

Common Instrument Interface Project

# Hosted Payload Interface Guide for Proposers

Mar. 22, 2018

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Change Log

Version	Date	Section Affected	Description
1.0	10-11-2021	Various	Made several editorial and error corrections Reviewed and updated for clarity and internal consistency*

\* Update: although the technical content of this document has not been updated since the March 2018 release, this document was reviewed in October 2021 and it was felt that the technical content should still be of use to instrument developers who are proposing to an Earth Venture opportunity.

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### **Special Acknowledgement**

The development of the Hosted Payloads Interface Guide (HPIG) is the direct result of a collaborative effort between the NASA Common Instrument Interface (CII) Project, the Earth System Science Pathfinder Program Office, the USAF Space and Missile Center's Hosted Payload Office, and The Aerospace Corporation. The HPIG provides a prospective Instrument Developer with technical recommendations to assist them in designing an Instrument or Payload that may be flown as a hosted payload on commercial satellites flown in Low Earth Orbit (LEO), or Geostationary Earth Orbit (GEO). The document is now in its second iteration, with several updates and improvements.

We wish to express a special thanks to our teammates within the USAF Space and Missile Center's Hosted Payload Office, and The Aerospace Corporation. Their contributions have been remarkable, timely, and accurate. In addition, the members of the USAF Space and Missile Center's Hosted Payload Office, and The Aerospace Corporation have gone above and beyond their tasks as defined by the Memorandum of Understanding. They have provided not only impeccable technical inputs to the development of the Hosted Payload Interface Guidelines but have assisted in the establishment and conduct of workshops, and other special meetings and events.

Without the contributions of the USAF and the Aerospace Corporation, the development of this document would not have been possible!!

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## EXECUTIVE SUMMARY

The Hosted Payload Interface Guidelines for Proposers (HPIG) document provides a prospective Instrument Developer with technical recommendations to assist them in designing an Instrument or Payload that may be hosted on commercial satellites flown to Low Earth Orbit (LEO), or Geostationary Earth Orbit (GEO).

Easily hosted payloads exhibit the following characteristics:

- Well-defined interface and mission requirements
- Simple interfaces to minimize integration complexity
- On-time delivery to the host on time with no impact to satellite I&T schedule
- Operations decoupled from host satellite operations

Most importantly, Hosted Payloads adhere to the “Do No Harm” criteria levied by the host. In other words, the Payload shall prevent itself or any of its components from damaging or otherwise degrading the mission performance of the Host Spacecraft.

The cost of hosting is proportional to the Payload science and design criteria, size, integration complexity, and schedule. The guidelines herein generally provide the most-restrictive Payload-to-Host interfaces, with the caveat that more demanding designs may be accommodated for negotiated cost, as agreed upon with the Host.

When dealing with hosted payloads, it is unlikely that the hosting bus will be able to provide all the required interfaces or the performance necessary to satisfy the payload’s mission requirements.

Unlike the previous version of this document, the guidelines contained within the *HPIG for Proposers* are separated into two sections, one for LEO and one for GEO.

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# 1 OVERVIEW

## 1.1 Introduction

This Hosted Payload Interface Guide (HPIG) for External Payloads was developed by the NASA Common Instrument Interface Project and funded by the Earth System Science Pathfinder Program Office. This HPIG provides a prospective Instrument Developer with technical recommendations to assist them in designing an Instrument or Payload that may be flown as a hosted payload on commercial satellites flown in Low Earth Orbit (LEO), or Geostationary Earth Orbit (GEO). This document supersedes the Common Instrument Interface Project’s Hosted Payload Guidelines Document previously published by the NASA Earth System Science Pathfinder (ESSP) Program Office.

This document includes the instrument/payload accommodations of most commercial spacecraft, including interfaces and environments that must be met and demonstrated to “do no harm” to the host in order to be compatible for launch. Instruments/payloads that are designed to be compatible with these guidelines will have a higher likelihood of being compatible with any of the commercial satellite buses, and thus maximizing launch opportunities as a hosted payload.

This document is referenced by NASA in the Common Instrument Interface Best Practices document, which provides guidance to NASA specifications and standards for instrument design. The Best Practices document is distinct and separate from this document and is a document. The Best Practices document may be obtained from the (<http://science.nasa.gov/about-us/smd-programs/earth-system-science-pathfinder/>). In case of any questions, please contact the ESSP PO Chief Engineer, Barry Dunn – [barry.j.dunn@nasa.gov](mailto:barry.j.dunn@nasa.gov).

## 1.2 What is a Hosted Payload and Other Definitions?

The verb “should” denotes a recommendation. “Will” denotes an expected future event.

Hosted Payload or Instrument is used interchangeably with “payload” refers to an integrated payload or instrument on a commercial or Government host satellite that is dependent upon one or more of the host spacecraft’s subsystems for functionality or use of available capabilities to include mass, power, and/or communications.

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

Instrument: the hosted payload to which these guidelines apply.

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

Host Spacecraft: the spacecraft bus that will provide the resources to accommodate the instrument or payload.

Host Spacecraft Manufacturer: the organization responsible for manufacturing the Host Spacecraft.

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Satellite Operator: the organization or satellite owner responsible for on-orbit and ground operations throughout the Host Spacecraft’s lifetime.

Systems Integrator: the organization responsible for the system engineering and integrating of the complete system including the Instrument, Host Spacecraft, and Ground System.

Unless otherwise specified, all quantities in this document are in either base or derived SI units of measure.

### 1.3 How the Document Was Developed

The content of this document is aggregated from several sources. The CII Project’s HPIG team used personal engineering experience, publicly available information, and privately held information provided by industry to define the primary technical components of this document and to establish its content. The HPIG team, leveraging stakeholder feedback and numerous peer review workshops to guide efforts, with this document, seeks to establish the appropriate levels of breadth and depth of the source material as a means to generate a general all-encompassing guidelines document.

In order to increase the likelihood that a guideline-compliant Instrument design would be technically compatible with a majority of the host spacecraft, an “all-satisfy” strategy was adopted. 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 this guide. Where considered necessary, the CII Projects’ HPIG document 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 microsattellites 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*.

### 1.4 How to Use this Document

The HPIG is a guidelines document only. It is not a requirements document! The content of this document represents recommendations, not requirements, and should be used as interface design guidelines only by the proposer. The CII Project HPIG 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 flexible to adapt to expected host satellites and limits any (incorrectly inferred) compliance burdens. This document should be used primarily for pre-proposal and proposal

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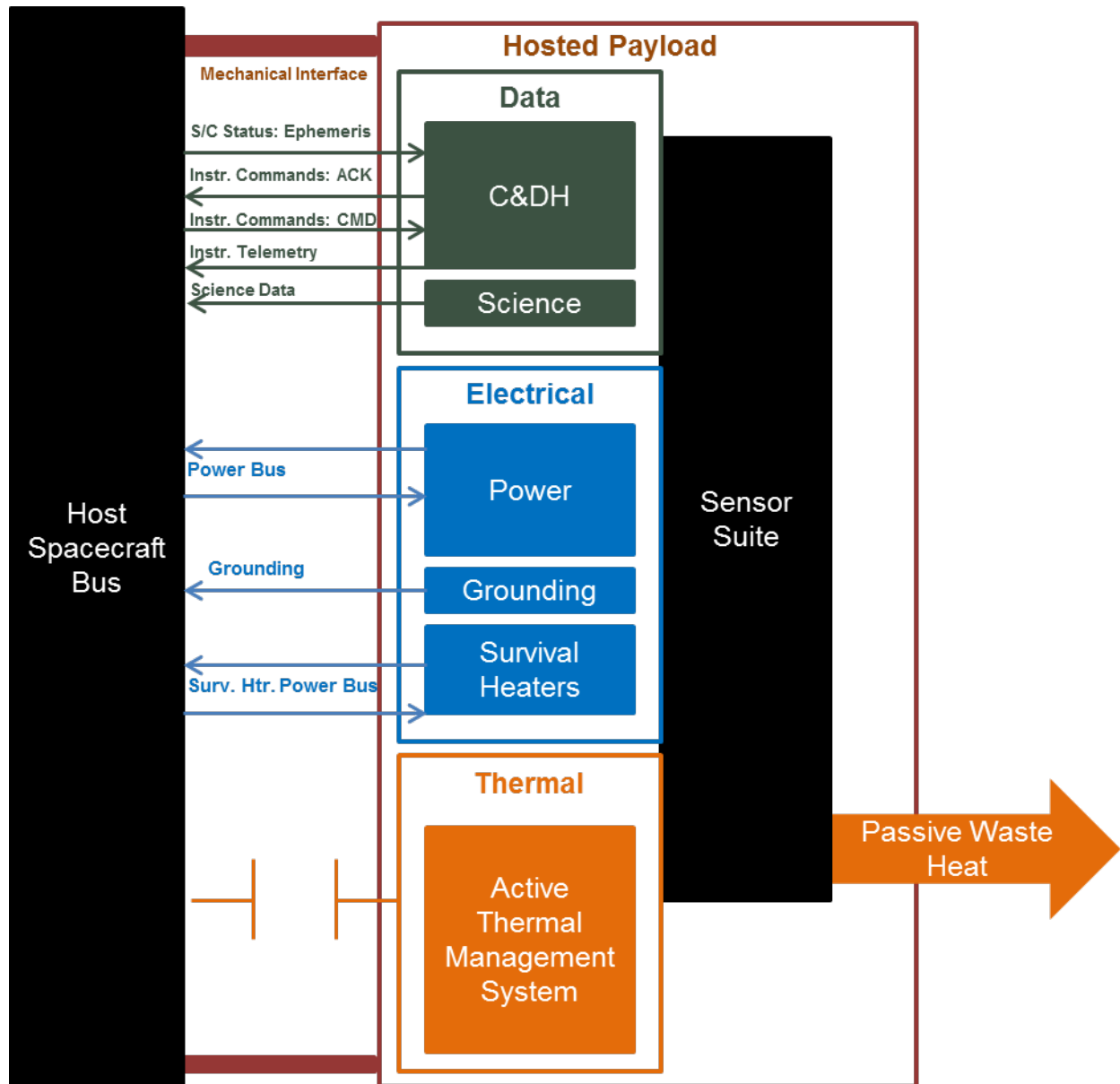
efforts. Once a user has determined a host spacecraft opportunity, then they will interface with the host spacecraft project for specific design and interface accommodations.

