz at 0.5 octaves/minute except from 40 to 55 Hz at 6 Hz/min		

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. This guidance represents the need to be compatible with the coupled dynamics loads that will be experienced during ground processing and launch ascent. The GEO guideline is the all-satisfy strategy scenario, based upon CII analysis of all publicly available launch vehicle payload planner's guides.

8.4.6 Random Vibration

[LEO] The Instrument should function according to its operational specifications after being subjected to a launch vehicle-induced transient environment represented by the random vibration environment defined in Table 8-3.

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 8-3: [LEO] Random Vibration Environment (derived from GEVS-SE, Table 2.4-4)

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 8-3 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.

[GEO] The Instrument should function according to its operational specifications after being subjected to a launch vehicle-induced transient environment represented by the random vibration environment defined in Table 8-4.

All flight article test durations are to be 1 minute per axis. Protoflight and non-flight article qualification test durations are to be 3 minutes per axis.

Table 8-4: [GEO] Random Vibration Environment

Zone/Assembly	Frequency (Hz)	Protoflight / Qualification	Acceptance
Instrument	20	0.2 g^2/Hz	0.14 g^2/Hz
	20 - 45	+6 dB/octave	+6 dB/octave
	45 - 500	1.0 g^2/Hz	0.71 g^2/Hz
	500 - 2000	-6 dB/octave	-6 dB/octave
	2000	0.06 g^2/Hz	0.04 g^2/Hz
	Overall	28.9 g_{rms}	24.2 g_{rms}

Table 8-4 represents the random vibration environment for all instruments

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Rationale: The Instrument must able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. This guidance represents the need to be compatible with the random vibration that will be experienced during launch ascent. 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. The guidelines are the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses (GEO only), the *GEVS-SE* (LEO and GEO), and *GOES-R GIRD* (GEO only).

8.4.7 Acoustic Noise

The Instrument should function according to its operational specifications after being subjected to a launch vehicle-induced transient environment represented by the acoustic noise spectra defined in Table 8-5.

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Table 8-5: Acoustic Noise Environment

1/3 Octave Band Center Frequency (Hz)"	Design/Qual/Protoflight (dB w/ 20 μPa reference)"	Acceptance (dB w/ 20 μPa reference)"
25	128.23	125.23
31.5	132	129
40	133.5	130.5
50	134	131
63	135	132
80	136.6	133.6
100	137.4	134.4
125	136.3	133.3
160	137.1	134.1
200	137.23	134.23
250	138.2	135.2
315	139	136
400	137.5	134.5
500	134.23	131.23
630	134.23	131.23
800	131.5	128.5
1000	129.23	126.23
1250	129.23	126.23
1600	124.8	121.8
2000	125	122
2500	124.23	121.23
3150	121.5	118.5
4000	120	117
5000	120	117
6300	118	115
8000	118	115
10000	119	116

Rationale: Acoustic design guidelines are based on maximum internal payload fairing sound pressure level spectra. The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. This guidance represents the need to be compatible with the acoustic noise that will be experienced during launch ascent. The guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses, all publically available launch vehicle Payload Planers Guides (with the exception of the Long March LV) and the *GOES-R GIRD.*

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 / protoflight exposure time is 2 minutes; acceptance exposure time is one minute.

8.4.8 Mechanical Shock

[GEO] The Instrument should function according to its operational specifications after being subjected to a spacecraft to launch vehicle separation or other shock transient accelerations represented by Table 8-6.

Table 8-6: [GEO] Mechanical Shock Environment

Frequency [Hz]	Acceleration [g]
100	**115.1**
600	**2000**
2000	**5000**
10000	**5000**

Rationale: The Instrument must able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. This guidance represents the need to be compatible with the mechanical shock that will be experienced during ground processing, launch ascent and on orbit. The GEO guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses, NASA GEVS and the GOES R GIRD. A quality factor (Q) of 10 is a typical value for a pyrotechnic separation system shock event. This value may be tailored based upon the shock environments anticipated/defined following the pairing of the Instrument and Host Spacecraft.

8.5 Operational Environment

The Operational Environment represents that time in the Instrument's lifecycle following the completion of the launch vehicle's final injection burn, when the Instrument is exposed to space and established in its operational orbit (Figure 8-4).

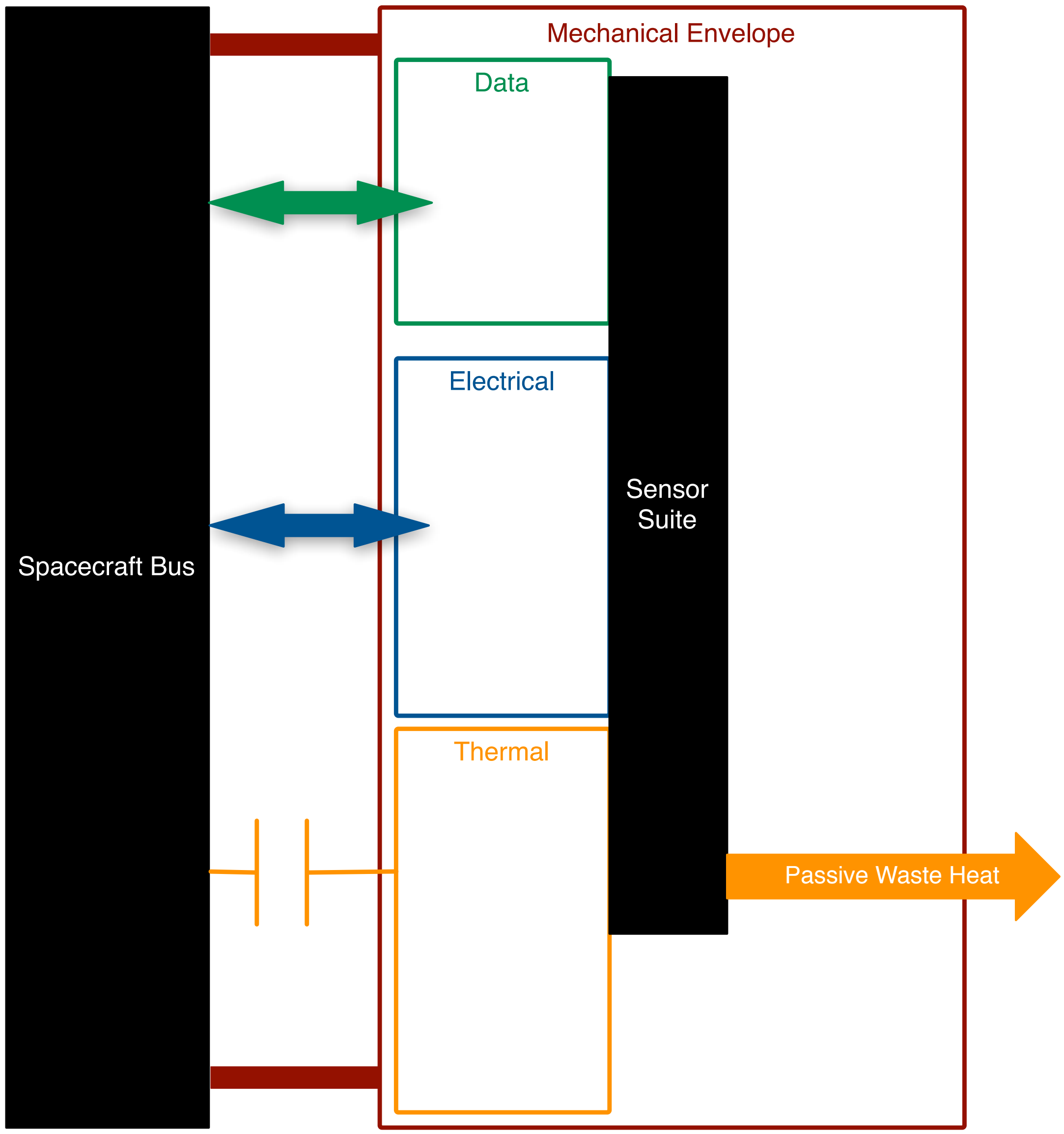

Figure 8-4: Operational Environment

Unless otherwise stated, the LEO guidelines are based upon a 98 degree inclination, 705 km altitude circular orbit. The GEO guidelines are based upon a zero degree inclination, 35786 km altitude circular orbit.

8.5.1 <u>Orbital Acceleration</u>

The Instrument should function according to its operational specifications after being subjected to a maximum spacecraft-induced acceleration of 0.04g.

Rationale: The Instrument in its operational configuration must able to withstand conditions typical of the on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the Host Spacecraft or other payloads. This guidance represents the need to be compatible with the accelerations that will be experienced on orbit. The guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses, the *GEVS-SE,* and *GOES-R GIRD*.

8.5.2 Corona

The Instrument should exhibit no effect of corona or other forms of electrical breakdown after being subjected to a range of ambient pressures from 101 kPa (~760 Torr) at sea level to 1.3×10^{-15} kPa (10^{-14} Torr) in space.

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. This guidance represents the need to be compatible with the environment that will be experienced during ground processing, launch ascent and on orbit. The guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses, the *GEVS-SE,* and *GOES-R GIRD*.

8.5.3 Thermal Environment

The Instrument should function according to its operational specifications after being subjected to a thermal environment characterized by Table 8-7.

Table 8-7: Thermal Radiation Environment

Domain	Solar Flux [W/m^2]	Earth IR (Long Wave) [W/m^2]	Earth Albedo
LEO	1290 to 1420	222 to 243	0.275 to 0.375
GEO		Insignificant	Insignificant

Rationale: The Instrument must be able to withstand conditions typical of the on-orbit environment without suffering degraded performance or being damaged or inducing degraded performance of or damage to the spacecraft host or other payloads. While the Earth albedo and long wave infrared radiation are non-zero values at GEO, their contribution to the overall thermal environment is less than 0.05% of that from solar flux.

8.5.4 Radiation Design Margin

Every hardware component of the Instrument should have a minimum RDM value of two.

Rationale: Exposure to radiation degrades many materials and will require mitigation to assure full instrument function over the design mission lifetime. This guidance defines the need to carry 100% margin against the estimated amount of radiation exposure that will be experienced in Earth orbit in support of said mitigation.

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.

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8.5.5 Total Ionizing Dose

The Instrument should function according to its operational specifications during and after exposure to the Total Ionizing Dose (TID) radiation environment based upon the specified mission orbit over the specified mission lifetime.

Table 8-8 shows the expected total ionizing dose for object in a 813 km, sun-synchronous orbit, for over the span of two years, while shielded by an aluminum spherical shell of a given thickness. Figure 8-5 plots the same data in graphical form. The data contain no margin or uncertainty factors.

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Table 8-8: [LEO] Total Ionizing Dose Radiation Environment

Shield Thickness [mil]	Trapped Electrons Rad [Si]	Bremsstrahlung Rad [Si]	Trapped Protons Rad [Si]	Solar Protons Rad [Si]	Total Rad [Si]
1	1.09E+06	1.84E+03	5.24E+04	6.52E+04	1.21E+06
3	5.23E+05	1.03E+03	1.70E+04	2.81E+04	5.69E+05
4	3.99E+05	8.30E+02	1.29E+04	2.18E+04	4.35E+05
6	2.44E+05	5.70E+02	8.86E+03	1.48E+04	2.68E+05
7	1.98E+05	4.87E+02	7.70E+03	1.29E+04	2.19E+05
9	1.38E+05	3.72E+02	6.30E+03	1.04E+04	1.55E+05
10	1.18E+05	3.32E+02	5.79E+03	9.47E+03	1.34E+05
12	9.04E+04	2.70E+02	5.01E+03	7.92E+03	1.04E+05
13	8.03E+04	2.46E+02	4.72E+03	7.31E+03	9.25E+04
15	6.45E+04	2.08E+02	4.28E+03	6.28E+03	7.53E+04
29	2.31E+04	9.80E+01	2.80E+03	2.96E+03	2.90E+04
44	1.23E+04	6.33E+01	2.18E+03	1.94E+03	1.65E+04
58	7.93E+03	4.75E+01	1.89E+03	1.47E+03	1.13E+04
73	5.24E+03	3.71E+01	1.70E+03	1.14E+03	8.12E+03
87	3.66E+03	3.06E+01	1.57E+03	9.30E+02	6.19E+03
117	1.81E+03	2.22E+01	1.39E+03	6.40E+02	3.86E+03
146	9.59E+02	1.76E+01	1.28E+03	4.52E+02	2.71E+03
182	4.38E+02	1.40E+01	1.19E+03	3.13E+02	1.95E+03
219	1.90E+02	1.17E+01	1.12E+03	2.47E+02	1.56E+03
255	8.38E+01	1.01E+01	1.06E+03	2.20E+02	1.38E+03
292	3.55E+01	8.97E+00	1.02E+03	1.98E+02	1.26E+03
365	5.72E+00	7.43E+00	9.34E+02	1.61E+02	1.11E+03
437	6.98E-01	6.46E+00	8.76E+02	1.38E+02	1.02E+03
510	4.96E-02	5.77E+00	8.32E+02	1.22E+02	9.60E+02
583	7.76E-04	5.26E+00	7.77E+02	1.05E+02	8.87E+02
656	1.06E-05	4.85E+00	7.38E+02	9.35E+01	8.36E+02
729	1.37E-07	4.49E+00	7.06E+02	8.50E+01	7.95E+02
875	0.00E+00	3.92E+00	6.42E+02	7.02E+01	7.16E+02
1167	0.00E+00	3.14E+00	5.42E+02	5.09E+01	5.96E+02
1458	0.00E+00	2.61E+00	4.67E+02	3.90E+01	5.09E+02

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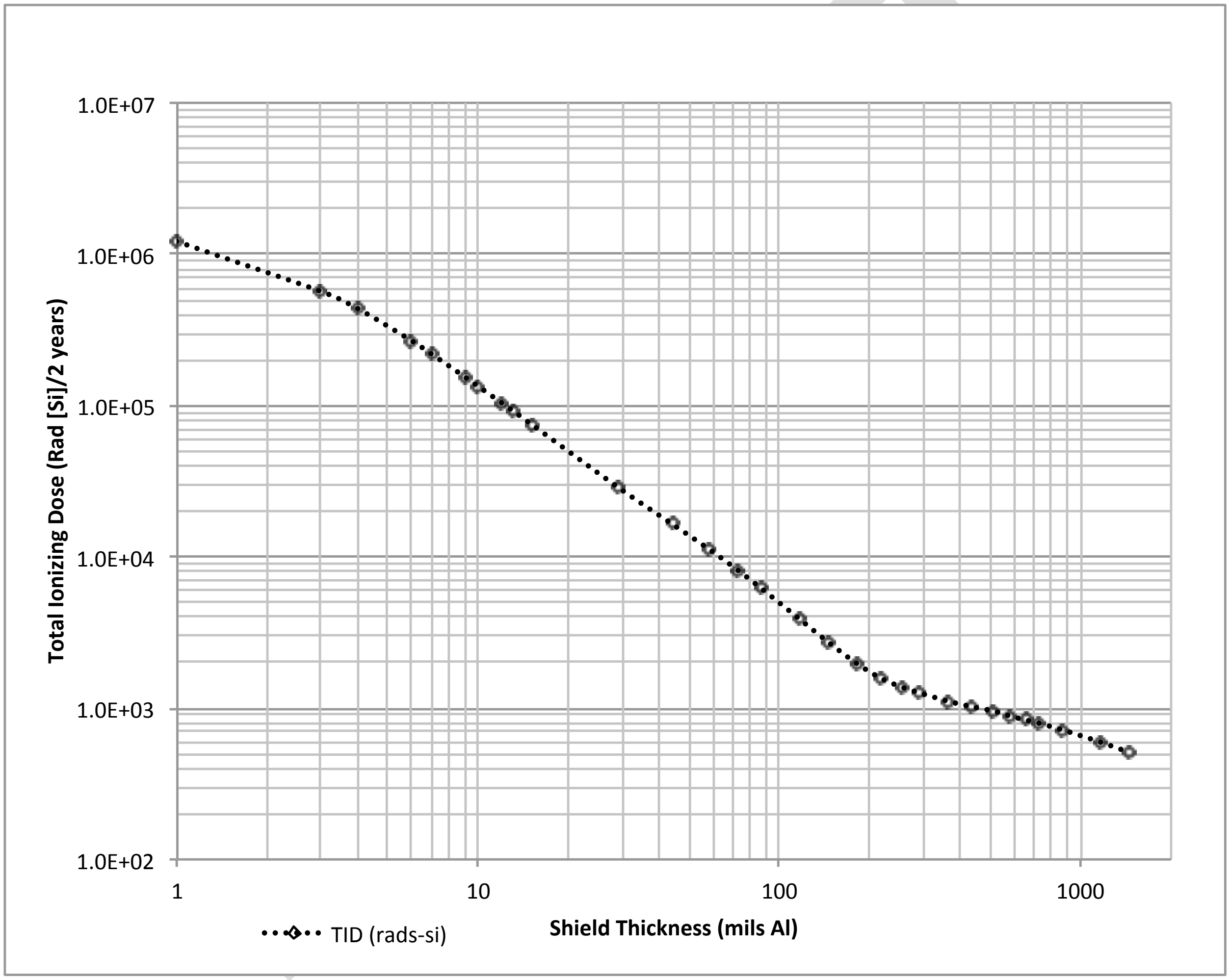

Figure 8-5: [LEO] TID versus Shielding Thickness

Table 8-9 shows the expected total ionizing dose for object in GEO, over the span of two years, while shielded by an aluminum spherical shell of a given thickness. Figure 8-6 plots the same data. The data contain no margin or uncertainty factors.