Instrument Developers are not required to comply with these guidelines, however conformance to these guidelines will enhance the host-ability of the payload to a commercial satellite host. These guidelines are not meant to replace Instrument Developer collaboration with Spacecraft Manufacturers, rather to provide familiarity of Spacecraft interfaces and accommodations in order to assist with such collaboration. Instruments that do not comply with guidelines specified in this document can be accommodated with additional resources that either offset the impact to existing HPO designs (*e.g.*, investments enhancing Instrument capability) or propose to enable compatibility after minor alterations to spacecraft performance (*e.g.*, investments enhancing Spacecraft capability). 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, the government can take advantage of the commercial space industries best practices and profit incentives to fully realize the benefits of hosted payloads. Because the financial contribution by the Instrument, via hosting fees, to the Satellite Operator are significantly smaller than the expected revenue of satellite operations, the government 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 Design Guidelines. With this exception, programmatic and business aspects of hosted payloads are outside the scope of this document.

One limitation of the “all-satisfy” strategy is that it constrains all instrument accommodation parameters to a greater degree than might be expected once the Instrument is paired with a Host Spacecraft. One size does not fit all in Hosted Payloads. Spacecraft Manufacturer tailor their bus design to each Satellite Operator’s requirements, which may allow Instrument Developers to negotiate an agreement for 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, but could significantly impact the cost and schedule of the program. Because the Instrument to Host Spacecraft pairing occurs in the vicinity timeframe of Key Decision Point (KDP) C (or instrument CDR), certain knowledge of these available accommodation resources will be delayed well into the Instrument’s development timeline.

## 1.5 Scope

This document’s scope is comprised of primarily interface guidelines. Figure 1-1 uses color to identify the scope: colored components are in scope; black components are out of scope.



**Figure 1-1: Hosted Payload Interfaces**

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. Even though this Figure does not contain environments, this document will provide guidelines on the environments as well. While guidelines are not requirements—using the verb “should” instead of “shall”—they try to follow the rules of writing proper requirements.



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The lessons learned capture additional technical information, including modifications agreed to by Host Spacecraft Manufacturers, from previous hosted payload studies that developers may find useful. Lessons learned can be found in Appendix A

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 bound the trade space.

Because the Host Spacecraft and Instrument begin development simultaneously and independently, some parameters will not be resolved prior to Instrument-to-Host Spacecraft pairing. These parameters are defined herein as Negotiated Parameters to be agreed upon by all participating parties as development continues. 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.

## 1.6 Document Heritage

As stated in Section 1.1, the guidelines contained herein were developed by NASA's Common Instrument Interface Project, and are the result of a collaborative project between the NASA Common Instrument Interface (CII) team, and LaRC's Earth System Science Pathfinder (ESSP) Program Office.

The CII Project HPIG team plans to release updated guidelines in 18-month intervals. 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 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.

The CII Project HPIG team has been working very closely with ESA over the past few years on a unified set of guidelines for electrical power and data interfaces in the LEO domain. One important note is that the ESA elements will be prescriptive requirements as opposed to this HPIG. Due to different sets of common practices between the American and European space industries, a small number of technical differences exist between the HPIG guidelines and ESA requirements that are summarized in the following Table 1-1:

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**Table 1-1: HPIG and ESA Hosted Payload Technical Guideline Differences**

<b>Interface</b>	<b>NASA</b>	<b>ESA</b>	<b>Comments</b>
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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## 2 Hosted Payload Imperative – Do No Harm

**The Instrument shall prevent itself or any of its components from propagating failures, damaging, or otherwise degrading the mission performance of the Host Spacecraft or any other payloads.**

Rationale: From the Host Satellite’s perspective, the most important constraint on a hosted payload is to “do no harm” to the Host Spacecraft or other payloads. The Satellite Operator has the ultimate authority to accept (or deny) the launch of a hosted payload if it poses an unacceptable risk to the host satellite’s mission. This decision point occurs after the completion of spacecraft system level testing (with the hosted payload) and prior to the spacecraft ship to the launch site. It should be noted that payload design reviews, mission integration analyses, and test results provide incremental confidence that “do no harm” criteria is being satisfactorily addressed prior to final acceptance by the satellite manufacturer and/or the satellite operator.

The Satellite Operator will have the authority and capability 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 Host 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.

This document serves as a set of guidelines to understanding potential interfaces and increasing hosting opportunities. Moreover, these guidelines serve to highlight critical interfaces where the “Do No Harm” imperative needs to be satisfactorily addressed. Potential sources of harm include, but are not limited to:

- Noise and offsets due to ground loops
- Coupling of Electrostatic Discharge
- Arcing due to partial pressure from inadequate venting
- Contamination of thermal and optical surfaces
- Glint and other field of view violations
- Ripple from antenna side lobes and other RF interference
- Use of more resources than allocated, or allocation of less resources than promised
- Noise, shorts or excessive loading on shared data buses
- Attitude disturbance or mechanical damage from moving or unsecured items
- Interlock configurations or spurious emissions that violate launch safety regulations
- Damage due to the launch dynamics environment

It is emphasized that the guidelines do not prescribe a “cookie cutter” basis for hosting payloads, and should not be used to replace the standard systems engineering process. It is important that both the Host Spacecraft and Hosted Payload ensure a mutual understanding of the interfaces necessary for a successful partnership from both parties’ perspectives.

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The Hosted Payload should:

- Adhere strictly to interfaces as negotiated with the Host
- Identify any unmitigated propagating faults so they may be mitigated by the Host
- Stay within agreed upon allocations
- Not impact the primary mission beyond agreed upon constraints

The “Do No Harm” effects resulting from “hard failures” of parts or connections, or interference on required Host Spacecraft performance requirements need evaluation. Interface checklists identifying incompatibilities or gaps are useful in addressing these concerns. Recommended analyses (e.g., worst-case, timing, stress, failure modes and effects, etc.) for determining the extent of these gaps may be used as inputs for selecting the appropriate mitigations during the design process.

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### 3 DESIGN GUIDELINES FOR LEO

#### 3.1 Assumptions

The HPIG guidelines assume the following regarding the Host Spacecraft:

- 1) Hosted Payload: The Host Spacecraft will have a primary mission different from that of the Instrument.
- 2) 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.
- 3) Responsibility for Integration: The Host Spacecraft Manufacturer will integrate the Instrument onto the Host Spacecraft with support from the Instrument Developer.

#### 3.2 Mission Risk

**The Instrument should comply with Mission Risk class, as required by the customer such as the AO.**

**Mission risk classes are a measure of the instrument’s mission design and reliability. Regardless of the instrument’s mission risk class, the instrument must conform to the host “do no harm” requirements.**

Rationale: Future AOs will specify the risk class of the instrument.

#### 3.3 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.”

#### 3.4 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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### 3.5 Data Guidelines

#### 3.5.1 Assumptions

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

1. During the pairing 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.

#### 3.5.2 Data Interface

**The Instrument-to-Host Spacecraft data interfaces should use RS-422, SpaceWire, LVDS, or MIL-STD-1553.**

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

#### 3.5.3 Data Accommodation

**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 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 that 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 3.5.10), 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).

#### 3.5.4 Command Dictionary

**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.*).

#### 3.5.5 Telemetry Dictionary

**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

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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.

### 3.5.6 Safe mode

**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 Host Spacecraft resources (e.g. power).

Note: Please see Appendix D for a discussion of the notional instrument mode scheme referenced in this document.

### 3.5.7 Command (SAFE mode)

**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.

### 3.5.8 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 Host Spacecraft Manufacturer, Satellite Operator, and ground operations team to devise and operate Fault Detection Isolation, and Recovery (FDIR) procedures, crucial for on-orbit operations.

### 3.5.9 Command (Acknowledgement)

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

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

### 3.5.10 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 Host Spacecraft. A spacecraft needs only enough buffer capacity to relay Instrument telemetry. Fewer resource impacts on the spacecraft maximize Instrument hosting opportunities.

## 3.6 **Electrical Power System Guidelines**

### 3.6.1 Assumptions

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

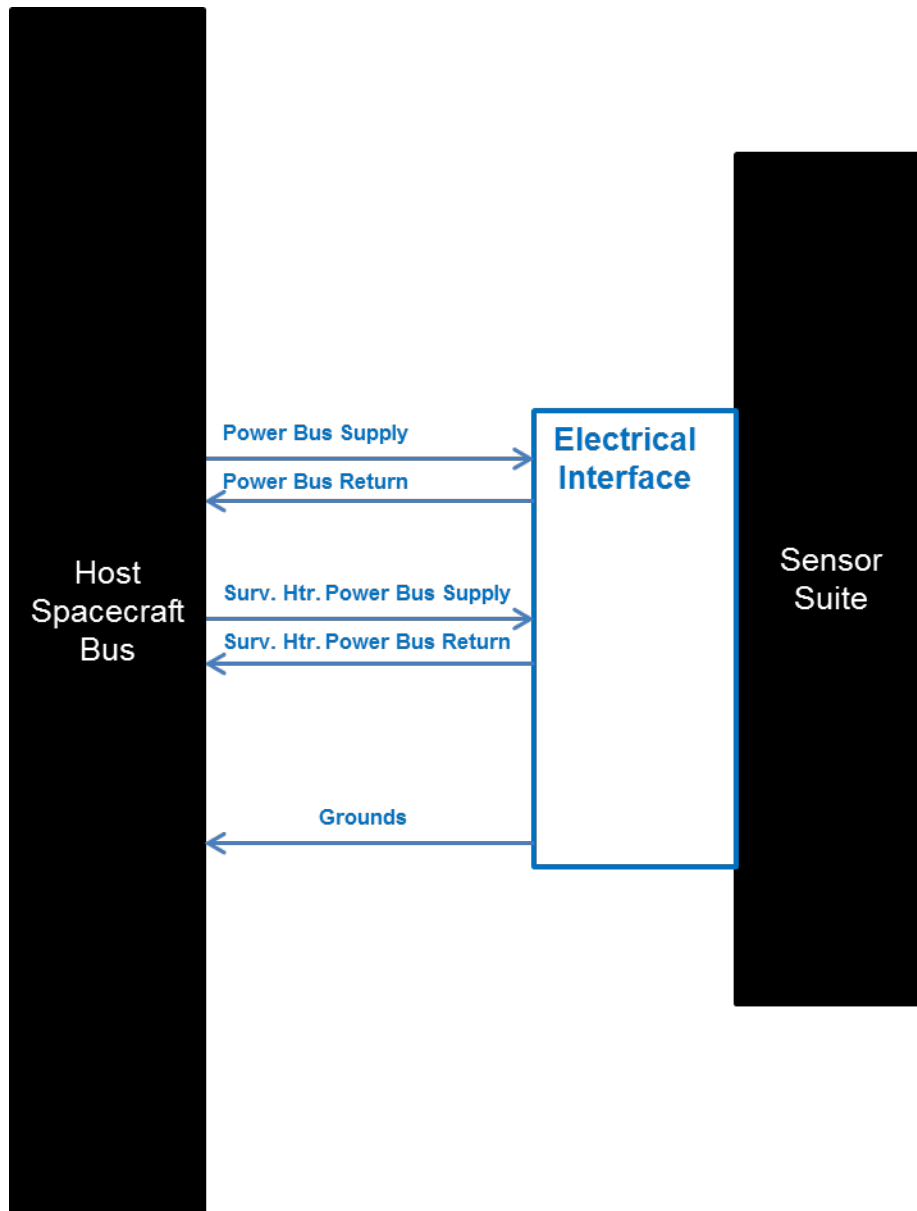
- 1) During the pairing process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the electrical power interface.

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The Electrical Power Interface Control Document (EICD) will record those parameters and decisions.

- 2) The Host Spacecraft will supply to the Instrument EPS unregulated (sun regulated) electrical power within the range of  $28 \pm 6$  VDC, including ripple and normal transients as defined below, and power distribution losses due to switching, fusing, harness and connectors.
- 3) The Host Spacecraft will provide connections to 100W (Orbital Average Power: OAP) power buses as well as a dedicated bus to power the Instrument's survival heaters. Orbital Average Power is the total power used by the payload over one orbit divided by orbit time (usually 90 minutes).
- 4) The Host Spacecraft will energize the Survival Heater Power Bus at approximately 30% (or possibly higher, as negotiated with the host provider) of the OAP in accordance with the mission timeline documented in the EICD.
- 5) The Host Spacecraft Manufacturer will supply a definition of the maximum source impedance by frequency band. Table 3-1 provides an example of this definition.
- 6) The Host Spacecraft Manufacturer will furnish all Host Spacecraft and Host Spacecraft-to-Instrument harnessing.
- 7) 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.
- 8) The Host Spacecraft will protect its own electrical power system via overcurrent protection devices on its side of the interface.
- 9) 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.
- 10) 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. The Host Spacecraft will continue to provide Survival Heater Power, but may remove Survival Heater Power if conditions deteriorate significantly.
- 11) 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 3-1: Host Spacecraft-Instrument Electrical Interface**

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**Table 3-1: Example of Power Source Impedance Function**

Frequency	Maximum Source Impedance [ $\Omega$ ]
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

### 3.6.2 Grounding

**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.

#### 3.6.2.1 *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.

### 3.6.3 Power Return

**The Instrument electrical power return should be via dedicated return line**

Rationale: The Instrument Electrical Power System (EPS) should return electrical power via electrical harness or ground strap to reduce the potential to introduce stray currents or ground loop currents into the Instrument, Host Spacecraft, or other payloads.

### 3.6.4 Power Supply Voltage

**The Instrument EPS should accept an unregulated input voltage of  $28 \pm 6$  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.

### 3.6.5 Power Bus Interface

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

12) 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.

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

### 3.6.6 Survival Heater Bus Interface

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

14) Rationale: The Survival Heaters, which are elements of the Thermal subsystem, require power to heat certain instrument components during off-nominal scenarios when the Power

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Buses are not fully energized. See Best Practices appendix Document for more discussion about survival heaters.

### 3.6.7 Bonding

**The Instrument bonding should comply with NASA-STD-4003 or equivalent.**

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

### 3.6.8 Mitigation of In-Space Charging Effects

**The Instrument should comply with NASA-HDBK-4002A, or equivalent, 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 Host Spacecraft components and external surfaces related to space plasmas and high-energy electrons and the consequences of that charge buildup.

### 3.6.9 EPS Accommodation

This section specifies the characteristics, connections, and control of the Host 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.

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.

#### 3.6.9.1 *Instrument Power Harness*

**Instrument power harnesses should be sized to the largest possible current value as specified by 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.

#### 3.6.9.2 *Allocation of Instrument Power*

**The EPS should draw no more power from the Host Spacecraft in each Instrument mode than defined in Table 3-2.**

Rationale: The 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.

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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 D.

**Table 3-2: Instrument Power Allocation**

Mode	LEO	
	Peak (W)	Average (W)
Off/ Survival	0/60	0/30
Activation	200	100
Safe	100	50
Operation	200	100

#### 3.6.9.3 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.

#### 3.6.9.4 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 could accidentally occur during assembly, integration, and test (AI&T).

#### 3.6.9.5 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. Ideally, the Instrument should power up in the minimum power draw state of the OFF/SURVIVAL Mode and then transition into the minimum power draw state of the INITIALIZATION Mode. The +28 VDC is inclusive of nominal voltage transients of  $\pm 6$  VDC for LEO Instruments.

#### 3.6.9.6 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.

### 3.7 Mechanical Guidelines

#### 3.7.1 Assumptions

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

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- 1) 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. (It should be noted that FOV requests are best accommodated during the initial configuration of the host. Therefore, FOV may be a limiting factor in determining which host spacecraft is a viable candidate for your payload.)
- 3) The Host Spacecraft Manufacturer will furnish all instrument mounting fasteners.
- 4) The Host Spacecraft Manufacturer will provide a glint analysis that demonstrates that no reflected light impinges onto the Instrument FOV, if requested by the Instrument Developer.
- 5) The Host Spacecraft Manufacturer will furnish the combined structural dynamics analysis results to the Instrument Developer.

### 3.7.2 Mechanical Interface

**The Instrument should be capable of fully acquiring science data when directly mounted to the Host Spacecraft. If precision mounting is required, the Instrument Provider should assume supplying a mounting plate to meet those requirements. Such an accommodation could affect the Instrument Providers mass budget.**

Rationale: Broad survey of industry hosted payload accommodations 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.

### 3.7.3 Mechanical Accommodation

#### 3.7.3.1 *Mass*

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

Rationale: Based on broad survey of industry hosted payload accommodations, instrument mass of approximately 30kg and below would provide maximum opportunity for finding a host. Opportunities above 100kg may exist but provide only a minimum hosting probability.

#### 3.7.3.2 *Volume*

**The Instrument and all of its components should remain within a volume of 0.15 m<sup>3</sup> during all phases of flight.**

Rationale: Based on broad survey of industry hosted payload accommodations, instrument volume of approximately 0.02 m<sup>3</sup> and below would provide maximum opportunity for finding a host. Opportunities above 0.15 m<sup>3</sup> may exist but provide only a minimum hosting probability.

### 3.7.4 Functionality in 1 g Environment

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

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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.

### 3.7.5 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.

### 3.7.6 Moveable Masses

**The Instrument should compensate for the momentum associated with the repetitive movement of large masses, relative to the mass of the Host Spacecraft.**

Rationale: This prevents moveable masses from disturbing the operation of the Host Spacecraft or other payloads. This will generally not apply to items deploying during startup/initiation of operations, and the applicability of the guideline will be negotiated with the Host Spacecraft Manufacturer and/or Satellite Operator during pairing.

### 3.7.7 Minimum Fixed-Base Frequency

**The Instrument should have a fixed-base frequency greater than 70 Hz.**

Rationale: Based on broad survey of industry hosted payload accommodations, this minimum fixed-based frequency meets or exceeds the composite guidance of a majority of the responding (LEO) Host Spacecraft manufacturers. Opportunities down to 25 Hz may exist but significantly decrease hosting possibilities. To some extent, the Instrument will affect the Host Spacecraft frequency depending on the payload's mass and mounting location. Host Spacecraft Manufacturers may negotiate for a greater fixed-based frequency for hosted payloads until the maturity of the instrument can support Coupled Loads Analysis.

## 3.8 Thermal Guidelines

### 3.8.1 Assumptions

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

- 1) During the pairing 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 its own side of the interface from the Integration through Disposal portions of its lifecycle.
- 3) The Host Spacecraft Manufacturer will be responsible for thermal hardware used to close out the interfaces between the Instrument and Host Spacecraft, such as closeout Multi-layer Insulation (MLI).