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Table 8-9: [GEO] Total Ionizing Dose Radiation Environment

Aluminum Shield Thickness [mil]	Total Dose [Rad]-Si
0	2.09E+08
10	2.62E+07
20	9.64E+06
30	4.78E+06
40	2.70E+06
50	1.60E+06
60	1.01E+06
70	6.60E+05
80	4.44E+05
90	3.19E+05
100	2.31E+05
110	1.69E+05
120	1.26E+05
130	9.37E+04
140	6.67E+04
150	5.26E+04
160	3.94E+04
170	2.87E+04
180	2.36E+04
190	1.88E+04
200	1.43E+04
210	1.17E+04
220	1.01E+04
230	8.57E+03
240	7.10E+03
250	5.96E+03
260	5.28E+03
270	4.63E+03
280	4.01E+03
290	3.41E+03
300	2.90E+03

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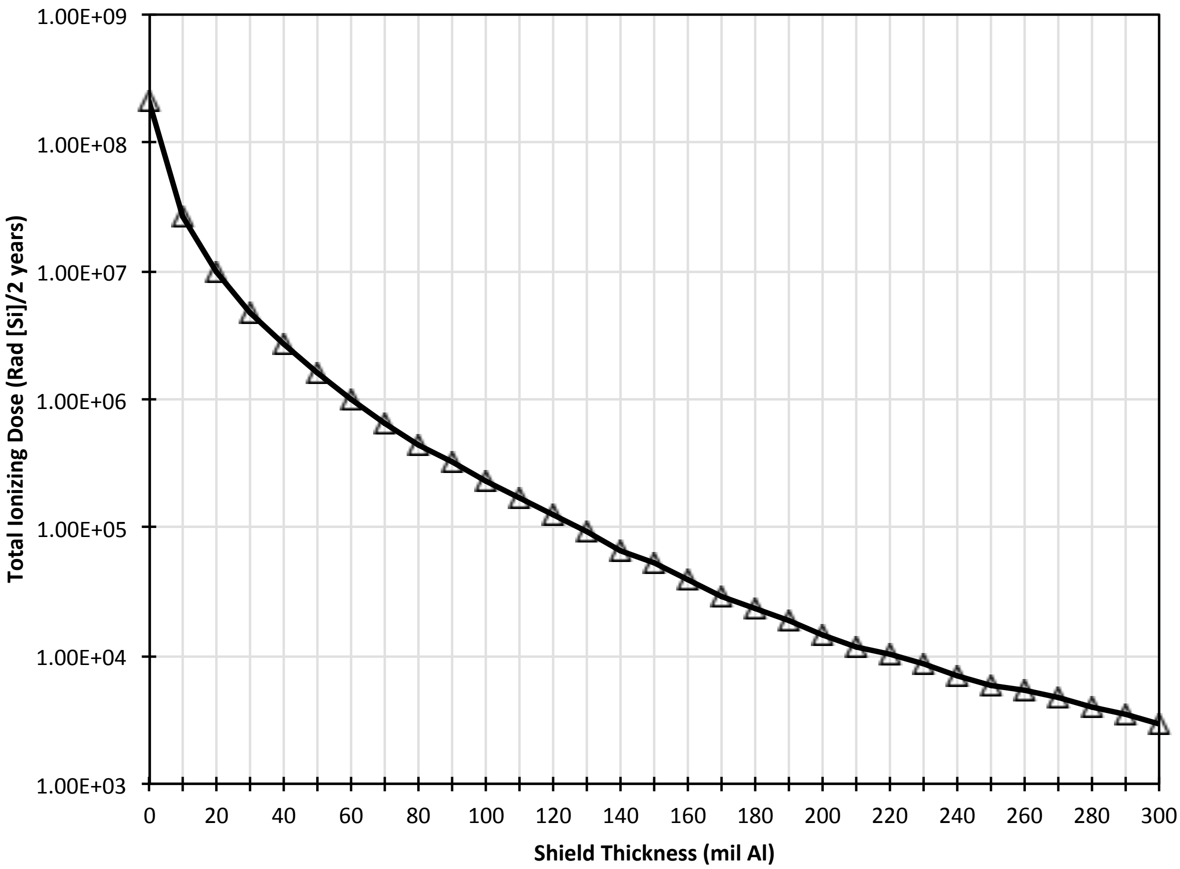

Figure 8-6: [GEO] TID versus Shielding Thickness

Rationale: Exposure to ionizing radiation degrades many materials and electronics in particular, and will require mitigation to ensure full instrument function over the design mission lifetime. Mitigation is typically achieved through application of the appropriate thickness of shielding. The LEO TID radiation environment is representative of exposure at an 813 km, sun-synchronous orbit. Analysis of dose absorption through shielding is based upon the SHIELDOSE2 model, which leverages NASA’s Radiation Belt Models, AE-8 and AP-8, and JPL’s Solar Proton Fluence Model. The GEO guideline is the all-satisfy strategy scenario, based upon CII analysis of the following sources of performance data: *CII RFI for GEO Hosted Payload Opportunities* responses and *The Radiation Model for Electronic Devices on GOES-R Series Spacecraft* (417-R-RPT-0027). The TID accrues as a constant rate and may be scaled for shorter and longer mission durations.

The LEO data represent conservative conditions for a specific orbit. While these data may envelop the TID environment of other LEO mission orbits (particularly those of lower altitude and inclination), Instrument Developers should analyze the TID environment for their Instrument’s specific orbit. Since TID environments are nearly equivalent within the GEO domain, these data likely envelop the expected TID environment for GEO Earth Science missions. The same caveat regarding Instrument Developer analysis of the TID environment also applies to the GEO domain.

8.5.6 [GEO] Instrument Interference

The Instrument should function according to specification in the operational environment when exposed to the particle fluxes defined by Table 8-10.

Rationale: The particle background causes increased noise levels in instruments and other electronics. No long term flux is included for solar particle events because of their short durations. This guidance is based upon “Long-term and worst-case particle fluxes in GEO behind 100 mils of aluminum shielding”, Table 4 of 417-R-RPT-0027.

Table 8-10: [GEO] Particle fluxes in GEO w/ 100 mils of Aluminum Shielding

Radiation:	Long-term flux [#/cm^2/s]	Worst-case flux [#/cm^2/s]
Galactic Cosmic Rays	2.5	4.6
Trapped Electrons	6.7×10^4	1.3×10^6
Solar Particle Events		2.0×10^5

8.5.7 Micrometeoroids

The Instrument Developer should perform a probability analysis to determine the type and amount of shielding to mitigate the fluence of micrometeoroids in the expected mission orbit over the primary mission.

Table 8-11 and Figure 8-7 provide a conservative micrometeoroid flux environment for both LEO and GEO.

Rationale: Impacts from micrometeoroids may cause permanently degraded performance or damage to the hosted payload instrument. This guidance provides estimates of the worst-case scenarios of micrometeoroid particle size and associated flux over the LEO and GEO domains. The data come from the Grün flux model assuming a meteoroid mean speed of 20 km/s. Of note, the most hazardous micrometeoroid environment in LEO is at an altitude of 2000 km. If a less conservative LEO environment is desired, the Instrument Developer should perform an analysis tailored to the risk tolerance.

Micrometeoroid and artificial space debris flux guidelines are separate due to the stability of micrometeoroid flux over time, compared to the increase of artificial space debris.

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Table 8-11: Worst-case Micrometeoroid Environment

Particle mass [g]	Particle diameter [cm]	Flux (particles/m^2/year) LEO	Flux (particles/m^2/year) GEO
1.00E-18	9.14E-07	1.20E+07	9.53E+06
1.00E-17	1.97E-06	1.75E+06	1.39E+06
1.00E-16	4.24E-06	2.71E+05	2.15E+05
1.00E-15	9.14E-06	4.87E+04	3.85E+04
1.00E-14	1.97E-05	1.15E+04	9.14E+03
1.00E-13	4.24E-05	3.80E+03	3.01E+03
1.00E-12	9.14E-05	1.58E+03	1.25E+03
1.00E-11	1.97E-04	6.83E+02	5.40E+02
1.00E-10	4.24E-04	2.92E+02	2.31E+02
1.00E-09	9.14E-04	1.38E+02	1.09E+02
1.00E-08	1.97E-03	5.41E+01	4.28E+01
1.00E-07	4.24E-03	1.38E+01	1.09E+01
1.00E-06	9.14E-03	2.16E+00	1.71E+00
1.00E-05	1.97E-02	2.12E-01	1.68E-01
1.00E-04	4.24E-02	1.50E-02	1.19E-02
1.00E-03	9.14E-02	8.65E-04	6.84E-04
1.00E-02	1.97E-01	4.45E-05	3.52E-05
1.00E-01	4.24E-01	2.16E-06	1.71E-06
1.00E+00	9.14E-01	1.02E-07	8.05E-08
1.00E+01	1.97E+00	4.72E-09	3.73E-09
1.00E+02	4.24E+00	2.17E-10	1.72E-10

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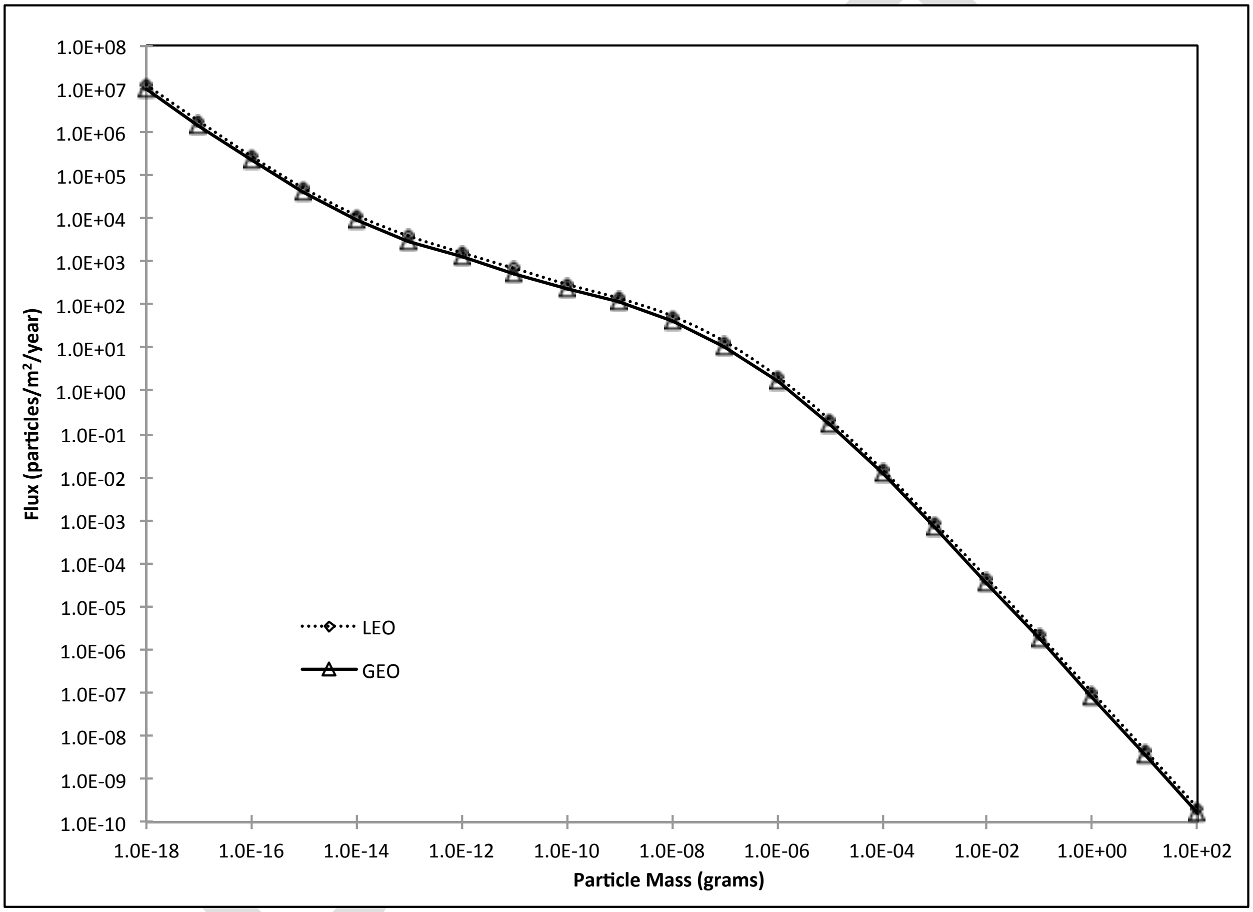


Figure 8-7: Worst-case Micrometeoroid Environment

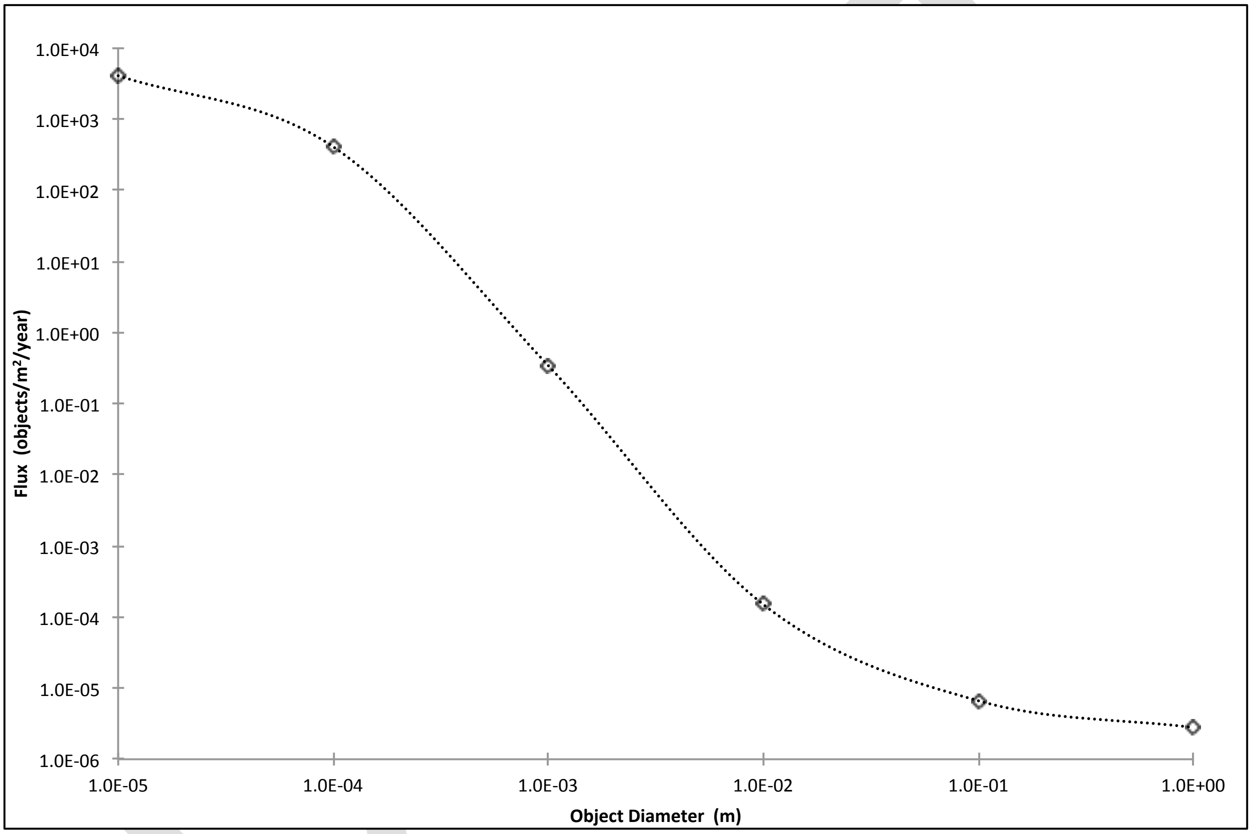
8.5.8 <u>Artificial Space Debris</u>

The Instrument Developer should perform a probability analysis to determine the type and amount of shielding to mitigate the fluence of artificial space debris in the expected mission orbit over the primary mission.

Table 8-12, Figure 8-8, Table 8-13, and Figure 8-9 provide conservative artificial space debris flux environments for both LEO and GEO.

Table 8-12: [LEO] Worst-case Artificial Space Debris Environment

Object Size [m]	Flux [objects/m^2/year]	Object Velocity [km/s]
1.00E-05	4.14E+03	12.02
1.00E-04	4.10E+02	9.25
1.00E-03	3.43E-01	10.63
1.00E-02	1.50E-04	10.53
1.00E-01	6.64E-06	9.10
1.00E+00	2.80E-06	9.34
	Average Velocity:	10.15

1062 

1063 **Figure 8-8: [LEO]: Worst-case Artificial Space Debris Environment**

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Table 8-13: [GEO] Worst-case Artificial Space Debris Environment

Object Diameter [m]	Flux [objects/m²/year]	Object Diameter [m]	Flux [objects/m²/year]	Object Diameter [m]	Flux [objects/m²/year]
1.00000E-03	2.08800E-05	2.06200E-02	1.56300E-08	4.25179E-01	3.93000E-09
1.14100E-03	1.58800E-05	2.35200E-02	1.40200E-08	4.84969E-01	3.89700E-09
1.30100E-03	9.74700E-06	2.68270E-02	1.13500E-08	5.53168E-01	3.85700E-09
1.48400E-03	6.06200E-06	3.05990E-02	1.02900E-08	6.30957E-01	3.83000E-09
1.69300E-03	4.70300E-06	3.49030E-02	9.74100E-09	7.19686E-01	3.81700E-09
1.93100E-03	3.38900E-06	3.98110E-02	8.92500E-09	8.20891E-01	3.76600E-09
2.20200E-03	2.32700E-06	4.54090E-02	8.07400E-09	9.36329E-01	3.75200E-09
2.51200E-03	1.55700E-06	5.17950E-02	7.06300E-09	1.06800E+00	3.73800E-09
2.86500E-03	1.10200E-06	5.90780E-02	6.36200E-09	1.21819E+00	3.73800E-09
3.26800E-03	7.81600E-07	6.73860E-02	5.88900E-09	1.38949E+00	3.73800E-09
3.72800E-03	5.16800E-07	7.68620E-02	5.52200E-09	1.58489E+00	3.73800E-09
4.25200E-03	3.73600E-07	8.76710E-02	5.30700E-09	1.80777E+00	3.73800E-09
4.85000E-03	2.88600E-07	1.00000E-01	4.91200E-09	2.06199E+00	3.38500E-09
5.53200E-03	2.15600E-07	1.14062E-01	4.66500E-09	2.35195E+00	3.38500E-09
6.31000E-03	1.60200E-07	1.30103E-01	4.56000E-09	2.68270E+00	3.38500E-09
7.19700E-03	1.20300E-07	1.48398E-01	4.39400E-09	3.05995E+00	3.38000E-09
8.20900E-03	8.21500E-08	1.69267E-01	4.27400E-09	3.49025E+00	3.37800E-09
9.36300E-03	6.42500E-08	1.93070E-01	4.18300E-09	3.98107E+00	1.95200E-09
1.06800E-02	5.00200E-08	2.20220E-01	4.14700E-09	4.54091E+00	1.95000E-09
1.21820E-02	4.05400E-08	2.51189E-01	4.08200E-09	5.17948E+00	1.94900E-09
1.38950E-02	3.00300E-08	2.86512E-01	4.02900E-09	5.90784E+00	1.94800E-09
1.58490E-02	2.36300E-08	3.26803E-01	3.99300E-09	6.73863E+00	1.94800E-09
1.80780E-02	1.92000E-08	3.72759E-01	3.96000E-09	7.68625E+00	1.36900E-13
				Average Velocity (km/s)	1.3333

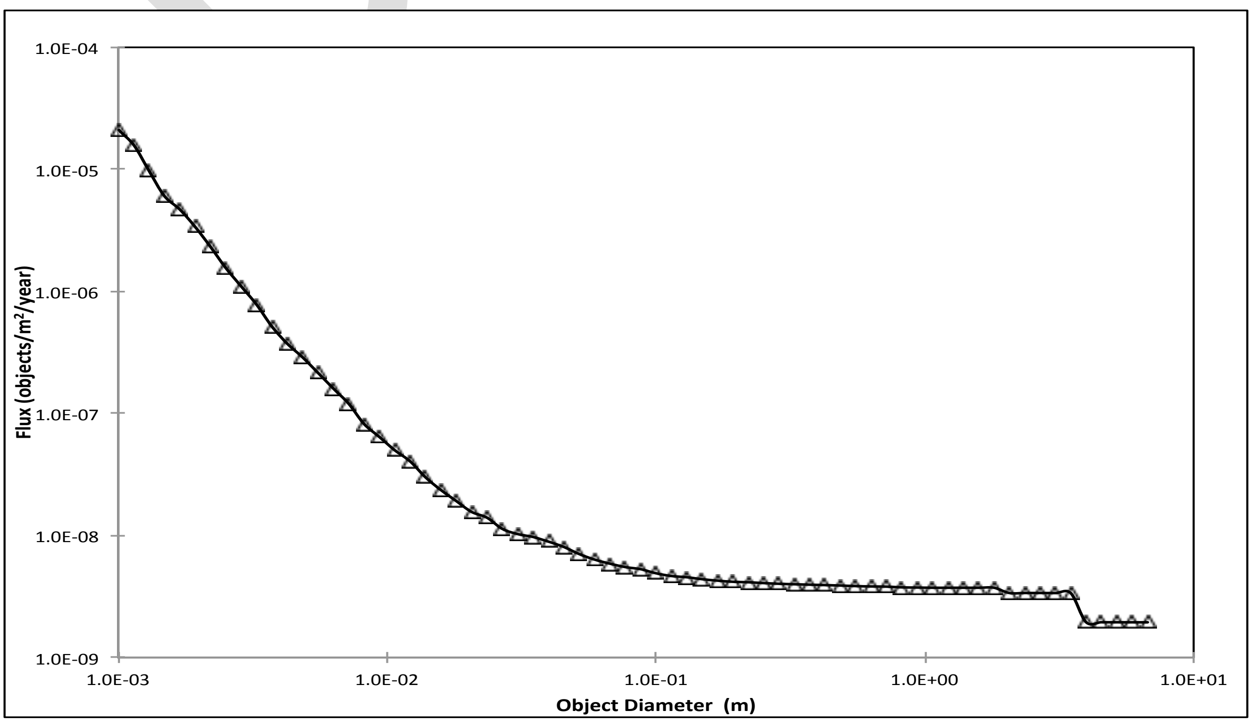

Figure 8-9: [GEO] Worst-case Artificial Space Debris Environment

Rationale: Impacts from artificial space debris may permanently degrade performance or damage the Instrument. This guidance estimates the maximum artificial space debris flux and impact velocities an Instrument can expect to experience for both LEO and GEO domains during the Calendar Year 2015 epoch. Expected artificial space debris flux increases over time as more hardware is launched into orbit.

The LEO analysis covers altitudes from 200 to 2000 km and orbital inclinations between 0 and 180 degrees. The ORDEM2000 model, developed by the NASA Orbital Debris Program Office at Johnson Space Center, is the source of the data.

Based upon analysis of ESA's 2009 MASTER (Meteoroid and Space Debris Environment) model, the GEO guidance aggregates the maximum expected artificial space debris flux, sampled at 20° intervals around the GEO belt.

Micrometeoroid and artificial space debris flux guidelines are listed separately due to the stability of micrometeoroid flux over time, compared to the increase of artificial space debris.

8.5.9 Atomic Oxygen Environment

The Instrument should function according to its specifications following exposure to the atomic oxygen environment, based on its expected mission orbit, for the duration of the Instrument primary mission.

Rationale: Exposure to atomic oxygen degrades many materials and requires mitigation to ensure full Instrument function over the design mission lifetime. Atomic oxygen levels in LEO are significant and may be derived using the Figure 8-10, which estimates the atomic oxygen flux, assuming an orbital velocity of 8 km/sec, for a range of LEO altitudes over the solar cycle inclusive of the standard atmosphere. Atomic oxygen levels in GEO are negligible and are only significant for GEO-bound Instruments that spend extended times in LEO prior to GEO transfer.

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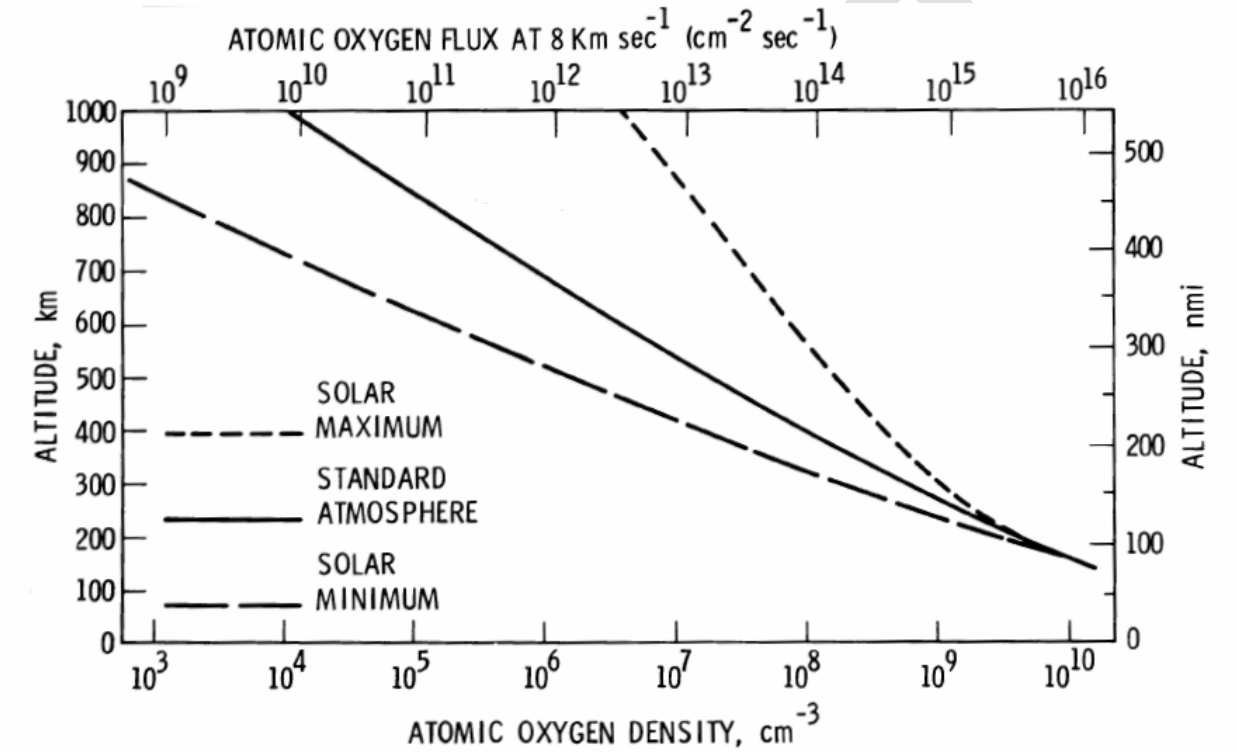

Figure 8-10: Atmospheric Atomic Oxygen density in Low Earth Orbit (Figure 2 from de Rooij 2000)

8.5.10 Electromagnetic Interference & Compatibility Environment

The Instrument should function according to its specification following exposure to the Electromagnetic Interference and Electromagnetic Compatibility (EMI/EMC) environments as defined in the applicable sections of MIL-STD-461.

Please note that the environments defined in MIL-STD-461 may be tailored in accordance with the Host Spacecraft, launch vehicle and launch range requirements.

Rationale: Exposure of the hosted payload instrument to electromagnetic fields may induce degraded performance or damage in the instrument electrical and/or electronic subsystems. The application of the appropriate environments as described in the above noted reference and in accordance with those test procedures defined in, or superior to, MIL-STD-461 or MIL-STD-462, will result in an instrument that is designed and verified to assure full instrument function in the defined EMI/EMC environments.

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9.0 REFERENCE MATERIAL / BEST PRACTICES

9.1 Data Interface Reference Material / Best Practices

9.1.1 <u>*CCSDS Data Transmission*</u>

The Instrument should transmit and receive all packet data using Consultative Committee for Space Data Systems (CCSDS) primary and secondary headers for packet sequencing and control.

Rationale: The use of CCSDS packets for data communication is common practice across aerospace flight and ground data systems.

9.1.2 <u>*Flight Software Update*</u>

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.

9.1.3 <u>*Flight Software Update (Partial)*</u>

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.

9.1.4 <u>Use of Preexisting Communication Infrastructure</u>

Instrument Developers should consider utilizing the communication infrastructure provided by the Host Spacecraft and Satellite Operator for all of the Instrument's space-to-ground communications needs.

Rationale: The size, mass, and power made available to the Instrument may not simultaneously accommodate a scientific Instrument as well as communications terminals, antennas, and other equipment. Additionally, the time required for the Instrument Developer to apply for and secure a National Telecommunications and Information Administration (NTIA) Spectrum Planning Subcommittee (SPS) Stage 4 (operational) Approval to transmit on a particular radio frequency band may exceed the schedule available, given the constraints as a hosted payload. A Satellite Operator will have already initiated the spectrum approval process that would cover any data the Instrument transmits through the Host Spacecraft. NPR 2570.1B, *NASA Radio Frequency (RF) Spectrum Management Manual*, details the spectrum approval process for NASA missions.

9.2 Electrical Power Interface Reference Material / Best Practices

9.2.1 <u>Discussion</u>

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 an HPO and launch vehicle and to progress the design of the host payload accommodations. The major sources from which these best practices were developed are the *General Interface Requirements Document (GIRD) for EOS Common Spacecraft/Instruments*, *Electrical Grounding Architecture* *for Unmanned Spacecraft,* (NASA-HDBK-4001), *Mitigating In-Space Charging Effects – A Guideline*, (NASA-HDBK-4002) and *Electrical Bonding for Launch Vehicles, Spacecraft, Payloads and Flight Equipment* (NASA-STD-4003).

Note: This section assumes that the Host Spacecraft will provide access to its Electrical Power System using the interface defined in Section 5.1.

9.2.2 Electrical Interface Definitions

9.2.2.1 *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.

Rationale: This describes the maximum nominal rate of change for instrument electrical current to bound nominal and anomalous behavior.

9.2.2.2 *Power Bus Isolation*

All Instrument power buses (both operational and survival) should be electrically isolated from each other and from the chassis.

Rationale: Circuit protection and independence.

9.2.2.3 *Power Bus Returns*

All Instrument power buses (both operational and survival heater) should have independent power returns.

Rationale: Circuit protection and independence.

9.2.3 Survival Heaters.

9.2.3.1 *Survival Heater Power Bus Circuit Failure*

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

Rationale: A stuck-on survival heater could lead to excessive power draw and/or over-temperature events in the Instrument or Host Spacecraft. This is normally accomplished by using series-redundant thermostats in each survival heater circuit.

9.2.3.2 *Survival Heater Power Bus Heater Type*

The Instrument should use only resistive heaters (and associated thermal control devices) to maintain the Instrument at survival temperature when the main power bus is disconnected from the Instrument.

Rationale: This preserves the survival heater power bus for exclusive use of resistive survival heaters, whose function is to maintain the Instrument at a minimum turn-on temperature when the Instrument Power Buses are not energized.

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9.2.3.3 *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.

Rationale: This precludes excessive power draw and/or over-temperature events in the Instrument or Host Spacecraft. This is normally accomplished via the application of thermostats with different set points in each redundant survival heater circuit.

9.2.4 <u>Voltage and Current Transients</u>

9.2.4.1 *Low Voltage Detection*

A voltage excursion that causes the spacecraft Primary Power Bus to drop below 22 VDC 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.

Rationale: Bounds nominal and anomalous design conditions. Describes "typical" spacecraft CONOPS to the noted anomaly for application to design practice.

9.2.4.2 *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.

Rationale: Describes a "standard" design practice.

9.2.4.3 *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.

Rationale: the Instrument needs to be able to tolerate appropriate electrical transients without affecting the Host Spacecraft.

9.2.4.4 *Abnormal Transients Undervoltage*

An abnormal undervoltage transient event is defined as a transient decrease in voltage on the Power 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.

9.2.4.5 *Abnormal Transients Tolerance*

The Instrument should ensure that overstress does not occur to the unit during an undervoltage event.

Rationale: The Instrument needs to tolerate the abnormal voltage transients, which can be expected to occur throughout its mission lifetime.

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9.2.4.6 *Abnormal Transients Recovery*

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

Rationale: The Instrument needs to tolerate the abnormal voltage transients, which can be expected to occur throughout its mission lifetime.

9.2.4.7 *Abnormal Transients Overvoltage*

An overvoltage transient event is defined as an increase in voltage on the Power 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.

Rationale: A necessary definition of an Abnormal Transient Overvoltage

9.2.4.8 *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.

Rationale: Bounds nominal and anomalous behavior.

9.2.4.9 *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.

Rationale: Bounds nominal and anomalous behavior.

9.2.4.10 *Instrument In-rush Current after 10 µs*

After 10 µs, the transient current peak should not exceed three times the maximum steady state current.

Rationale: Bounds nominal and anomalous behavior.

9.2.4.11 *Instrument Steady State Operation*

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

Rationale: Bounds nominal and anomalous behavior with a maximum transient duration of 50 ms.

9.2.4.12 *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.

Rationale: Bounds nominal behavior.

9.2.4.13 *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.

Rationale: Describes design “standard practice.”

9.2.4.14 *Reflected Ripple Current – Mode Changes*

The load current ripple due to motor rotation speed mode changes should not exceed 2 times the steady state current during the period of the motor spin-up or spin-down.

Rationale: Bounds nominal behavior.

9.2.4.15 *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.

Rationale: Bounds nominal behavior.

9.2.4.16 *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.

Rationale: Bounds nominal behavior.

9.2.5 Overcurrent Protection

9.2.5.1 *Overcurrent Protection Definition*

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

Rationale: Describes conditions necessary for inclusion in the “standard” design practice.