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### 3.8.2 Thermal Interface

**The Instrument should be thermally isolated from the Host Spacecraft.**

Rationale: As a hosted payload, the Instrument should manage its own heat transfer needs without depending on the Host Spacecraft.

### 3.8.3 Thermal Design at the Mechanical Interface

**The Instrument thermal design should be decoupled from the Host Spacecraft at the mechanical interface between the spacecraft and neighboring payloads to the maximum practical extent.**

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.

### 3.8.4 Conductive Heat Transfer

**The conductive heat transfer at the Instrument-Host Spacecraft mechanical interface should be less than 15 W/m<sup>2</sup> or 4 W, whichever is greater.**

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

### 3.8.5 Radiative Heat Transfer

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

Rationale:

- 1) 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.
- 2) Hosted payload with science instruments requiring radiators operating at cold temperatures (below 25 °C) should consider the backloading and field of view blockage from parts of the spacecraft on the radiators. Although configuration details will be unknown prior to Instrument-to-Host pairing, the Hosted Payload should assess its sensitivity to some degree of blockage. For example, solar array and antennas can impose significant backloading if the radiator has any view of them (see Table 3-3 and Figure 3-2).

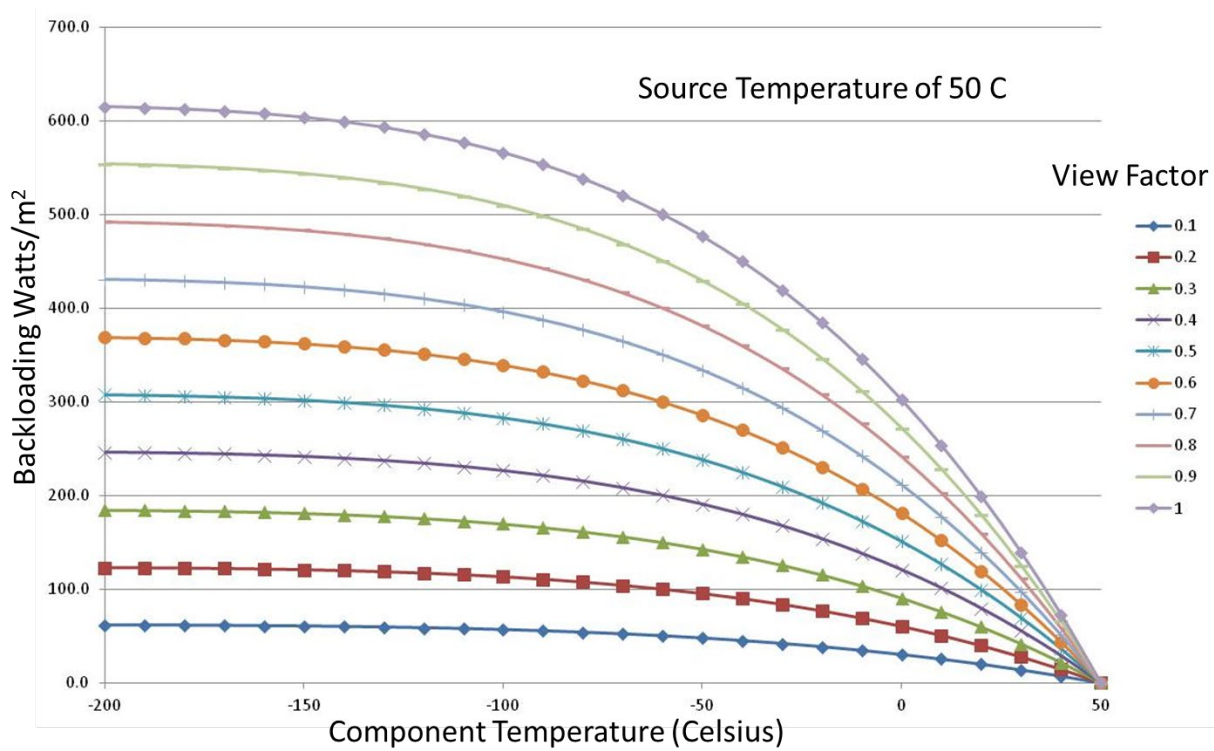
**Table 3-3: Worst-case Backloading on Payload Radiator**

S/C Source Temp., °C	Payload Radiator Temp., °C	Load, W/m <sup>2</sup> VF=0.1	Load, W/m <sup>2</sup> VF=0.2
50	50	0	0
50	40	7	15
50	30	14	28
50	20	20	40

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50	10	25	51
50	0	30	60
50	-10	35	69
50	-20	38	77
50	-30	42	84
50	-40	45	90
50	-50	48	95
50	-60	50	100
100	50	48	96
100	40	55	110
100	30	62	124
100	20	68	136
100	10	73	147
100	0	78	156
100	-10	82	165
100	-20	86	173
100	-30	90	180
100	-40	93	186
100	-50	96	191
100	-60	98	196





**Figure 3-2: Worst-case Backloading on Payload Radiator**

**The Instrument should maintain its own instrument temperature requirements.**

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

**3.8.6 Temperature Maintenance Responsibility**

**The Instrument should maintain its own instrument temperature requirements.**

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

**3.8.7 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.

**3.8.8 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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### 3.9 Instrument Models

**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.

### 3.10 Environmental Guidelines

#### 3.10.1 Assumptions

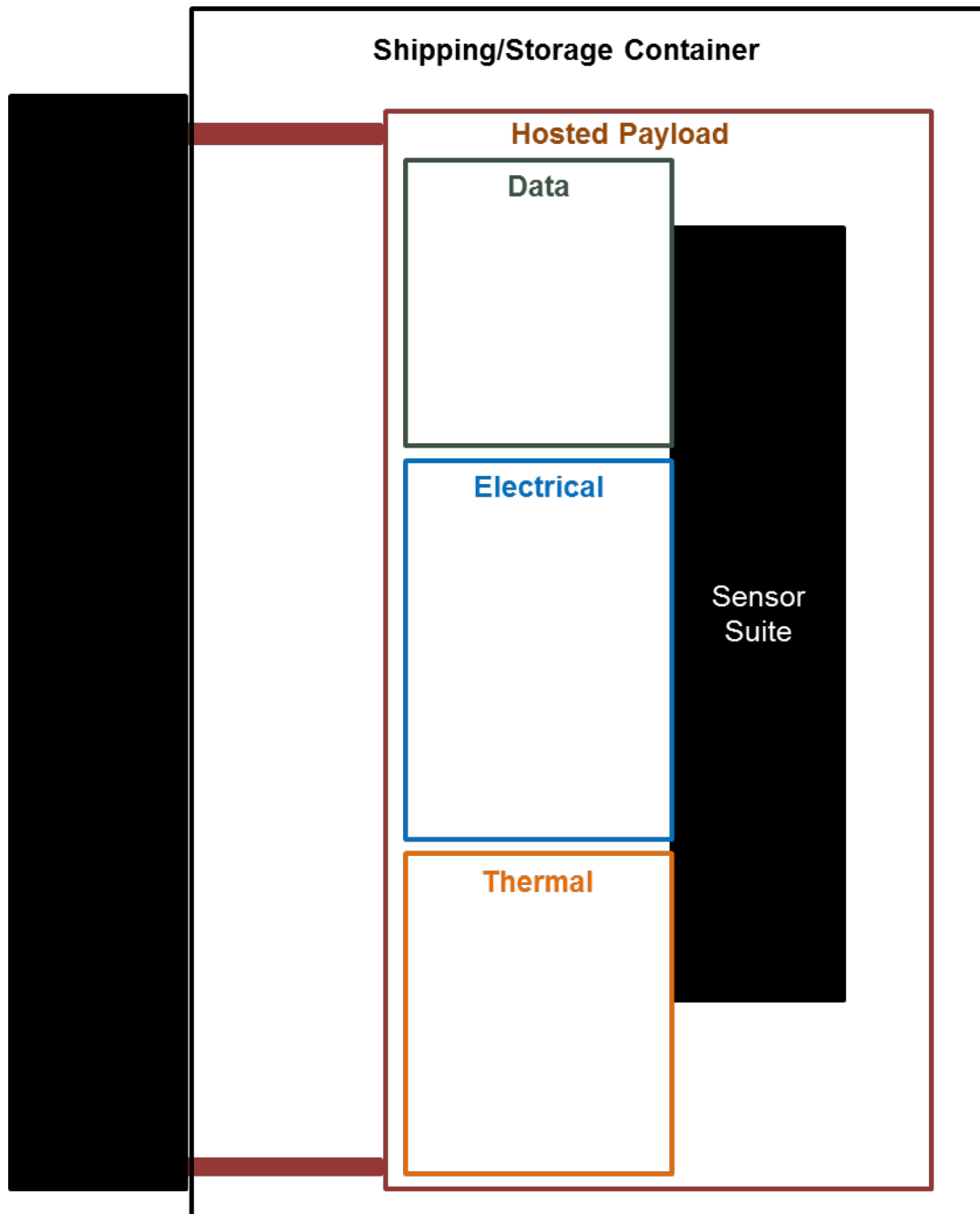
The HPIG environmental guidelines assume the following regarding the Host Spacecraft, launch vehicle, and/or integration and test facilities:

- 1) During the pairing process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the environmental interface. The Environmental Requirements Document (ERD) 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 D.

#### 3.10.2 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 Host Spacecraft Manufacturer/Systems Integrator. The Instrument is dormant and attached mechanically to its container (see Figure 3-3).



**Figure 3-3: Shipping / Storage Environment**

*3.10.2.1 Documentation*

The ERD will document the maximum allowable environment the Instrument will experience between the departure from the Instrument assembly facility and arrival at the Host Spacecraft integration facility.

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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.

### *3.10.2.2 Instrument Configuration*

**The ERD 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.

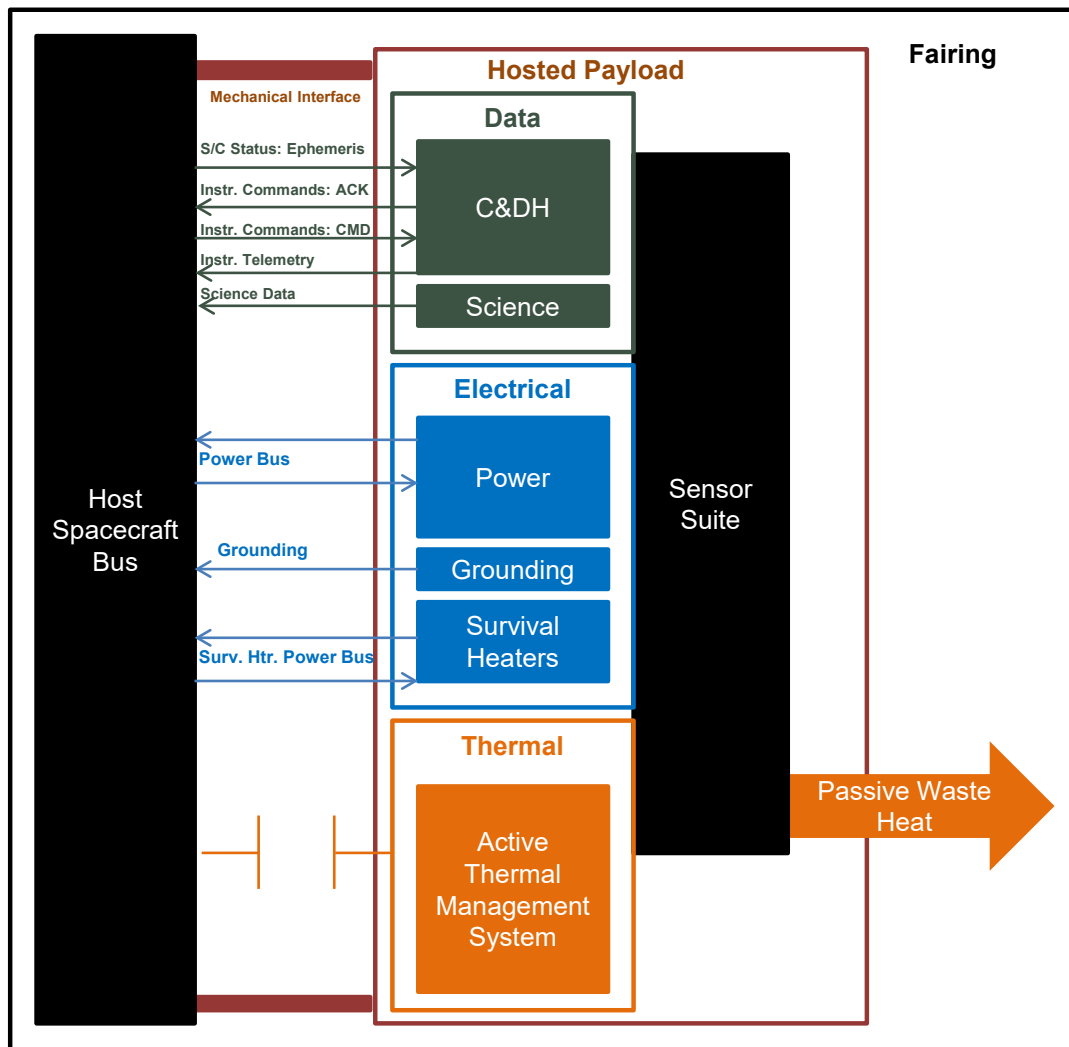
The Instrument will likely be in the OFF/SURVIVAL mode while in this environment.

### 3.10.3 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 Host 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 spacecraft bus and verify that system performs as designed throughout various environmental and dynamics regimes. The Instrument may be attached to the spacecraft bus or to various ground support equipment that transmits power, thermal conditioning, and diagnostic data (see Figure 3-4).

The instrument should be designed to minimize integrated tests with the spacecraft during the system level I&T phase. This is especially important during test activities in the environmental chambers. To the extent practical for the instrument, all performance testing should be performed prior to arrival at the spacecraft facility. Interface compatibility should be tested and the instrument should be powered down for the majority of spacecraft system level activities. This approach is to minimize schedule, cost, and complexity with the host.

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**Figure 3-4: Integration and Test Environment**

### 3.10.3.1 Documentation

**The ERD will document the maximum allowable 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 ERD may either record these data explicitly or refer to a negotiated Test and Evaluation Master Plan (TEMP).

### 3.10.3.2 Instrument Configuration

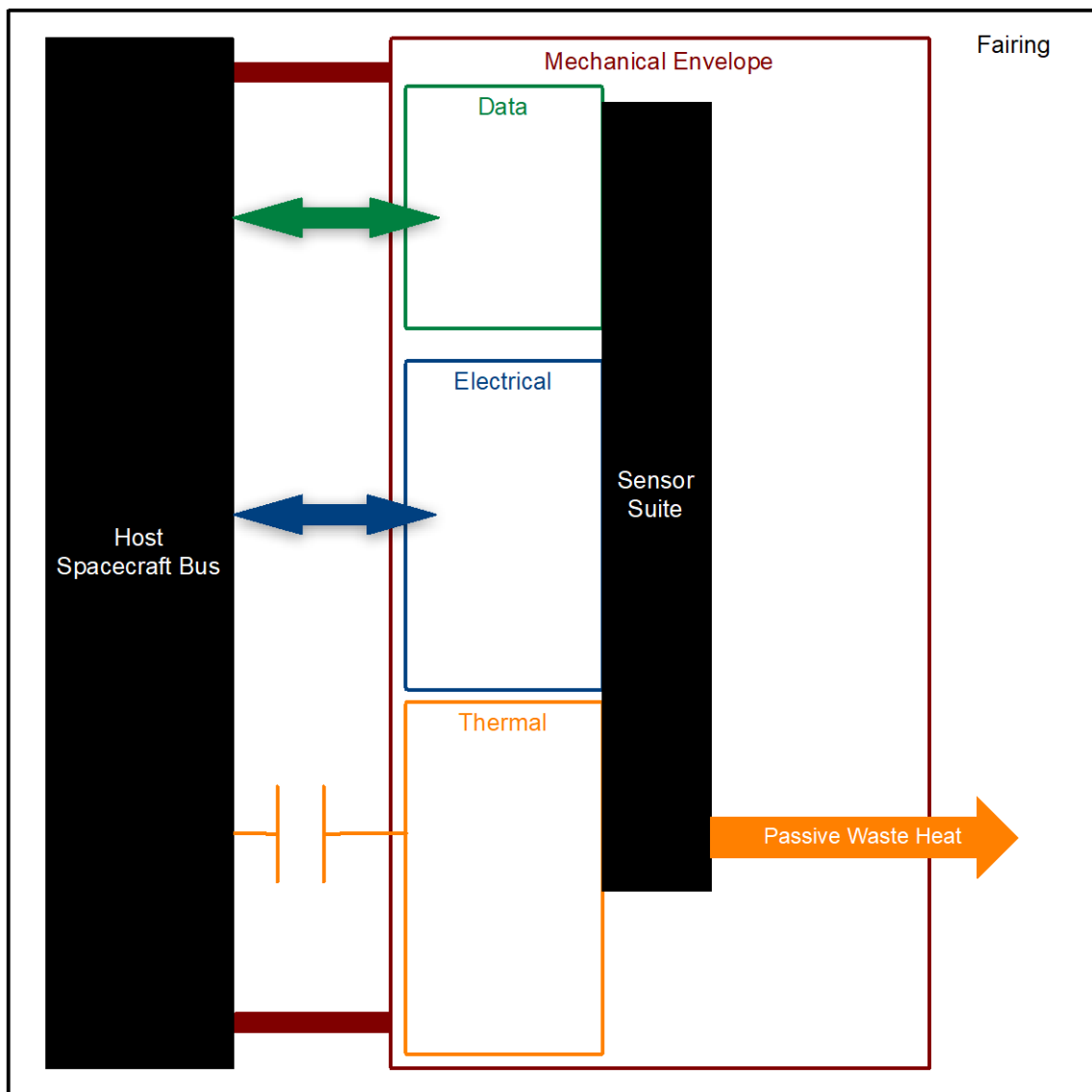
**The ERD will document the configuration and operational mode of the Instrument during the Integration and Test phase.**

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

### 3.10.4 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 3-5).



**Figure 3-5: Launch Environment**

#### 3.10.4.1 Documentation

**The ERD will document the maximum allowable environments the Instrument will experience between Launch and Host Spacecraft / Launch Vehicle separation.**

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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.

#### 3.10.4.2 Instrument Configuration

**The ERD 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.

#### 3.10.4.3 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, damage, 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 launch vehicles.

#### 3.10.4.4 Quasi-static Acceleration

**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 3-4**

**Table 3-4: Mass Acceleration Curve Design Limit Loads**

Mass [kg]	Limit Load [g] (any direction)
1	68.0
5	49.0
10	39.8
20	31.2
40	23.8
60	20.2
80	17.8
100	16.2
125	14.7
150	13.5
175	12.6
200 or Greater	12.0

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Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft or other payloads. This guidance represents the need to be compatible with the quasi-static loads that will be experienced during launch ascent. The LEO guideline is the all-satisfy strategy scenario, and the loads shown in Table 3-4 should be updated based on a launch vehicle specific set of MAC loads or the results of coupled loads analysis when this information becomes available.