9.2.5.2 *Overcurrent Protection – Harness Compatibility*

Harness wire sizes should be consistent with overcurrent protection device sizes and de-rating factors.

Rationale: Describes a “standard” design practice.

9.2.5.3 *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.

The EICD will document the type, size, and characteristics of the overcurrent protection devices.

Rationale: Describes “standard practice” EICD elements.

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9.2.5.4 *Instrument Overcurrent Protection*

All Instrument overcurrent protection devices should be accessible at the Spacecraft integration level without any disassembly of the Instrument.

Rationale: Accessible overcurrent protection devices allow Systems Integrator technicians to more easily restore power to the Instrument in the event of an externally-induced overcurrent.

9.2.5.5 *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.

Rationale: This preserves redundancy by keeping faulty power circuits from impacting alternate power sources.

9.2.5.6 *Testing of Instrument High-Voltage Power Supplies in Ambient Conditions*

Instrument high-voltage power supplies should operate nominally in ambient atmospheric conditions.

Rationale: This allows simplified verification of the high-voltage power supplies.

If the high-voltage power supplies cannot operate nominally in ambient conditions, then the Instrument should enable a technician to manually disable the high-voltage power supplies.

Rationale: This allows verification of the Instrument by bypassing the HV power supplies that do not function in ambient conditions.

9.2.5.7 *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.

Rationale: This prevents the power supply’s discharge from damaging the Host Spacecraft or other payloads.

9.2.6 Connectors

The following best practices apply to the selection and use of all interface connectors.

9.2.6.1 *Instrument Electrical Power System Connector and Harnessing*

The Instrument electrical power system harnessing and connectors should conform to GSFC-733-HARN, IPC J-STD-001ES and NASA-STD-8739.4.

Rationale: Describes the appropriate design practices for all Instrument electrical power connections and harnessing.

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9.2.6.2 *Connector Savers*

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

Rationale: This practice serves to preserve the number of mate/de-mate cycles any particular flight connector experiences. Mate/de-mate cycles are a connector life-limiting operation. This practice also protects flight connects form damage during required connector mate/de-mate operations.

9.2.6.3 *Connector Separation*

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

The Instrument should physically separate the electric interfaces for each of the following functions using distinct connectors:

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

Rationale: A “standard” design practice to preclude mismating and to simplify test and anomaly resolution.

9.2.6.4 *Command and Telemetry Returns*

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

Rationale: A “standard” design practice to simplify testing and anomaly resolution.

9.2.6.5 *Connector Usage and Pin Assignments*

Harness side power connectors and all box/bracket-mounted connectors supplying power to other components should have female contacts.

Rationale: Unexposed power supply connector contacts preclude arcing, mismating, and contact shorting.

9.2.6.6 *Connector Function Separation*

Incompatible functions should be physically separated.

Rationale: A “standard” design practice to ensure connector conductor self-compatibility that precludes arcing and inductive current generation.

9.2.6.7 *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.

Rationale: A “standard” design practice.

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9.2.6.8 *Connector Access*

At least 50 mm of clearance should exist around the outside of mated connectors.

Rationale: Ensures the ability to perform proper connector mate/de-mate operations.

9.2.6.9 *Connector Engagement*

Connectors should be mounted to ensure straight and free engagement of the contacts.

Rationale: This precludes mismating connectors.

9.2.6.10 *Power Connector Type*

The Instrument power connectors should be space-flight qualified MIL-DTL-24308, Class M, Subminiature Rectangular connectors with standard density size 20 crimp contacts and conform to GSFC S-311-P-4/09.

Rationale: Connector sizes and types selected based upon familiarity, availability, and space flight qualification.

9.2.6.11 *Power Connector Size and Conductor Gauge*

The Instrument power connectors should be 20 AWG, 9 conductor (shell size 1) or 15 conductor (shell size 2) connectors.

Rationale: Application of stated design practices to the CII instrument power bus connectors.

9.2.6.12 *Power Connector Pin Out*

The Instrument power connectors should utilize the supply and return pin outs defined in Table 9-1 and identified in Figure 9-1 thru Figure 9-3. *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 9-1: Instrument Power Connector Pin Out Definition

Power Bus	Circuit	Supply Conductor Position	Return Conductor Position
#1	A & B	14,15,16,23,24,25	1,2,3,11,12,13
#2	A & B	9,10,11,13,14,15	1,2,3,6,7,8
Survival Heater	A & B	6,7,8,9	1,2,4,5

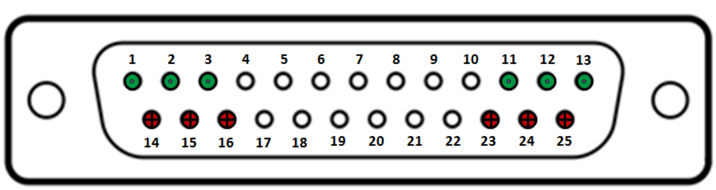

Figure 9-1: Instrument Side Power Bus #1 Circuit A & Circuit B

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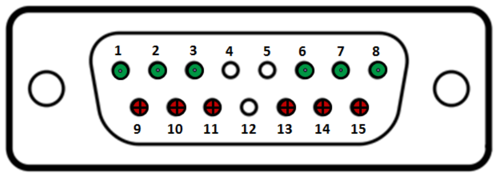

Figure 9-2: Instrument Side Power Bus #2 Circuit A & Circuit B

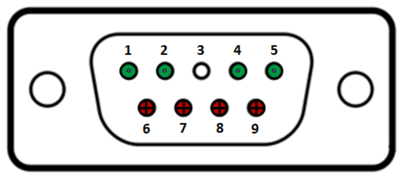

Figure 9-3: Instrument Side Survival Heater Power Bus Circuit A & Circuit B

9.2.6.13 *SpaceWire Connectors and Harnessing*

The Instrument SpaceWire harnessing and connectors should conform to ECSS-E-ST-50-12C.

Rationale: Describes the appropriate design practice for all SpaceWire connections and harnessing.

9.2.6.14 *Power Connector Provision*

The Instrument Provider should furnish all instrument power mating connectors (Socket Side) to the Spacecraft Manufacturer for interface harness fabrication.

Rationale: Describes the appropriate design practice for all SpaceWire connectors.

9.2.6.15 *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.

Rationale: Application of the conductor size and type selected for the CII instrument power bus connectors to the corresponding instrument power connectors.

9.2.6.16 *Power Connector Keying*

The instrument power connectors should be keyed as defined in Figure 9-4.

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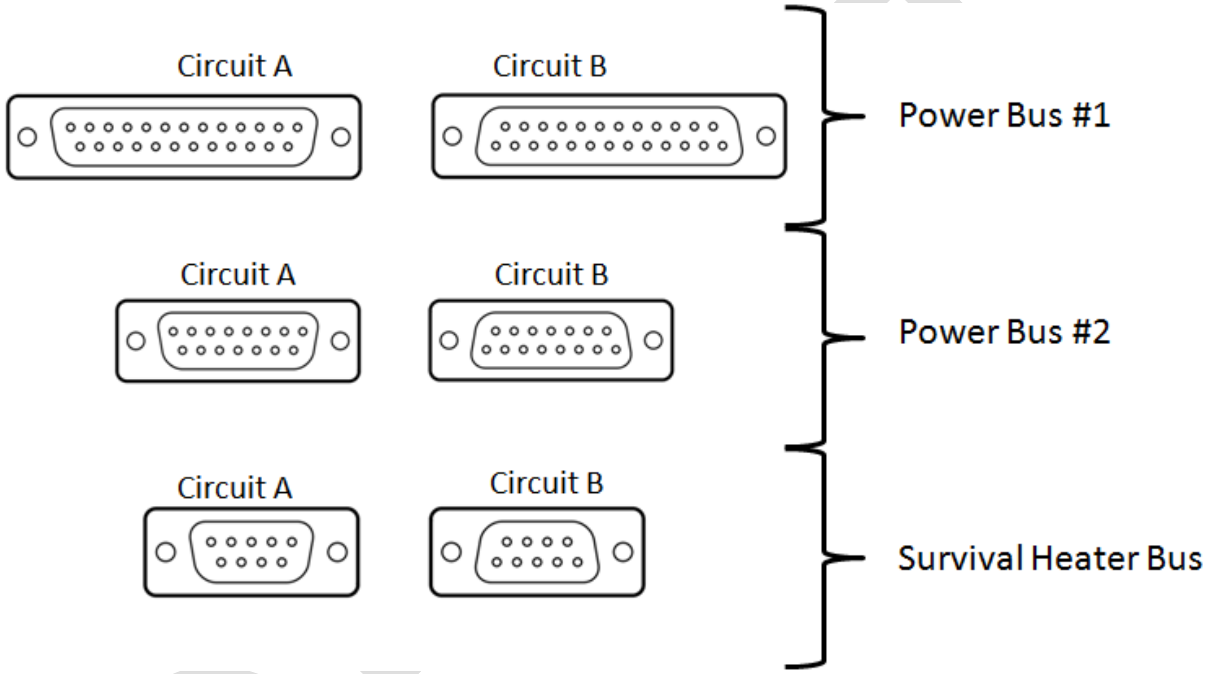

Figure 9-4: Power Connector Keying

9.2.6.17 *Connector Type Selection*

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

Rationale: Utilizing the GSPC PPL simplifies connector selection, since all of its hardware is spaceflight qualified.

9.2.6.18 *Flight Plug Installation*

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

Rationale: Ensures necessary access.

9.2.6.19 *Test Connector Location and Types*

Test connector and coupler ports should be accessible without disassembly throughout integration of the Instrument and Host Spacecraft.

Rationale: This reduces the complexity and duration of integrated testing and simplifies preflight anomaly resolution.

9.3 Mechanical Interface Reference Material / Best Practices

9.3.1 <u>Minimum Fixed-Base Frequency</u>

The Instrument should have a fixed based frequency greater than 50 Hz.

Rationale: This minimum fixed-based frequency exceeds the composite guidance of publically available Launch Vehicle Payload Planner's Guidebooks as applicable to primary spacecraft structures operating in both LEO and GEO regimes. To some extent, the Instrument will affect the spacecraft frequency depending on the payload's mass and mounting location. Spacecraft Manufacturers may negotiate for a greater fixed-based frequency for hosted payloads until the maturity of the instrument can support Coupled Loads Analysis.

9.3.2 Mass Centering

The Instrument center of mass should be less than 5 cm radial distance from the $Z_{instrument}$ axis, defined as the center of the Instrument mounting bolt pattern.

Rationale: Engineering analysis determined guideline Instrument mass centering parameters based on comparisons to spacecraft envelope in the *STP-SIV Payload User's Guide*.

The Instrument center of mass should be located less than half of the Instrument height above the Instrument mounting plane.

Rationale: Engineering analysis determined guideline Instrument mass centering parameters based on comparisons to spacecraft envelope in the *STP-SIV Payload User's Guide*.

9.3.3 Documentation of Mechanical Properties

9.3.3.1 *Envelope*

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

Rationale: Defines the actual maximum envelope within which the instrument resides.

9.3.3.2 *Mass*

[LEO] The MICD will document the mass of the Instrument, measured to ± 1%.

[GEO] The MICD will document the mass of the Instrument, measured to less than 0.2%.

Rationale: To ensure that accurate mass data is provided for analytic purposes.

9.3.3.3 *Center of Mass*

[LEO] The MICD will document the launch and or-orbit centers of mass of each Instrument, references to the Instrument coordinate axes and measured to ± 5 mm.

[GEO] The MICD will document the launch and on-orbit centers of mass of each Instrument, referenced to the Instrument coordinate axes and measured to ± 1 mm.

Rationale: To ensure that accurate CG data is provided for analytic purposes.

9.3.3.4 *Moment of Inertia*

[LEO] The MICD will document the moments of inertia, measured to less than 10%

[GEO] The MICD will document the moments of inertia, measured to less than 1.5%.

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Rationale: To ensure that accurate moments of inertia data is provided for analytic purposes.

9.3.3.5 *Constraints on Moments of Inertia*

The MICD will document the constraints to the moments and products of inertia available to the Instrument.

Rationale: To define the inertial properties envelope within which the Instrument may operate and not adversely affect spacecraft and primary instrument operations.

9.3.4 <u>Dynamic Properties</u>

9.3.4.1 *Documentation of Dynamic Envelope or Surfaces*

The MICD will document the initial and final configurations, as well as the swept volumes of any mechanisms that cause a change in the external envelope or external surfaces of the Instrument.

Rationale: To define variations in envelope caused by deployables.

9.3.4.2 *Documentation of Dynamic Mechanical Elements*

The MICD will document the inertia variation of the Instrument due to movable masses, expendable masses, or deployables.

Rationale: Allows spacecraft manufacturer to determine the impact of such variations on spacecraft and primary payload.

9.3.4.3 *Caging During Test and Launch Site Operations*

Instrument mechanisms that require caging during test and launch site operations should cage when remotely commanded.

Rationale: To allow proper instrument operation during integration and test.

Instrument mechanisms that require uncaging during test and launch site operations should uncage when remotely commanded.

Rationale: To allow proper instrument operation during integration and test.

Instrument mechanisms that require caging during test and launch site operations should cage when accessible locking devices are manually activated.

Rationale: To allow proper instrument operation during integration and test.

Instrument mechanisms that require uncaging during test and launch site operations should uncage when accessible unlocking devices are manually activated.

Rationale: To allow proper instrument operation during integration and test.

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9.3.5 <u>Instrument Mounting</u>

9.3.5.1 *Documentation of Mounting*

The MICD will document the mounting interface, method, and geometry, including ground strap provisions and dimensions of the holes for mounting hardware.

Rationale: To ensure no ambiguity of mounting interface between instrument and spacecraft.

9.3.5.2 *Documentation of Instrument Mounting Location*

The MICD will document the mounting location of the Instrument on the Host Spacecraft.

Rationale: To ensure no ambiguity of mounting location on spacecraft.

9.3.5.3 *Metric Units*

The MICD will specify whether mounting fasteners will conform to SI or English unit standards.

Rationale: Metric hardware are not exclusively used industry wide. Choice of unit system likely will be set by spacecraft manufacturer.

9.3.5.4 *Documentation of Finish and Flatness Guidelines*

The MICD will document finish and flatness guidelines for the mounting surfaces.

Rationale: To ensure no ambiguity of finish and flatness requirements at instrument interface.

9.3.5.5 *Drill Template Usage*

The MICD will document the drill template details and serialization.

Rationale: Drill template details will be on record.

The Instrument Developer should drill spacecraft and test fixture interfaces using the MICD defined template.

Rationale: A common drill template will ensure proper alignment and repeatability of mounting holes.

9.3.5.6 *Kinematic Mounts*

The Instrument Provider should provide all kinematic mounts.

Rationale: If the instrument requires kinematic mounts, they should be the responsibility of the instrument provider due to their knowledge of the instrument performance requirements.

9.3.5.7 *Fracture Critical Components of Kinematic Mounts*

Kinematic mounts should comply with all analysis, design, fabrication, and inspection requirements associated with fracture critical components as defined by NASA-STD-5019.

Rationale: Kinematic mount failure is a potential catastrophic hazard to the Instrument and the Host Spacecraft.

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9.3.6 Instrument Alignment

9.3.6.1 *Documentation of Coordinate System*

The MICD will document the Instrument Reference Coordinate Frame.

Rationale: To ensure there is no ambiguity between instrument and spacecraft manufacturers regarding the Instrument Reference Coordinate System.

9.3.6.2 *Instrument Interface Alignment Cube*

If the Instrument has critical alignment requirements, the Instrument should contain an Interface Alignment Cube (IAC), an optical cube that aligns with the Instrument Reference Coordinate Frame.

The Spacecraft should contain an IAC that aligns with the Instrument Reference Coordinate Frame.

Rationale: To aid in proper alignment of the instrument to the spacecraft during Integration and Test.

9.3.6.3 *Interface Alignment Cube Location*

The Instrument Developer should mount the IAC such that it is visible at all stages of integration with the Spacecraft from at least two orthogonal directions.

Rationale: Observation of IAC from at least two directions is required for alignment.

9.3.6.4 *Interface Alignment Cube Documentation*

The MICD will document the location of all optical alignment cubes on the Instrument.

Rationale: To have a record of the IAC locations.

9.3.6.5 *Instrument Boresight*

The Instrument Developer should measure the alignment angles between the IAC and the Instrument boresight.

Rationale: Since this knowledge is critical to the instrument provider they should be responsible for taking the measurement.

The MICD will document the alignment angles between the IAC and the Instrument boresight.

Rationale: To record the actual alignment angle in case it is needed for later analysis.

9.3.6.6 *Pointing Accuracy, Knowledge, and Stability*

The MICD will document the Host Spacecraft's required pointing accuracy, knowledge, and stability capabilities in order for the Instrument to meet its operational requirements.

Rationale: To establish that spacecraft's pointing accuracy, knowledge and stability specifications meet requirements of instrument operation.

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9.3.7 Integration and Test

9.3.7.1 *Installation/Removal*

The Instrument should be capable of being installed in its launch configuration without disturbing the primary payload.

The Instrument should be capable of being removed in its launch configuration without disturbing the primary payload.

Rationale: Primary payload safety.

9.3.7.2 *Mechanical Attachment Points*

The Instrument should provide mechanical attachment points that will be used by a handling fixture during integration of the instrument.

Rationale: The handling fixtures will be attached to the Instrument while in the Integration and Test environment.

The MICD will document details of the mechanical attachment points used by the handling fixture.

Rationale: To ensure handling fixture attachment points are properly recorded.

9.3.7.3 *Load Margins*

Handling and lifting fixtures should function according to their operational specifications at five (5) times limit load for ultimate.

Handling and lifting fixtures should function according to their operational specifications at three (3) times limit load for yield.

Handling fixtures should be tested to two (2) times working load.

Rationale: All three load margins maintain personnel and instrument safety.

9.3.7.4 *Responsibility for Providing Handling Fixtures*

The Instrument Provider should provide proof-tested handling fixtures for each component with mass in excess of 16 kg.

Rationale: This guideline protects personnel safety.

9.3.7.5 *Accessibility of Red Tag Items*

All items intended for pre-flight removal from the Instrument should be accessible without disassembly of another Instrument component.

Rationale: Instrument safety.

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9.3.7.6 *Marking and Documentation of Test Points and Test Guidelines*

All test points and I&T interfaces on the Instrument should be visually distinguishable from other hardware components to an observer standing 4 feet away.

Rationale: Clear visual markings mitigate the risk that Integration and test personnel will attempt to connect test equipment improperly, leading to Instrument damage. Four feet exceeds the length of most human arms and ensures that the technician would see any markings on hardware he intends to connect test equipment to.

The MICD will document all test points and test guidelines.

Rationale: To ensure no ambiguity of Integration and Test interfaces and test points and to aide in developing I&T procedures.

9.3.7.7 *Orientation Constraints During Test*

The MICD will document instrument mechanisms, thermal control, or any exclusions to testing and operations related to orientations.

Rationale: This documents any exceptions to the 1g functionality described in section 6.2.1

9.3.7.8 *Temporary Items*

All temporary items to be removed following test should be visually distinguishable from other hardware components to an observer standing 4 feet away.

Rationale: Any preflight removable items need to be obvious to casual inspection to mitigate the risk of them causing damage or impairing spacecraft functionality during launch/operations.

The MICD will document all items to be installed prior to or removed following test and all items to be installed or removed prior to flight.

Rationale: To ensure no ambiguity of installed and/or removed items during Integration and Test through documentation.

9.3.7.9 *Temporary Sensors*

The Instrument should accommodate temporarily installed sensors and supporting hardware to support environmental testing.

Rationale: To facilitate environmental testing.

Example sensors include acceleration sensors and thermal monitors.

9.3.7.10 *Captive Hardware*

The Instrument Developer should utilize captive hardware for all items planned to be installed, removed, or replaced during integration, except for Instrument mounting hardware and MLI.

Rationale: Captive hardware reduces the danger to the Host Spacecraft, Instrument, and personnel from fasteners dropped during integration.

9.3.7.11 *Venting Documentation*

The MICD will document the number, location, size, vent path, and operation time of Instrument vents.

Rationale: This eliminates ambiguity regarding venting the Instrument and how it may pertain to the Host Spacecraft and primary instrument operations.

9.3.7.12 *Purge Documentation*

The MICD will document Instrument purge guidelines, including type of purge gas, flow rate, gas purity specifications, filter pore size, type of desiccant (if any), and tolerable interruptions in the purge (and their duration).

Rationale: This ensures compatibility of instrument purging procedures with respect to the Host Spacecraft and primary instrument.

9.3.7.13 *Combined Structural Dynamics Analysis Results*

The Spacecraft Manufacturer should furnish the combined structural dynamics analysis results to the respective Instrument Provider.

Rationale: To ensure the combined structural dynamics does not impede instrument operations.

9.3.7.14 *Non-Destructive Evaluation*

Kinematic mount flight hardware should show no evidence of micro cracks when inspected using Non-Destructive Evaluation (NDE) techniques following proof loading.

Rationale: To ensure kinematic mounts meet load requirements without damage.

The MICD will document the combined structural dynamics analysis.

Rationale: Record maintenance.

9.4 Thermal Interface Reference Material / Best Practices

9.4.1 Heat Management Techniques

9.4.1.1 *Heat Transfer Hardware*

The Instrument Developer should consider implementing heat pipes and high thermal conductivity straps to transfer heat within the Instrument.

Rationale: A spacecraft would likely more easily accommodate an instrument whose thermal design is made more flexible by the inclusion of heat transfer hardware.

9.4.1.2 *Survivability at Very Low Temperature*

The Instrument Developer should consider using components that can survive at -55° C to minimize the survival power demands on the Spacecraft.

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Rationale: -55° C is a common temperature to which space components are certified. Using components certified to this temperature decreases the survival heater power demands placed upon the Host Spacecraft.

9.4.1.3 *Implementation of Cooling Function*

The Instrument Developer should consider implementing thermoelectric coolers or mechanical coolers if cryogenic temperatures are required for the instrument to ease the restrictions on Instrument radiator orientations.

Rationale: Thermoelectric or mechanical coolers provide an alternative technique to achieve very low temperatures that do not impose severe constraints on the placement of the radiator.

9.4.1.4 *Implementation of High Thermal Stability*

The Instrument Developer should consider implementing high thermal capacity hardware, such as phase change material, in order to increase the Instrument's thermal stability.

Rationale: Some optical instruments require very high thermal stability and given the relatively low masses expected in CII Instruments, incorporating phase change material for thermal storage is a useful technique.

9.4.2 <u>Survival Heaters</u>

The use of survival heaters is a technique to autonomously apply heat to an Instrument in the event that the thermal subsystem does not perform nominally, either due to insufficient power from the Host Spacecraft or an inflight anomaly.

9.4.2.1 *Survival Heater Responsibility*

The Instrument Provider should provide and install all Instrument survival heaters.

Rationale: Survival heaters are a component of the Instrument.

9.4.2.2 *Mechanical Thermostats*

The Instrument should control Instrument survival heaters via mechanical thermostats.

Rationale: Mechanical thermostat allows control of the survival heaters while the instrument avionics are not operating.

9.4.2.3 *Survival Heater Documentation*

The TICD will document survival heater characteristics and mounting details.

Rationale: This will capture the agreements negotiated by the Spacecraft Manufacturer and Instrument Developer.

9.4.2.4 *Minimum Turn-On Temperatures*

The Instrument should maintain the temperature of its components at a temperature no lower than that required to safely energize and operate the components.

Rationale: Some electronics require a minimum temperature in order to safely operate.

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9.4.3 Thermal Performance and Monitoring

9.4.3.1 *Surviving Arbitrary Pointing Orientations*

The Instrument should be capable of surviving arbitrary pointing orientations without permanent degradation of performance for a minimum of four (4) orbits with survival power only.

Rationale: This is a typical NASA earth orbiting science instrument survival requirement.

9.4.3.2 *Documentation of Temperature Limits*

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

Rationale: This will provide values for the Integration and Test technicians to monitor and manage.

9.4.3.3 *Documentation of Monitoring Location*

The TICD will document the location of all Instrument temperature sensors.

Rationale: This is the standard means to documents the agreement between the spacecraft and instrument

9.4.3.4 *Temperature Monitoring During OFF Mode*

The Instrument Designer should assume that the Host Spacecraft will monitor only one temperature on the spacecraft side of the payload interface when the payload is off. During extreme cases such as host anomalies, however, even this temperature might not be available.

Rationale: This limits the demands that the Instrument may place on the Host Spacecraft.

9.4.3.5 *Thermal Control Hardware Documentation*

The TICD will document Instrument thermal control hardware.

Rationale: This is the standard means to documents the agreement between the spacecraft and instrument

9.4.3.6 *Thermal Performance Verification*

The Instrument Developer should verify the Instrument thermal control system's ability to maintain hardware within allowable temperature limits either empirically by thermal balance testing or by analysis for conditions that cannot be ground tested.

Rationale: These verification methods ensure that the Instrument's thermal performance meets the guidelines and agreements documented in the TICD.

9.5 Environmental Reference Material / Best Practices

9.5.1 Introduction

The following environmental best practices provide potential Earth Science instrument and HPO spacecraft manufacturers a common design and analysis basis in order to progress the design of instrument prior to selection of a Host Spacecraft and launch vehicle.

9.5.2 Radiation-Induced SEE

The following best practices describe how the Instrument should behave in the event that a radiation-induced SEE does occur.

9.5.2.1 *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.

Rationale: Identifies that a temporary loss of function and/or data is permissible in support of correcting anomalous operations. This includes autonomous detection and correction of anomalous operations as well as power cycling.

9.5.2.2 *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.

Rationale: Identifies that autonomous fault detection and correction should be implemented.

9.5.2.3 *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 10^7 $ions/cm^2$.

Rationale: Identifies limitations for radiation induced latch-up and prescribes both a LET and an ion fluence immunity level

9.6 Software Engineering Reference Material / Best Practices

The Instrument System's software should comply with Class C software development requirements and guidelines, in accordance with NPR 7150.2A

Rationale: NPR 7150.2A Appendix E assigns Class C to "flight or ground software that is necessary for the science return from a single (non-primary) instrument." NASA Class C software is any flight or ground software that contributes to mission objectives, but whose correct functioning is not essential to the accomplishment of primary mission objectives. In this context, primary mission objectives are exclusively those of the Host Spacecraft.

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9.7 Contamination Reference Material / Best Practices

9.7.1 Assumptions

1) During the matching process, The Host Spacecraft Owner/Integrator and the Instrument Developer will negotiate detailed parameters regarding contamination control. The Contamination Interface Control Document (CICD) will record those parameters and decisions.
2) The Instrument Developer will ensure that any GSE accompanying the Instrument is cleanroom compatible in accordance with the CICD.
3) The Instrument Developer will ensure that any GSE accompanying the Instrument into a vacuum chamber during Spacecraft thermal-vacuum testing is vacuum compatible in accordance with the CICD.
4) The Host Spacecraft Manufacturer/Systems Integrator will attach the Instrument to the Host Spacecraft such that the contamination products from the vents of the Instrument do not directly impinge on the contamination-sensitive surfaces nor directly enter the aperture of another component of the Spacecraft system.
5) The Host Spacecraft Manufacturer/Systems Integrator will install protective measures as provided by the Instrument Provider to protect sensitive Instrument surfaces while in the Shipment, Integration and Test, and Launch environments.
6) The Launch Vehicle Provider will define the upper limit for the induced contamination environment. This is typically defined as the total amount of molecular and particulate contamination deposited on exposed spacecraft surfaces from the start of payload fairing encapsulation until the upper stage separation and contamination collision avoidance maneuver (CCAM).

9.7.2 Instrument Generated Contamination

9.7.2.1 *Verification of Cleanliness*

The Instrument Developer should verify by test the cleanliness of the instrument exterior surfaces documented in the CICD, prior to delivery to the Spacecraft Manufacturer/Systems Integrator.

Rationale: The Instrument must meet surface cleanliness requirements that are consistent with the cleanliness requirements as specified for the Host Spacecraft by the Spacecraft Manufacturer. A record of the cleanliness verification should be provided to the Host Spacecraft Manufacturer prior to Instrument integration with the Host Spacecraft.

9.7.2.2 *Instrument Sources of Contamination*

The CICD will document all sources of contamination that can be emitted from the Instrument.

Rationale: This determines the compatibility of the Instrument with the Host Spacecraft and mitigate the risk of Instrument-to-Host-Spacecraft cross contamination.

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9.7.2.3 *Instrument Venting Documentation*

The CICD will document the number, location, size, vent path, and operation time of all Instrument vents.

Rationale: Mitigation of Instrument-to-Host-Spacecraft cross contamination (See 9.7.2.2)

9.7.2.4 *Flux of outgassing products*

The CICD will document the flux ($g/cm^2/s$) of outgassing products issuing from the primary Instrument vent(s).

Rationale: Mitigation of Instrument-to-Host-Spacecraft cross contamination (See 9.7.2.2)

9.7.2.5 *Sealed Hardware*

The Instrument should prevent the escape of actuating materials from Electro-explosive devices (EEDs), hot-wax switches, and other similar devices.

Rationale: Mitigation of Instrument-to-Host-Spacecraft cross contamination (See 9.7.2.2)

9.7.2.6 *Nonmetallic Materials Selection*

The Instrument design should incorporate only those non-metallic materials that meet the nominal criteria for thermal-vacuum stability: Total Mass Loss (TML) ≤ 1.0 %, Collected Volatile Condensable Material (CVCM) ≤ 0.1 %, per ASTM E595 test method.

Rationale: Host Spacecraft Manufacturers generally require that all nonmetallic materials conform to the nominal criteria for thermal-vacuum stability. A publicly accessible database of materials tested per ASTM E595 is available at: www.outgassing.nasa.gov Note: Some Host Spacecraft Manufacturers may require lower than the nominal levels of TML and CVMC.

9.7.2.7 *Wiring and MLI Cleanliness Guidelines*

The CCID will document thermal vacuum bakeout requirements for Instrument wiring harnesses and MIL.

Rationale: Thermal vacuum conditioning of materials and components may be necessary to meet Spacecraft contamination requirements.

9.7.2.8 *Particulate Debris Generation*

The Instrument design should avoid the use of materials that are prone to produce particulate debris.

Rationale: Host Spacecraft Manufacturers generally prohibit materials that are prone to produce particulate debris, either from incidental contact or though friction or wear during operation. Therefore, such materials, either in the construction of the payload or ground support equipment, should be avoided. Where no suitable alternative material is available, an agreement with the Host Spacecraft will be necessary and a plan to mitigate the risk posed by the particulate matter implemented.

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9.7.2.9 *Spacecraft Integration Environments*

The Instrument should be compatible with processing in environments ranging from IEST-STD-1246 ISO-6 to ISO-8.

Rationale: Spacecraft integration facilities may vary in cleanliness and environmental control capabilities depending on the Spacecraft Manufacturer and integration/test venue. Instruments and associated ground support equipment should be compatible with protocols contamination control of ISO-6 cleanroom environments. Instruments should be compatible with operations in up to ISO-8 environments, employing localized controls such as bags, covers, and purges to preserve cleanliness; such controls must be integrated into the spacecraft integrations process.

9.7.3 <u>Accommodation of Externally Generated Contamination</u>

9.7.3.1 *Protective Covers: Responsibility*

The Instrument Developer should provide protective covers for any contamination-sensitive components of the Instrument.

Rationale: Preservation of Instrument cleanliness during Spacecraft I&T.

9.7.3.2 *Protective Covers: Documentation*

The CICD will document the requirements and procedures for the use of protective covers (such as bags, draping materials, or hardcovers).

Rationale: Preservation of Instrument cleanliness during Spacecraft I&T.

9.7.3.3 *Instrument Cleanliness Requirements*

The CICD will document the cleanliness goals for all contamination-sensitive instrument surfaces that will be exposed while in the Integration and Test Environment.

Rationale: Enables the Spacecraft Manufacturer and Instrument Provider to negotiate appropriate and reasonable instrument accommodations or determine the degree of deviation from the defined goals.

9.7.4 <u>Instrument Purge Requirements</u>

The CICD will 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.

Rationale: The Spacecraft Manufacturer generally will provide access to a gas supply of the desired type, purity, and flow rate. The Instrument provider is responsible to provide the necessary purge interface ground support equipment (See 9.7.4.1).

9.7.4.1 *Instrument Purge Ground Support Equipment (GSE)*

The Instrument Provider should provide purge ground support equipment (GSE) incorporating all necessary filtration, gas conditioning, and pressure regulation capabilities.

Rationale: The Instrument provider is responsible for control of the gas input to the instrument during Spacecraft Integration & Test. This purge GSE is the interface between the Instrument and the gas supply provided by the Spacecraft Manufacturer.

9.7.4.2 *Spacecraft to Instrument Purge Interface*

The MICD will document any required mechanical interface of the Instrument purge between the Instrument and Spacecraft.

Rationale: If the Instrument Purge requires a mechanical interface with the Spacecraft, that interface shall be documented in the MICD. The Spacecraft Manufacturer will negotiate with the Launch Vehicle Provider any resultant required purge interface between the Spacecraft and Launch Vehicle.

9.7.4.3 *Instrument Inspection and Cleaning During I&T: Responsibility*

The Instrument Provider should be responsible for cleaning the Instrument while in the Integration and Test Environment.

Rationale: The Instrument Provider is responsible for completing any required inspections during I&T. The Instrument Provider may, upon mutual agreement, designate a member of the Spacecraft I&T team to perform inspections and cleaning.

9.7.4.4 *Instrument Inspection and Cleaning During I&T: Documentation*

The CICD will document any required inspection or cleaning of the Instrument while in the Integration and Test Environment.

Rationale: Instrument inspections and cleaning consume schedule resources and must be conducted in coordination with other Spacecraft I&T activities.

9.7.4.5 *Spacecraft Contractor Supplied Analysis Inputs*

The CICD will document the expected Spacecraft-induced contamination environment.

Rationale: Mitigate the risk of Instrument-Spacecraft cross contamination. The Spacecraft Contractor may perform analyses or make estimates of the expected spacecraft-induced contamination environment, which will be documented in CICD. The results of such assessments may include a quantitative estimate of the deposition of plume constituents to Instrument surfaces and be used to determine the allowable level of contamination emitted from the Instrument.