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

#### 3.10.4.5 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 3-5.**

**Table 3-5: Sinusoidal Vibration Environment**

Frequency (Hz)	Amplitude	
	Flight Level	Protoflight/ Qual Level
5 – 20	12.7 mm (double amplitude)	16 mm (double amplitude)
20 – 100	10.0	12.5 g
Protoflight/Qual Sweep Rate: From 5 to 100 Hz at 4 octaves/minute		
Flight Level Sweep Rate: From 5 to 100 Hz at 2 octaves/minute except from 40 to 55 Hz at 6 Hz/min		
Input levels may be notched to limit component CG response to the design limit loads specified in Table 3-1		

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft or other payloads. Table 3-5 provides a generic sine environment for the preliminary design of components and subsystems. The sine sweep vibration levels shown in Table 3-5 are defined at the hardware mounting interface. This guidance represents the need to be compatible with the coupled dynamics loads that will be experienced during ground processing and launch ascent.

#### 3.10.4.6 Random Vibration

**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 3-6.**

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.



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**Table 3-6: Random Vibration Environment (derived from GEVS-SE, Table 2.4-4)**

Zone/Assembly	Frequency (Hz)	Protoflight / Qualification	Acceptance
Instrument	20	0.026 g <sup>2</sup> /Hz	0.013 g <sup>2</sup> /Hz
	20 – 50	+6 dB/octave	+6 dB/octave
	50 - 800	0.16 g <sup>2</sup> /Hz	0.08 g <sup>2</sup> /Hz
	800 - 2000	-6 dB/octave	-6 dB/octave
	2000	0.026 g <sup>2</sup> /Hz	0.013 g <sup>2</sup> /Hz
	Overall	14.1 g <sub>rms</sub>	10.0 g <sub>rms</sub>

Table 3-6 represents the random vibration environment for instruments with mass less than or equal to 25 kg and having resonant frequencies greater than 80 Hz. 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  $\pm 6$  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<sup>2</sup>/Hz at 20 Hz and at 2000 Hz for qualification testing and an ASD of 0.005 g<sup>2</sup>/Hz at 20 Hz and at 2000 Hz for acceptance testing.
- 3) The random vibration levels given in Table 3-6 should be updated based on test data or acoustic analysis of the payload once the launch vehicle specific acoustic environment has been defined
- 4) Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft 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. Based upon CII analysis of the following sources of performance data: the *The CII Guidelines Document, Revision A, GEVS-SE* and the *USAF HoPS studies data*, an overall protoflight/qual level of 23.1 g (rms) would maximize hosting opportunity.

#### 3.10.4.7 Acoustic Noise

**The Instrument should function according to its operational specifications after being subjected to a launch vehicle-induced transient environment. A generic acoustic environment is shown in Table 3-7.**

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**Table 3-7: Acoustic Noise Environment**

<b>1/3 Octave Band Center Frequency (Hz)"</b>	<b>Design/Qual/Protoflight (dB w/ 20 µPa reference)"</b>	<b>Acceptance (dB w/ 20 µPa reference)"</b>
20	129.5	126.5
25	130.7	127.7
31.5	130.0	127.0
40	131.5	128.5
50	133.0	130.0
63	134.5	131.5
80	135.5	132.5
100	136.0	133.0
125	136.8	133.8
160	136.7	133.7
200	136.0	133.0
250	136.0	133.0
315	136.0	133.0
400	134.0	131.0
500	132.0	129.0
630	131.4	128.4
800	131.6	128.6
1000	129.9	126.9
1250	126.1	123.1
1600	121.3	118.3
2000	119.5	116.5
2500	118.0	115.0
3150	116.1	113.1
4000	115.5	112.5
5000	114.8	111.8
6300	114.0	111.0
8000	113.0	110.0
10000	112.1	109.1

Rationale: Acoustic design guidelines are an envelope of a number of common launch vehicles. This acoustic environment should be used for preliminary design of components and subsystems if a specific launch vehicle has not been defined. While all hardware should be assessed for sensitivity to direct acoustic impingement, unless the component or subsystem has structure which is light-weight and has large surface area (typically a surface to weight ratio of > 150 in<sup>2</sup>/lb), it is expected that the random environment specified in Section 3.10.4.6 will be the dominant high-frequency loading condition rather than the acoustic environment defined in Table 3-7.

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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 / proto-flight exposure time is 2 minutes; acceptance exposure time is one minute.

#### 3.10.4.8 Mechanical Shock

**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 3-8.**

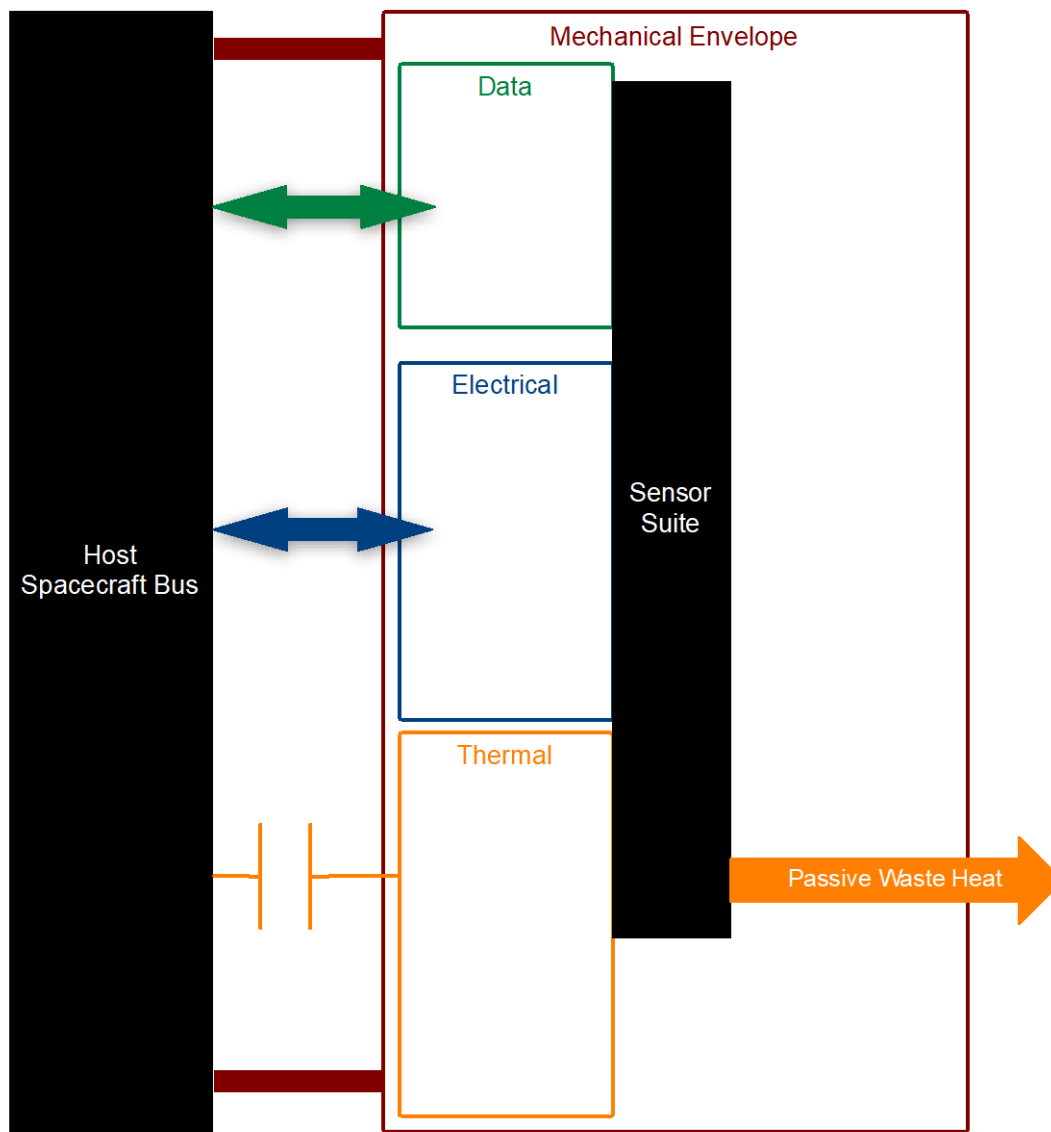
**Table 3-8: Shock Response Spectrum (Q=10)**

Frequency (Hz)	Acceptance Level (g)	Protoflight/Qualification (g)
100	160	224
630	1000	1400
10000	1000	1400
	The shock levels given assume that a component is located at least 60 cm (2 ft) from a shock source	The shock levels given assume that a component is located at least 60 cm (2 ft) from a shock source

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft 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. Table 3-8 provides a generic shock environment that may be used for hardware design until the mission specific shock environments can be defined. Based on broad survey of industry hosted payload accommodations, designing for higher shock levels (up to 5000 g for 1600 Hz and 10000 Hz) would maximize opportunity for finding a host. After pairing, the levels shown in Table 3-8 should be updated once all payload shock sources have been defined.

#### 3.10.5 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 3-6).



**Figure 3-6: Operational Environment**

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

#### *3.10.5.1 Orbital Acceleration*

**The Instrument should function according to its operational specifications after being subjected to a maximum spacecraft-induced acceleration of 0.15g.**

Rationale: The Instrument in its operational configuration 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 Host Spacecraft or other payloads. This guidance represents the need to be compatible with the accelerations that will be experienced on

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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*.

#### 3.10.5.2 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  $1.3 \times 10^{-15}$  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, damage, or inducing degraded performance of or damage to the Host Spacecraft 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*.

#### 3.10.5.3 Thermal Environment

**The Instrument should function according to its operational specifications after being subjected to a thermal environment characterized by Table 3-9.**

**Table 3-9: Thermal Radiation Environment**

Domain	Solar Flux [W/m <sup>2</sup> ]	Earth IR (Long Wave) [W/m <sup>2</sup> ]	Earth Albedo
LEO	1290 to 1420	222 to 233	0.275 to 0.375

Rationale: The Instrument must be able to withstand conditions typical of the on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft or other payloads. The Host Spacecraft Manufacturer will document the expected Free Molecular Heating rate seen by the exposed surface of the payload during the launch ascent in the TICD. This guidance defines the solar flux over the entire spectrum. In the UV portion of the spectrum ( $\lambda \leq 300$  nm), the solar flux is approximately 118 W/m<sup>2</sup> and the integrated photon flux is approximately  $2.28 \times 10^{15}$  photons/cm/sec. Reference NASA TM4527 for additional detail regarding the UV spectrum and associated photon flux.

#### 3.10.5.4 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.

3.10.5.5 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 3-10 shows the expected total ionizing dose for an object in an 813 km, sun-synchronous orbit, over the span of two years, while shielded by an aluminum spherical shell of a given thickness. Figure 3-7 plots the same data in graphical form. The data contain no margin or uncertainty factors.

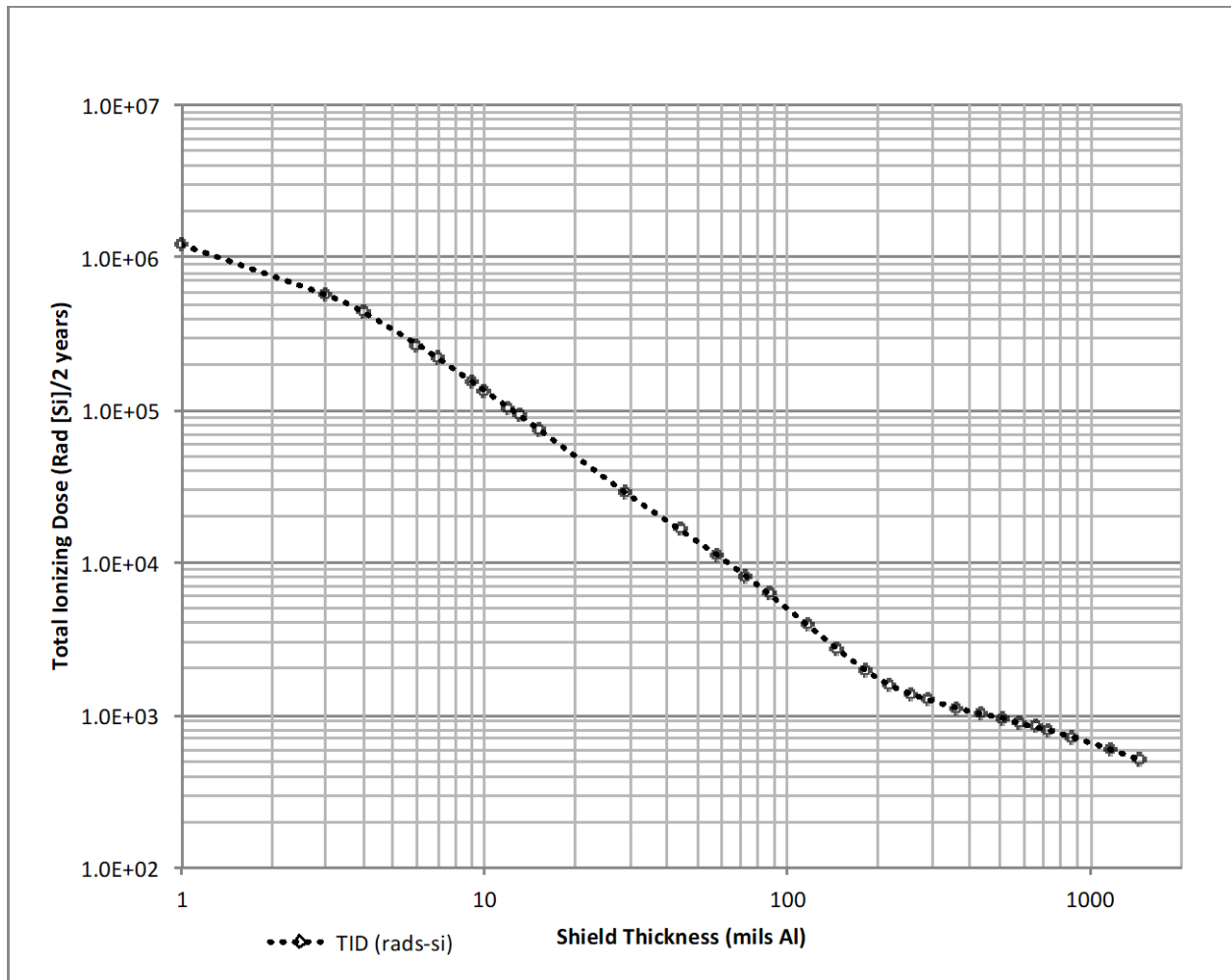


Figure 3-7: [LEO] TID versus Shielding Thickness

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**Table 3-10: [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

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 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

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NASA’s Radiation Belt Models, AE-8 and AP-8, and JPL’s Solar Proton Fluence Model. 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.

### 3.10.5.6 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 3-11 and Figure 3-8 provide a conservative micrometeoroid flux environment for LEO.

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 domains. The data come from the Grün flux model assuming a meteoroid mean velocity of 20 km/s and a constant average particle density of 2.5 g/cm<sup>3</sup>. 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.

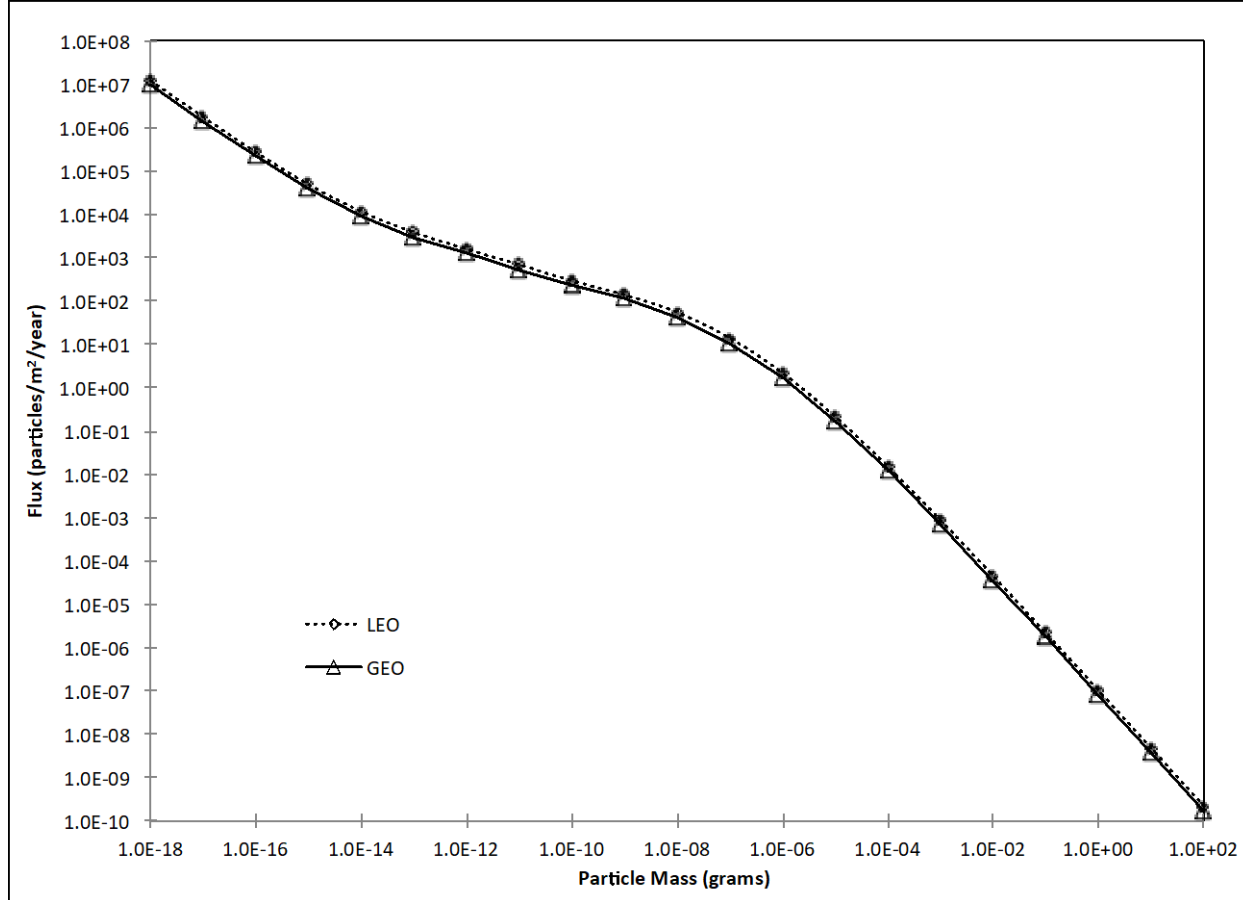
Micrometeoroid and artificial space debris flux guidelines are separate due to the stability of micrometeoroid flux over time, compared to the increase of artificial space debris.