9.7.4.6 *Launch Vehicle Contractor Supplied Analysis Inputs*

The CICD will document the Launch Vehicle-induced contamination environment

Rationale: Most Launch Vehicle Providers are able to provide nominal information regarding the upper bound of molecular and particulate contamination imparted to the Spacecraft Payload surfaces; frequently such information is found in published User Guides for specific Launch Vehicles. Spacecraft and Instrument Providers should use this information in developing

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mitigations against the risk of contamination during integrated operations with the Launch Vehicle.

9.8 Model Guidelines and Submittal Details

9.8.1 Finite Element Model Submittal

The Instrument Developer should supply the Spacecraft Manufacturer with a Finite Element Model in accordance with the GSFC GIRD.

Rationale: The GIRD defines a NASA Goddard-approved interface between the Earth Observing System Common Spacecraft and Instruments, including requirements for finite element models. As of the publication of this guideline document, Gird Rev B is current, and the Finite Element Model information is in Section 11.1.

9.8.2 Thermal Math Model

The Instrument Developer should supply the Spacecraft Manufacturer with a reduced node geometric and thermal math model in compliance with the following sections.

Rationale: The requirements and details for the Thermal Model submittal listed in this section are based on commonly used NASA documents such as GSFC GIRD and JPL spacecraft instrument interface requirement documents.

9.8.2.1 *Model Format*

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

9.8.2.2 *Units of Measure*

Model units should be SI.

9.8.2.3 *Radiating Surface Element Limit*

Radiating surface elements should be limited to less than 200.

9.8.2.4 *Thermal Node Limit*

Thermal nodes should be limited to less than 500.

9.8.2.5 *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.

9.8.2.6 *Steady-State and Transient Analysis*

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

9.8.2.7 *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:

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9.8.2.1.7 Node(s) Location

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

9.8.2.2.7 Electrical Heat Dissipation

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

9.8.2.3.7 Active Thermal Control

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

9.8.2.4.7 Boundary Notes

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

9.8.2.5.7 Environmental Heating

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

9.8.2.6.7 User Generated Logic

A description of any user generated software logic

9.8.3 Thermal Analytical Models

The Instrument Provider should furnish the Spacecraft Manufacturer 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.

9.8.4 Mechanical CAD Model

9.8.4.1 *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.

Rationale: The Spacecraft Manufacturer may need Mechanical CAD models for hosted payload assessment studies.

9.8.5 Mass Model

9.8.5.1 *Instrument Mass Model*

The Instrument Provider should provide all physical mass models required for spacecraft mechanical testing.

Rationale: The Spacecraft Manufacturer may fly the mass model in lieu of the Instrument in the event that Instrument delivery is delayed.

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1929 # Appendix A Acronyms

AI&T	Assembly, Integration, and Test
AP	Average Power
ASD	Acceleration Spectral Density
AWG	American Wire Gauge
CCSDS	Consultative Committee for Space Data Systems
CE	Conducted Emissions
CICD	Contamination ICD
CII	Common Instrument Interface
COTS	Commercial Off The Shelf
CS	Conducted Susceptibility
CVCM	Collected Volatile Condensable Material
DICD	Data ICD
EED	Electro-explosive Device
EICD	Electrical Power ICD
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOS	Earth Observing System
EPS	Electrical Power System
ESA	European Space Agency
EVI	Earth Venture Instrument
FDIR	Fault Detection, Isolation, and Recovery
FOV	Field of View
GCR	Galactic Cosmic Ray
GEO	Geostationary Earth Orbit
GEVS	General Environmental Verification Standard
GIRD	General Interface Requirements Document
GOES	Geostationary Operational Environmental Satellites
GSE	Ground Support Equipment
GSFC	Goddard Spaceflight Center
GTO	Geostationary Transfer Orbit
HPO	Hosted Payload Opportunity
HPOC	Hosted Payload Operations Center
HSOC	Host Spacecraft Operations Center
I&T	Integration and Test
IAC	Interface Alignment Cube
ICD	Interface Control Document
KDP	Key Decision Point
LEO	Low Earth Orbit
LET	Linear Energy Transfer
LVDS	Low Voltage Differential Signaling
MAC	Mass Acceleration Curve

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MICD	Mechanical ICD
MLI	Multi-layer Insulation
NDE	Non-Destructive Evaluation
NICM	NASA Instrument Cost Model
NPR	NASA Procedural Requirement
NTIA	National Telecommunications and Information Administration
OAP	Orbital Average Power
PI	Principal Investigator
PPL	Preferred Parts List
RDM	Radiation Design Margin
RE	Radiated Emissions
RFI	Request for Information
RS	Radiated Susceptibility
RSDO	Rapid Spacecraft Development Office
SEE	Single Event Effect
SI	Système Internationale
SMC	US Air Force Space and Missile Systems Center
SMC/XRFH	SMC Hosted Payload Office
SPS	Spectrum Planning Subcommittee
SRS	Shock Response Spectrum
TEMP	Test and Evaluation Master Plan
TICD	Thermal ICD
TID	Total Ionizing Dose
TML	Total mass Loss
VDC	Volts Direct Current

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Appendix C Units of Measure and Metric Prefixes

Table C-1: Units of Measure

Abbreviation	Unit
A	ampere
Arcsec	arc-Second
B	bel
bps	bits per second
eV	electron-volt
F	farad
g	gram
Hz	hertz
J	joule
m	meter
N	newton
Pa	pascal
Rad [Si]	radiation absorbed dose ≡ 0.01 J/(kg of Silicon)
s	second
T	tesla
Torr	torr
V	volt
Ω	ohm

Table C-2: 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})

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Appendix D CII Hosted Payload Concept of Operations

D.1 INTRODUCTION

This CII Hosted Payloads Concept of Operations (CONOPS) provides a prospective Instrument Developer with technical recommendations to help them design an Instrument that may be flown as a hosted payload either in LEO or GEO. This document describes the systems, operational concepts, and teams required to develop, implement, and conduct a hosted payload mission. More specifically, this CONOPS document primarily supports stakeholders involved in NASA Science Mission Directorate (SMD) Earth Science Division's investigations. What follows is a CONOPS applicable to those ESD payloads to be hosted as a secondary payload, including those developed under the EVI solicitation.

D.1.1 **Goals and Objectives**

The CONOPS documents the functionality of a hosted payload mission and defines system segments, associated functions, and operational descriptions. The CONOPS represents the operational approaches used to develop mission requirements and provides the operational framework for execution of the major components of a hosted payload mission.

The CONOPS is not a requirements document, but rather, it provides a functional view of a hosted payload mission based upon high-level project guidance. All functions, scenarios, figures, timelines, and flow charts are conceptual only.

D.1.2 **Document Scope**

The purpose of this CONOPS)document is to give an overview of LEO and GEO satellites operations, with an emphasis on how such operations will impact hosted payloads.

This CONOPS is not a requirements document and will not describe the Instrument Concept of Operation in detail or what is required of the Instrument to operate while hosted on LEO/GEO satellites.

D.2 COMMON INSTRUMENT INTERFACE PHILOSOPHY

This CONOPS supports the "Do No Harm" concept as described in section 2.2.1.

D.3 LEO/GEO SATELLITE CONCEPT OF OPERATIONS SUMMARY

This section is intended to be a summary of the Concept of Operations for both Low Earth Orbit Satellites [LEO] and Commercial Geostationary Communications Satellites [GEO], to give the Instrument provider an idea of what to expect when interfaced to the Host Spacecraft.

D.3.1 **General Information**

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[LEO] Nominal Orbit: The Host Spacecraft will operate in a Low Earth Orit with an altitude between 350 and 2000 kilometers with eccentricity less than 1 and inclination between zero and 180°, inclusive (see section 2.1.3). LEO orbital periods are approximately 90 minutes.

[LEO] The frequencies used for communicating with LEO spacecraft vary, but S-Band (2–4 GHz) with data rates up to 2 Mbps are typical. Since communication with ground stations requires line-of-site, command uplink and data downlink are only possible periodically and vary considerably depending on the total number of prime and backup stations and their locations on Earth. Communication pass durations are between 10–15 minutes for a minimum site angle of 10°.

[GEO] Nominal Orbit: The Host Spacecraft will operate in a Geostationary Earth Orbit with an altitude of approximately 35786 kilometers and eccentricity and inclination of approximately zero (see section 2.1.2.) GEO satellites remain in the same fixed location over the ground location for the life of the mission. Station keeping is required on a regular basis to maintain that fixed position. Current commercial communication satellite locations are as shown in Figure D-1. If full continental United States coverage is desired, a location of around 95^0W - 100^0W may be desired as shown in Figure D-2.

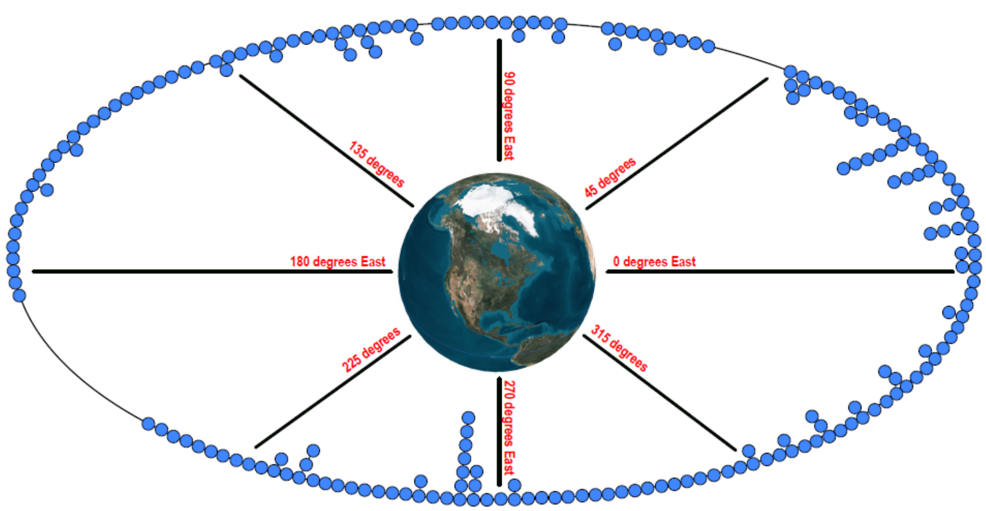

Figure D-1: Geostationary Locations

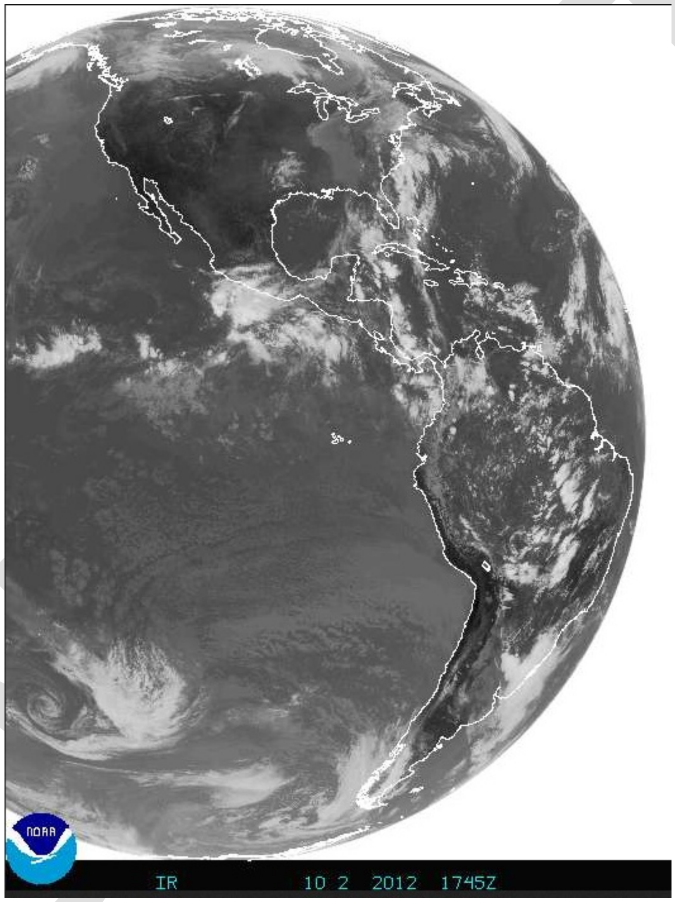

Figure D-2: GOES-14 Image at 100^0W

[GEO] The Instrument approach provides the advantage of utilizing the Commercial Satellite's location, features, and services. Due to the location, the Instrument will have minimal data latency due to continuous real-time bi-directional communications links. As older commercial communications satellites are going out of service, newer more sophisticated satellites are replacing them.

[GEO] The Instrument will have the option to either purchase command and/or telemetry and/or data services from the Host Spacecraft or provide their own. The Instrument will have continuous direct data transfer to/from the Host Spacecraft during normal operations. The Instrument will have continuous direct data broadcast with the ground via the Host Spacecraft ground system during normal operations, as shown in Figure D-3.

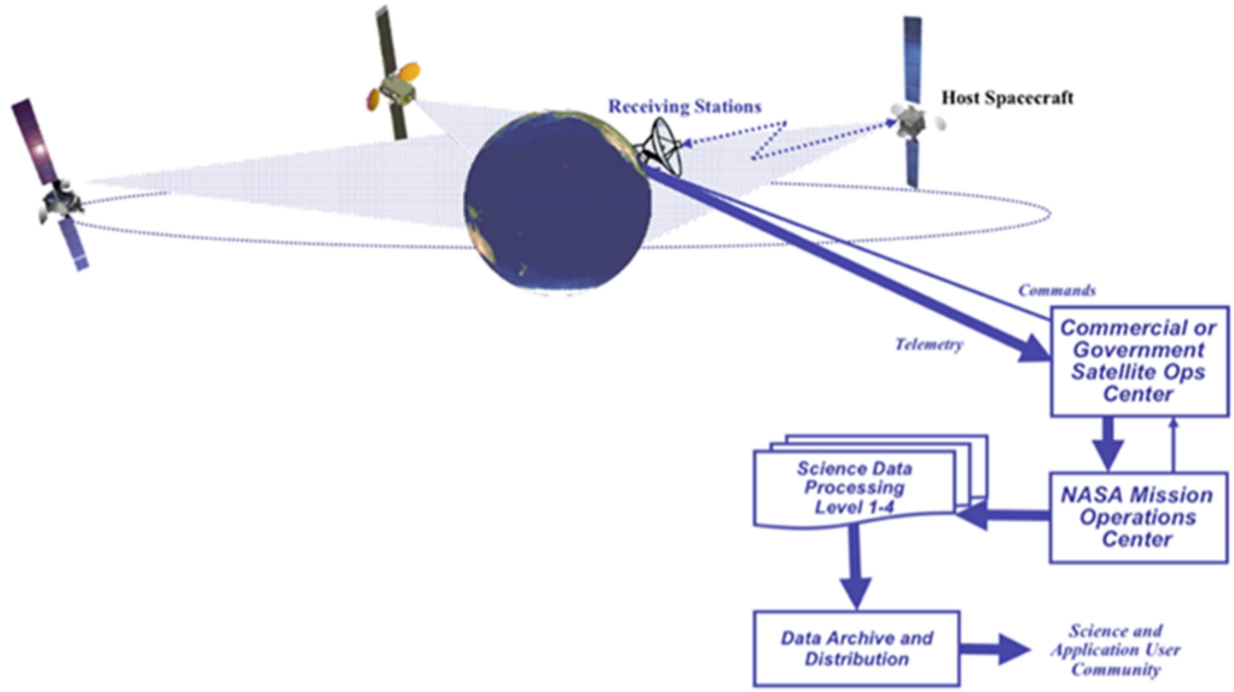

Figure D-3: Ground System Interfaces

D.3.2 Phases of Operation

The Host Spacecraft will have numerous phases of operation, which can be described as launch & ascent, [GEO] Geostationary Transfer Orbit (GTO), checkout, normal operations, and safehold. The Instrument will have similar phases that occur in parallel with the Host Spacecraft. A summary of the transition from launch to normal operations is as shown in Figure D-4.

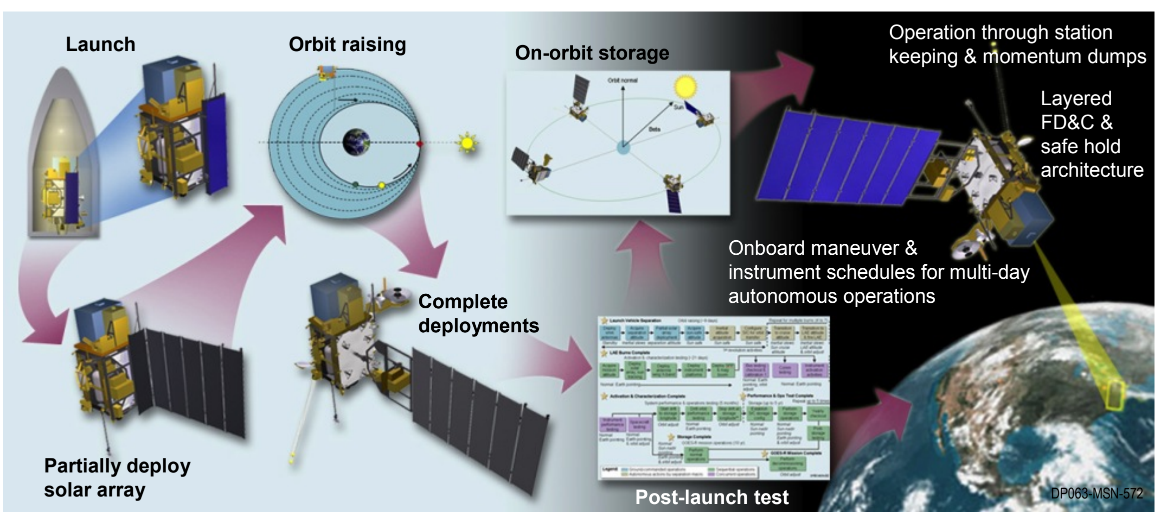

Figure D-4: Summary of Transition to Normal Operations

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Launch and Ascent
During this phase, the satellite is operating on battery power and is in a Standby power mode, minimal hardware is powered on, *e.g.*, computer, heaters, RF receivers, etc.

Heaters, the RF receiver and the satellite computer will be powered on collecting limited health and status telemetry and when the payload fairing is deployed, the RF transmitter may automatically be powered on to transmit health and status telemetry of the satellite, this is vendor specific.

Instrument Launch and Ascent
The Instrument will be powered off, unless it is operating on its own battery power and the Host Spacecraft has agreed to allow it to be powered. No commutations between the Instrument and the Host Spacecraft or the ground (in the event the Instrument has a dedicated RF transponder) will take place. The Host Spacecraft may provide survival heater power to the Instrument during this phase, as negotiated with the Host Spacecraft.

Orbit Transfer ([GEO] GTO)
[GEO] During this phase, the satellite is in transition to its orbital location and will take several days, depending on the method of transfer and the propulsion. Conventional propulsion systems can take up to 10 days, while electric propulsion systems can take up to 6 months. Typically, prior to the first burn, the solar array is partial deployed to allow more satellite hardware to be powered, provide power to the electric propulsion system if used and charge the batteries, as shown in Figure D-4.

[LEO] The satellite will be injected directly into its orbit location as part of the launch and ascent phase.

Instrument Orbit Transfer
The Instrument will be powered off and no commutations between the Instrument and the Host Spacecraft or the ground (in the event the Instrument has a dedicated RF transponder) will take place, unless negotiated otherwise with the Host Spacecraft due to the science data to be collected. If the Instrument is powered off, the Host Spacecraft will provide survival heater power, as negotiated.

If the Instrument is powered on during this phase, the Host Spacecraft will provide primary power as negotiated.

On-Orbit Storage
[GEO] The satellite may inject into a storage location to either perform its checkout or if the operational satellite hasn't been decommissioned. The checkout for the Host Spacecraft as well as the instrument will be performed at this location. At the appointed time, the satellite will perform a series of maneuvers to re-locate to the operational location.

[LEO] An on-orbit storage location may be used if the Host Spacecraft is part of a constellation and the current operational spacecraft hasn't been decommissioned. The satellite may inject into this location to perform its checkout, as will the instrument. Upon completion of the checkout or if the operational satellite has been decommissioned, the Host Spacecraft will perform a series of maneuvers to re-locate into its location within the constellation.

Checkout

After orbit transfer and the final burn is completed and the orbital location has been successfully achieved, full solar array deployment will take place and the satellite check process will begin. Each subsystem will be fully powered and checked out in a systematic manor. Once the satellite is successfully checked-out and operational, the communication payload checkout begins, also in a systematic manor. When both the satellite and communications payload are successfully checked-out, the owner/operator will transition to normal operations.

Normal Operations

The satellite is in this phase as long as all hardware and functions are operating normally and will remain in this phase for the majority of its life.

Once the transition to normal operations is achieved, only then is the hosted payload powered on and the checkout process begun.

Instrument Checkout

After the Host Spacecraft has achieved normal operations, the Instrument will be allowed to power on and begin its checkout process. Calibration of the Instrument would be during this phase as well. Any special maneuvering required of the Host Spacecraft will be negotiated.

Instrument Normal Operations

The Instrument will remain in this phase as long as all hardware and functions are operating normally and will remain in this mode for the majority of its life.

Safehold

While not technically an operational phase, this mode is achieved when some sort of failure of the Host Spacecraft has occurred. This mode can be achieved either autonomously or manually. During this mode, all non-essential subsystems are powered off, the communications payload maybe powered off, depending on the autonomous trigger points programmed in the flight software, the hosted payload will be powered off, and the satellite will be maneuvered into a power-positive position. When the Host Spacecraft enters Safehold the Instrument may be commanded into Safehold, but will most likely be powered-off.

After the failure has been understood and it is safe to do so, the owner/operator Mission Operations Center will transition the Host Spacecraft back to normal operations. After normal operations have been achieved, the hosted payload will be powered back on.

Instrument Safehold

The Instrument will go into this mode one of two ways, either due to a Host Spacecraft failure or an Instrument failure. In the event the Host Spacecraft experiences a failure, the Host Spacecraft will immediately cut the power to the Instrument and the Instrument will have to be able to successfully recover from this event.

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In the event the Instrument experiences a failure of some sort, it will have to autonomously move into this mode without manual intervention. The Instrument Mission Operations Center will manually perform the trouble shooting required and manually transition the Instrument back to normal operations.

Instrument Safehold Recovery
If Host Spacecraft operations require the Instrument to be powered off with no notice, the Instrument must autonomously recover in a safe state once power has been restored. Once health and status telemetry collection and transmission via the Host Spacecraft has been restored, the Instrument operations center may begin processing data.

Host Spacecraft Normal Operations After Instrument End of Life
Commercial spacecraft are designed to have operational lifetimes of typically less than 10 years in LEO, while GEO lifetimes of 15 years or more are common. Instrument lifetimes are prescribed by their mission classification (Class C, no more than 2 years). The Instrument lifetime may be extended due to nominal performance and extended missions may be negotiated (Phase E). Since the Host Spacecraft may outlive the Instrument, especially commercial GEO satellites, the Instrument must be capable of safely decommissioning itself.

During the end of life phase, the Instrument will be completely unpowered, unless survival heaters are required to assure host satellite safety, and, essentially, inert. This may involve the locking of moving parts and the discharge of any energy or consumables in the payload. This process will be carried out such that it will not perturb the Host Spacecraft in any way. Upon completion, the Host Spacecraft will consider the Instrument as a simple mass model that does not affect operations.

De-commissioning
At the end of the Host's mission life, it will perform a series of decommissioning maneuvers to de-orbit to clear the geostationary location. The Instrument will have been configured into the lowest possible potential energy state and then powered down at the end of its mission. The host maneuvers may span several days to relocate where it will be powered down and its mission life will end.

D.4 HOSTED PAYLOAD OPERATIONS

The Host Spacecraft will have a primary mission different than that of the Instrument. The Instrument's most important directive is to not interfere or cause damage to the Host Spacecraft or any of its payloads, and to sacrifice its own safety for that of the Host Spacecraft.

The Host Spacecraft has priority over the Instrument. Special or anomalous situations may require temporary suspension of Instrument operations. Instrument concerns are always secondary to the health and safety of the Host Spacecraft and the objectives of primary payloads. Suspension of Instrument operations may include explicitly commanding the Instrument to Safe mode or powering it off. The Host Spacecraft operator may or may not inform the Instrument operators prior to suspension of operations.

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D.4.1 Instrument Modes of Operation

Table D-1 shows the command and control responsibilities of the commercial Host Spacecraft Operations Center (HSOC) and Hosted Payload Operations Center (HPOC) for hosted payload missions. Hosted payload power control will be performed by HSOC commands to the commercial satellite with hosted payload commanding performed by the HPOC after power is enabled. Operation of the hosted payload will be performed by the HPOC. In case of any space segment anomalies, the HSOC and HPOC will take corrective actions with agreed upon procedures and real-time coordination by the respective control teams.

Table D-1:GEO/LEO Instrument Operating Modes Based Upon Mission Phase

Instrument Mission Phase	Launch	Orbit Transfer	On Orbit Storage	Checkout	Nominal Operations	Anomalous Operations	End of Life
Survival Power	OFF/ON	ON	ON	ON	ON	ON	ON/OFF
Instrument Power	OFF	OFF	OFF	OFF/ON	ON	ON	ON/OFF
Mode	OFF/ SURVIVAL	OFF/ SURVIVAL	OFF/ SURVIVAL	INITIALIZE/ OPERATION/ SAFE	OPERATION	SAFE	SAFE/ OFF/ SURVIVAL
Command Source	NA	NA	NA	HPOC	HPOC	HPOC	HPOC/ NA

Note: Host Spacecraft controls Instrument power.

The following are a set of short descriptions of each of the basic modes of operation. A more detailed set of guidance regarding these basic modes and transitions may be found in Appendix G.

<u>OFF/SURVIVAL Mode</u>

In the OFF/SURVIVAL Mode, the Instrument is always unpowered and the instrument survival heaters are in one of two power application states. In the survival heater OFF state of the OFF/SURVIVAL mode, the survival heaters are unpowered. In the survival heater ON state of the OFF/SURVIVAL Mode, the survival heaters are powered. The spacecraft should verify that the power to the survival heaters is enabled after the command to enter the survival heater ON state of the OFF/SURVIVAL mode has been actuated. Nominal transitions into the OFF/SURVIVAL mode are either from the INITIALIZATION mode, the SAFE mode or the OPERATION mode with the preferred path being a transition from the SAFE mode. The only transition possible out of the OFF/SURVIVAL mode is into the INITIALIZATION mode.

It is important to note that the Instrument should be capable of withstanding a near instantaneous transition into the OFF/SURVIVAL mode at any time and from any of the other three Instrument modes. Such a transition may be required by the Spacecraft host and would result from the sudden removal of operational power. This could occur without advance warning or notification and with no ability for the Instrument to go through an orderly shutdown sequence. This sudden removal of instrument power could also be coupled with the near instantaneous activation of the survival heater power circuit(s).

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INITIALIZATION Mode

When first powered-on, the Instrument transitions from the OFF/SURVIVAL mode to the INITIALIZATION mode and conducts all the internal operations that are necessary in order to transition to the OPERATION mode or to the SAFE mode. These include, but are not limited to, activation of command receipt and telemetry transmission capabilities, initiation of health and status telemetry transmissions and conducting instrument component warm-up/cool-down to nominal operational temperatures. The only transition possible into the INITIALIZATION mode is from the OFF/SURVIVAL mode. Nominal transitions out of the INITIALIZATION mode are into the OFF/SURVIVAL mode, the SAFE mode or the OPERATION mode.

OPERATION Mode

The Instrument should have a single OPERATION mode during which all nominal Instrument operations occur. It is in this mode that science observations are made and associated data are collected and stored for transmission at the appropriate time in the spacecraft operational timeline. Within the OPERATION mode, sub-modes may be defined that are specific to the particular operations of the Instrument (*e.g.* STANDBY, DIAGNOSTIC, MEASUREMENT, etc.). When the Instrument is in the OPERATION mode, it should be capable of providing all health and status and science data originating within the instrument for storage or to the Spacecraft for transmission to the ground operations team. Nominal transitions into the OPERATION mode are either from the INITIALIZATION mode or the SAFE mode. Nominal transitions out of the OPERATION mode are into either the OFF/SURVIVAL mode or the SAFE mode.

SAFE Mode

The Instrument SAFE mode is a combined Instrument hardware and software configuration that is intended to protect the Instrument from possible internal or external harm while using a minimum amount of Spacecraft resources (*e.g.* power). When the instrument is commanded into SAFE mode, it should notify the Spacecraft after the transition into this mode has been completed. Once the Instrument is in SAFE mode, the data collected and transmitted to the HPOC should be limited to health and status information only. Nominal transitions into the SAFE mode are either from the INITIALIZATION mode or the OPERATION mode. Nominal transitions out of the SAFE mode are into either the OFF/SURVIVAL mode or the OPERATION mode.

D.4.2 **Instrument Interfaces**

The instrument should refer to the referenced Guidelines document for all Instrument/Host Spacecraft interfaces.

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Appendix E Supporting Analysis for LEO 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 Table E-1. 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 E-1: 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 E-1.

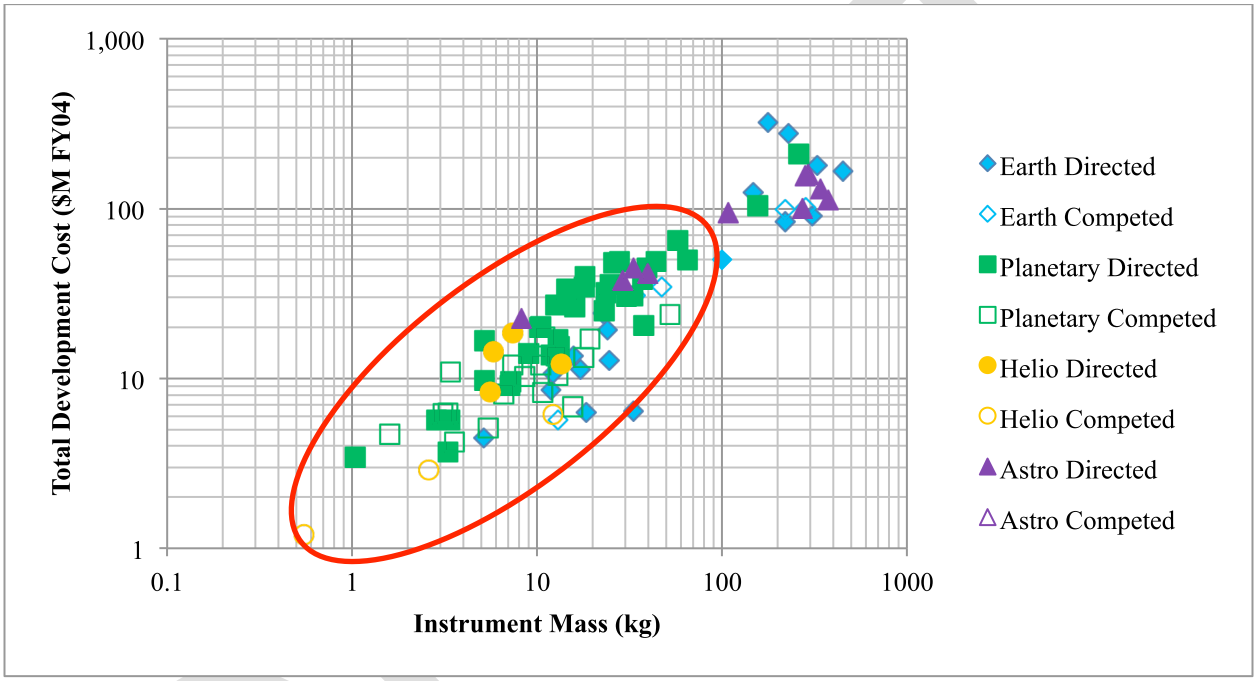

Figure E-1: Instrument Mass vs. Development Cost

In further examination of the data, specifically the Earth Science instruments that are outside the ellipse in Figure E-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 E-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.

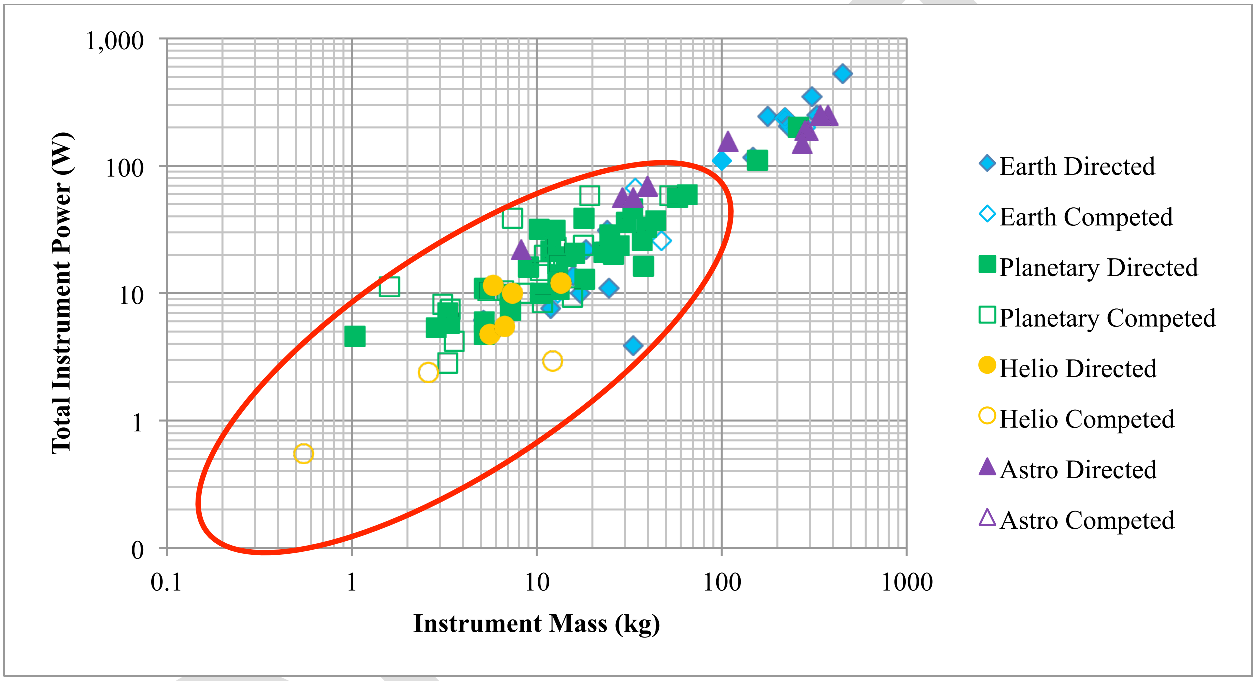

Figure E-2: Power as a Function of Mass

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 E-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.

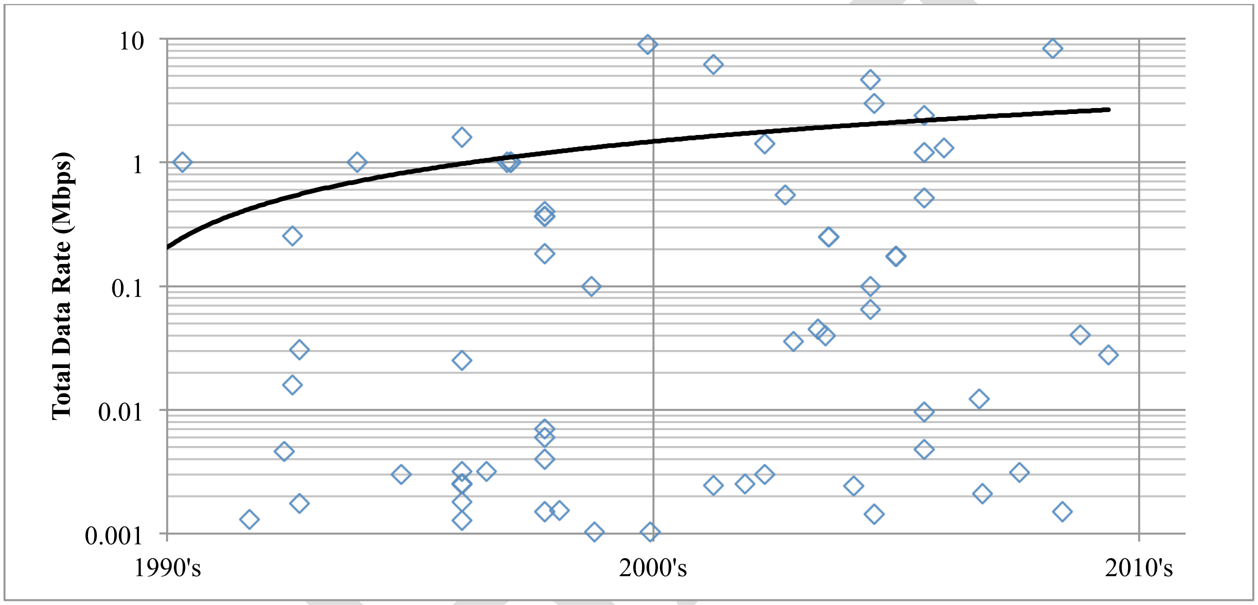

Figure E-3: Trend of Mean Instrument Data Rates

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.

Appendix F Supporting Analysis for GEO Guidelines

On 29 March 2012, NASA Langley Research Center released a *Request for Information (RFI) for Geostationary Earth Orbit (GEO) Hosted Payload Opportunities (HPO) and Accommodations*, upon whose responses the CII Team primarily established our GEO guidelines.

NASA Langley Research Center is hereby soliciting information about potential sources for Geostationary Earth Orbit (GEO) Hosted Payload Opportunities (HPO) and Accommodations.

Background

NASA's Earth Science Division (ESD) will be developing Earth Science Instruments, some of which may be suitable to fly as hosted payloads on HPO's. The development of the instruments as well as the HPO's will be conducted independently of each other with the goal of matching a specific instrument with a specific HPO by the instrument Preliminary Design Review (PDR) timeframe.

In an effort to facilitate matching instruments to HPO's, ESD initiated the Common Instrument Interface (CII) Project. The charter for the CII Project is to work with industry, academia, and other governmental agencies to develop a set of common instrument-to-spacecraft interfaces that could serve as guidelines for instrument developers. If used properly by instrument developers, these guidelines would help produce instruments that have a less complex interface and would improve the probability of matching a given instrument with a HPO or platform.

The CII Project has recently completed a draft set of Low Earth Orbit (LEO) guidelines and a draft HPO Database document. Additional information on the CII project may be found at this website: http://science.nasa.gov/about-us/smd-programs/earth-system-science-pathfinder/common-instrument-interface-workshop/. Current CII Guideline and HPO database documents may be found on the Earth Venture Instruments 1 Program Library: http://essp.larc.nasa.gov/EV-I/evi_programlibrary.html

Current Intention

The CII Project is now interested in identifying HPO's, and their associated accommodations, for future GEO missions in order to develop a draft set of GEO guidelines to complement our LEO guidelines, and to update the publically-available HPO database document. Additionally, the CII Project is investigating flying and operating a hosted payload on an upcoming GEO HPO as a pathfinder initiative (hereafter described as the "Initiative") to better understand the programmatic and

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technical challenges for a commercially-hosted NASA science payload. The CII Project would document lessons learned from conducting the GEO pathfinder initiative and feed them back into the GEO guidelines document to help developers intending to fly a payload on a future GEO HPO.

The purpose of this RFI is to:

1) Identify the GEO HPO's for the period from 2013-2023;
2) Obtain a description of available HPO payload accommodations on future GEO HPO's; and
3) Obtain information on all of the steps required to fly the "Initiative" as described later in this RFI.

The CII Project can accommodate responses containing properly-marked proprietary information. The CII Project will safeguard the proprietary information on hosted payload opportunities (Requested Information #1) and payload accommodations (Requested Information #2) within the Project organization. The CII Project intends to utilize the non-proprietary portions of Requested Information #1 to update the publicly available HPO database. The CII Project intends to use Requested Information #2 to bound/envelope the payload accommodation parameters that will inform the future GEO Guidelines Document. NASA may also use Requested Information #1 and #2 to assess the suitability of hosted payload-to-spacecraft matches associated with future NASA Earth Science missions.

The CII Project will use the requested information for the GEO pathfinder initiative (Requested Information #3 above) to assess the feasibility of such an Initiative, to provide an overview of the hosted payload process in the future GEO Guidelines Document, and to inform future Earth Science hosted payload planning and programming activities.

Requested Information

1. Please identify your organization's HPO's for the period of 2013-2023 with their associated mission parameters including but not limited to:

- Mission Name
- Launch Date
- Owner/Operator
- Primary Customer
- Spacecraft Bus Manufacturer
- Spacecraft Bus Model
- Launch Vehicle
- Orbital Longitude

If the data are not available beyond your current business cycle, please suggest a technique for the CII Project to obtain those data once they do become available.

2. Please describe what Payload Accommodation and Interface resources your HPO’s can provide to a prospective hosted payload without significant modifications to your nominal manufacturing, integration, test and launch processes. Please also describe the environment the prospective hosted payload might expect to encounter:

- Payload Accommodation Parameters and Interface
    - Maximum Payload Mass Available without System Redesign [kg]
    - Maximum Payload Orbital Average Power without System Redesign [W]
    - Maximum Payload Peak Power without System Redesign[W]
    - Main Bus Nominal Voltage [V]
    - Volume (l x w x h) [mm x mm x mm]
    - Sensor Mounting Location on Spacecraft (e.g. Nadir, Zenith, Ram, Wake, North, South, East, West, …)
    - Command and Control Interface (1553B, RS-422, SpaceWire, etc.) with average and peak data rates [kbps]
    - Payload-to-Transponder Interface (RS-422, SpaceWire, etc.) for Science Data Transmission with average and peak data rates [Mbps]
    - Host spacecraft constraints or preferences for digital formats most suitable for conversion to RF in system architecture
    - Payload command and control encryption requirements
    - Pointing Control [arcsec]
    - Pointing Knowledge [arcsec]
    - Pointing Stability [arcsec / sec]
    - Spacecraft absolute position accuracy, each axis [m]
    - Spacecraft absolute velocity accuracy, each axis [m/s]
    - Limitations with respect to payload-induced uncompensated torques [N x m] by frequency [Hz]
    - Limitations with respect to payload-induced uncompensated forces [N] by frequency [Hz]
    - Typical Integration and Test Facility Cleanliness [Cleanroom Class]
    - Thermal Rejection With Heat Pipes [W]
    - Thermal Rejection Without Heat Pipes [W]

- Payload Environment
    - Temperature Range

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   - Quasi-static loads
   - Minimum resonant frequency
   - Random vibration and acoustic loads
   - Shock environment
   - Disturbance torque
   - RF Field EMI/EMC/ESD
   - Molecular contamination as a function of mission elapsed time and hosted payload location

3. As a specific potential near term opportunity, please provide information on all of the programmatic and technical steps required to fly a GEO Pathfinder Initiative on your HPO's as described below.

<u>GEO Pathfinder Initiative Information</u>

The Initiative will also provide NASA with experience with the commercially-hosted payload process. The Initiative will also mitigate space environmental risks to future GEO missions by measuring vibration and contamination of an Instrument Suite hosted on a commercial GEO spacecraft. Both objectives will reduce risk on future commercially-hosted GEO Earth Science missions. See attached Figure 1 for an example of a notional Instrument Suite, which the CII Project will develop and provide, with the following characteristics:

- Mass: 50 kg
- Power: 125 W
- Volume: 1000 x 500 x 500 mm
- Data Rate: 60 Mbps
- Thermal Control: Electronics thermally isolated, with exterior boxes insulated with multi-layer insulation (MLI).
- The Instrument Suite is presumed to be mounted on the host spacecraft nadir deck.
- The Instrument Suite has a nominal operational lifetime of 3 years

Note: The Initiative is designed to exercise the GEO hosted payload process whose parameters are a subset and likely smaller than those of a typical future science flight mission.

<u>GEO Pathfinder Initiative Requested Information</u>

Please provide information related to the accommodation of the Instrument Suite by your mission:

- Date the contract needs to be signed relative to Launch Date

- Government-provided technical / programmatic deliverables required (e.g. mass and thermal models)
- Instrument Suite delivery date required relative to Launch Date
- Rough Order of Magnitude Price Estimate to fly and operate the Initiative. In addition to the Total price, please estimate the following components:
    - Integration, Test, and Launch
    - Operations
- Any concerns with FAR Part 12 terms and conditions: https://acquisition.gov/far/html/FARTOCP12.html
- Concept of operating hosted payload, including communications architecture
- Safety and mission assurance requirements levied upon hosted payload
- The level of NASA participation allowed during spacecraft development and instrument integration (e.g. spacecraft design reviews, environmental tests, etc.)

NASA is seeking capability statements from all interested parties, including Small, Small Disadvantaged (SDB), 8(a), Woman-owned (WOSB), Veteran Owned (VOSB), Service Disabled Veteran Owned (SD-VOSB), Historically Underutilized Business Zone (HUBZone) businesses, and Historically Black Colleges and Universities (HBCU)/Minority Institutions (MI) for the purposes of determining the appropriate level of competition and/or small business subcontracting goals.

No solicitation exists; therefore, do not request a copy of the solicitation. If a solicitation is released it will be synopsized in FedBizOpps and on the NASA Acquisition Internet Service. It is the potential offeror’s responsibility to monitor these sites for the release of any solicitation or synopsis.

Vendors having the capabilities necessary to meet or exceed the stated requirements are invited to submit appropriate documentation, literature, brochures, and references.

Please advise if the requirement is considered to be a commercial or commercial-type product. A commercial item is defined in FAR 2.101.

This synopsis is for information and planning purposes and is not to be construed as a commitment by the CII Project nor will the CII Project cover any costs for information submitted in response to the RFI.

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Technical questions should be directed to Craig Jones at Craig.D.Jones@nasa.gov. All other questions should be directed to Brad Gardner at Robert.B.Gardner@nasa.gov. All responses shall be submitted to Brad Gardner at Robert.B.Gardner@nasa.gov and to Craig Jones at Craig.D.Jones@nasa.gov no later than May 11, 2012. Respondents may e-mail files up to 10MB in size to Brad Gardner; respondents shall submit larger files on optical storage media (CD/DVD) via postal mail to the following address:

Brad Gardner
Office of Procurement
Building 2101, MS 12
NASA Langley Research Center
Hampton, VA 23681

Please reference CII-GEO in any response.

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Appendix G Instrument Modes

This section shows one way to set up a notional Instrument mode scheme and also provides context for those guidelines, especially data and electrical power, which reference various modes.

G.1 MODE GUIDELINES

<u>Basic Modes</u>
Instruments should support four basic modes of operation: OFF/SURVIVAL, INITIALIZATION, OPERATION, and SAFE (see Figure G-1). Within any mode, the Instrument may define additional sub-modes specific to their operation (*e.g.* STANDBY, DIAGNOSTIC, MEASUREMENT, etc.).

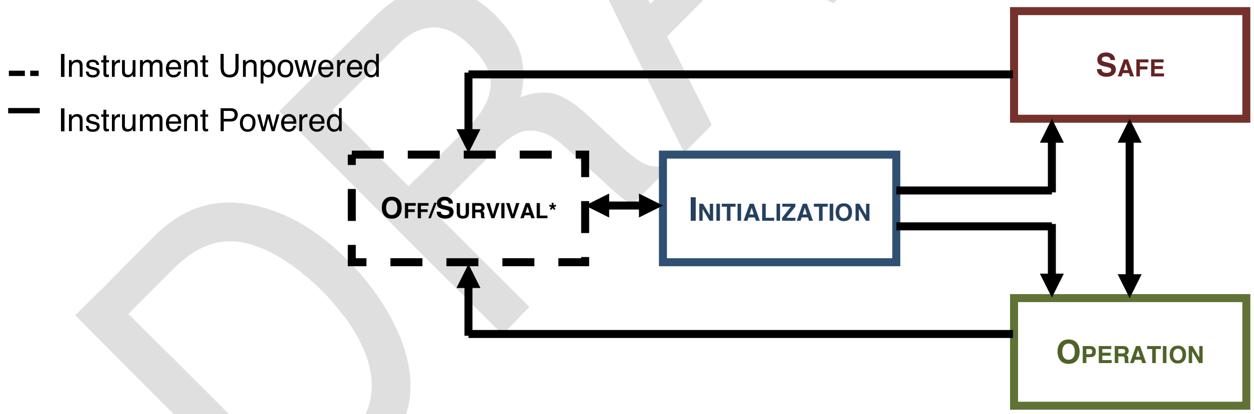

Figure G-1: Instrument Mode Transitions

<u>OFF/SURVIVAL Mode, Survival Heater OFF State</u>
The Instrument is unpowered, and the survival heaters are unpowered in survival heater OFF state of the OFF/SURVIVAL mode.

OFF/SURVIVAL Mode Power Draw
The Instrument should draw no operational power while in OFF mode.

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.

<u>OFF/SURVIVAL Mode, Survival Heater ON State</u>
The Instrument is unpowered, and the survival heaters are powered-on in the survival heater ON state of the OFF/SURVIVAL mode.

Spacecraft Verification of Instrument Survival Power
The Spacecraft should verify Instrument survival power is enabled upon entering the survival heater ON state of the OFF/SURVIVAL mode.

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..

Instrument Susceptibility to Unanticipated Transition to SURVIVAL Mode

The Instrument should be able to withstand the sudden and immediate transition to instrument OFF/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).

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.

Power Application

The Instrument should be in INITIALIZATION mode upon application of electrical power.

Thermal Conditioning

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

Command and Telemetry

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

Health and Status Telemetry

When in INITIALIZATION mode, the Instrument should send the Spacecraft health and status telemetry.

OPERATION Mode

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

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.).

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.

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

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).

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

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

G.2 MODE TRANSITIONS

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

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

SURVIVAL Mode Transitions
Trigger
The Spacecraft should transition the Instrument to OFF/SURVIVAL mode in the event of a severe Spacecraft emergency.

Instrument Operational Power
The Spacecraft should remove Instrument operational power during transition to OFF/SURVIVAL mode.

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

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

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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.

<u>SAFE Mode Transitions</u>

Command Trigger

The Instrument should transition to SAFE mode upon receipt of a command from the Spacecraft or ground operations team.

Missing Time Message Trigger

The Instrument should transition to SAFE mode upon the detection of 10 consecutive missing time messages.

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.

Orderly Transition

The Instrument should conduct all transitions to SAFE mode in an orderly fashion.

Duration of SAFE Mode Transition

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

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.

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.

<u>OPERATION Mode Transitions</u>

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.

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.

2700 # Appendix H Significant Differences among CII, ESA, and
2701 SMC Hosted Payload Guidelines

2702 **Table H-1: CII and ESA Hosted Payload Technical Guideline Differences**

Interface	NASA	ESA	Comments
Data Interface	SpaceWire, RS422, Mil-STD-1553	SpaceWire	
On-board data storage	Instrument	Spacecraft	
Power	28 ± 6 VDC	18 to 36 VDC	
Discrete PPS line	Optional	Required	
Redundancy	Optional	Required	Data, power, Survival Heaters
EMI/EMC	Tailored MIL-STD-461F Based on inputs	Will be tailored from MIL-STD-461F	Inputs from RFI responders
Overcurrent protection	Open	Latching Current Limiters (LCL)	

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