**Table 3-11: Worst-case Micrometeoroid Environment**

Particle mass [g]	Particle diameter [cm]	Flux (particles/m <sup>2</sup> /year] LEO
1.00E-18	9.14E-07	1.20E+07
1.00E-17	1.97E-06	1.75E+06
1.00E-16	4.24E-06	2.71E+05
1.00E-15	9.14E-06	4.87E+04
1.00E-14	1.97E-05	1.15E+04
1.00E-13	4.24E-05	3.80E+03
1.00E-12	9.14E-05	1.58E+03
1.00E-11	1.97E-04	6.83E+02
1.00E-10	4.24E-04	2.92E+02
1.00E-09	9.14E-04	1.38E+02
1.00E-08	1.97E-03	5.41E+01
1.00E-07	4.24E-03	1.38E+01
1.00E-06	9.14E-03	2.16E+00
1.00E-05	1.97E-02	2.12E-01
1.00E-04	4.24E-02	1.50E-02
1.00E-03	9.14E-02	8.65E-04
1.00E-02	1.97E-01	4.45E-05



1.00E-01	4.24E-01	2.16E-06
1.00E+00	9.14E-01	1.02E-07
1.00E+01	1.97E+00	4.72E-09
1.00E+02	4.24E+00	2.17E-10



**Figure 3-8: Worst-case Micrometeoroid Environment**

### 3.10.5.7 Artificial Space Debris

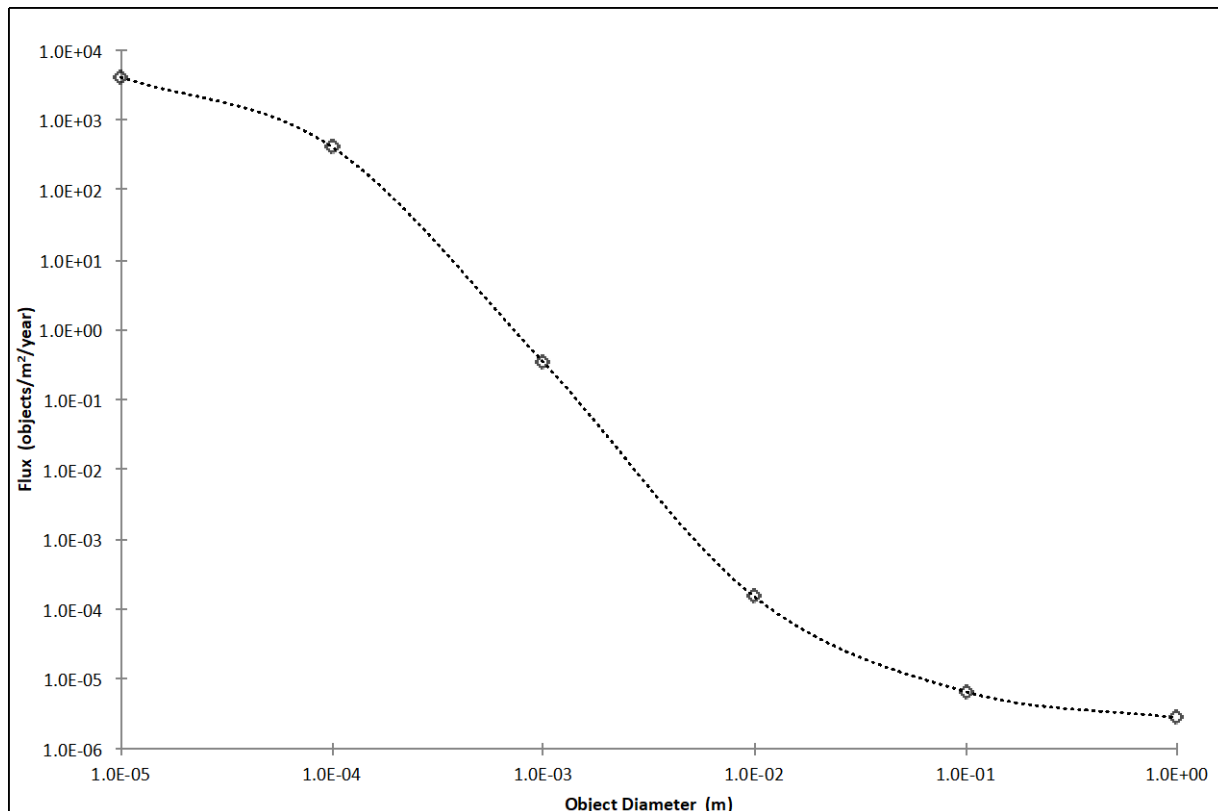
The Instrument Developer should perform a probability analysis to determine the type and amount of shielding to mitigate the fluence of artificial space debris in the expected mission orbit over the primary mission.

Table 3-12, and Figure 3-9, provide conservative artificial space debris flux environments for LEO.

**Table 3-12: [LEO] Worst-case Artificial Space Debris Environment**

Object [m]	Size	Flux [objects/m <sup>2</sup> /year]	Object [km/s]	Velocity
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



**Figure 3-9: [LEO]: 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 LEO 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.

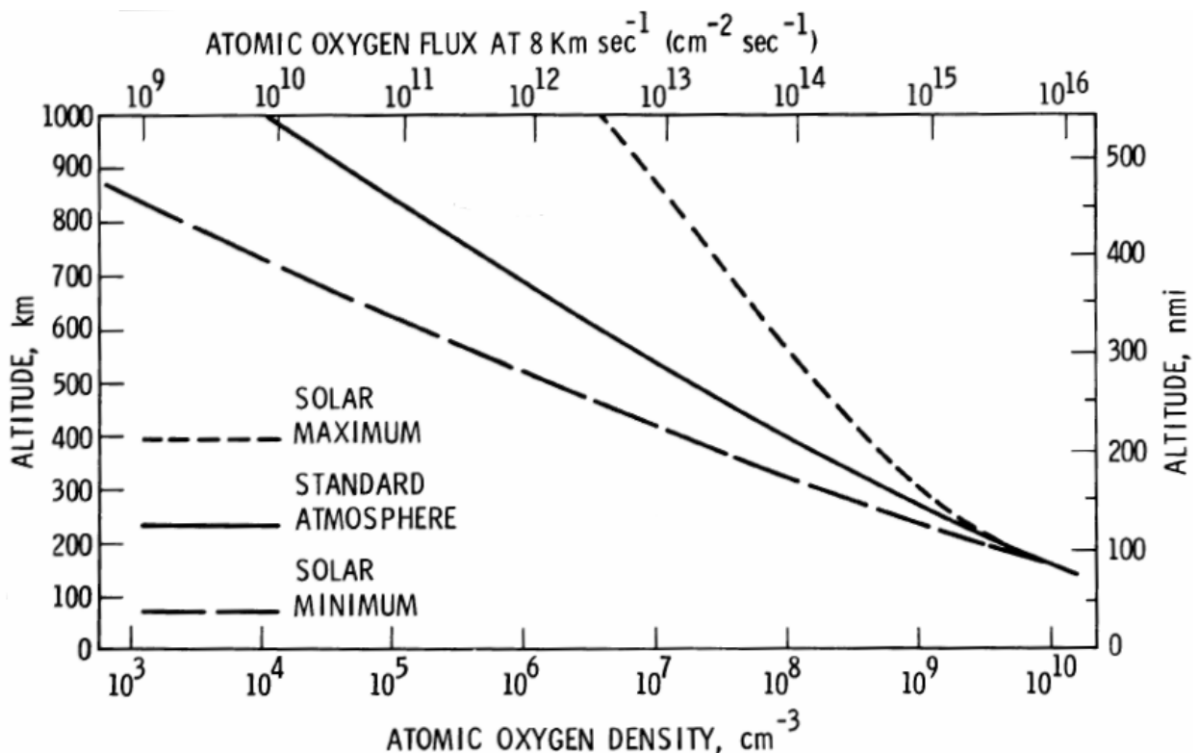
Micrometeoroid and artificial space debris flux guidelines are listed separately due to the stability of micrometeoroid flux over time, compared to the increase of artificial space debris. The premier and overriding guidance is that the Instrument will “do no harm” to the Host Spacecraft or other payloads. This implies that the Instrument will not generate orbital debris.

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### 3.10.5.8 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 estimated using the Figure 3-10, 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. Instrument Developers should conservatively estimate the atomic oxygen environment for their Instrument's specified orbit, orbital lifetime and launch date relative to the solar cycle. One source for predictive models is the Community Coordinated Modeling Center (CCMC) at <http://ccmc.gsfc.nasa.gov/index.php>



**Figure 3-10: Atmospheric Atomic Oxygen density in Low Earth Orbit (Figure 2 from de Rooij 2000)**

### 3.10.6 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.**

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

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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.

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

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## 4 DESIGN GUIDELINES FOR GEO

### 4.1 Assumptions

The HPIG guidelines assume the following regarding the Host Spacecraft:

- 1) Hosted Payload: The Host Spacecraft will have a primary mission different from that of the Instrument.
- 2) Nominal Orbit: The Host Spacecraft will operate in GEO with an altitude of approximately 35786 kilometers and eccentricity and inclination of approximately zero.
- 3) Responsibility for Integration: The Host Spacecraft Manufacturer will integrate the Instrument onto the Host Spacecraft with support from the Instrument Developer.

### 4.2 Mission Risk

**The Instrument should comply with Mission Risk class, as required by the customer such as the AO.**

**Mission risk classes are a measure of the instrument’s mission design and reliability. Regardless of the instrument’s mission risk class, the instrument must conform to the host “do no harm” requirements.**

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.”<sup>1</sup> An instrument designed as a 1-yr demonstration mission (Class C/D) must satisfy the reliability and “do no harm” requirements associated with a 15-yr GEO spacecraft (Class A/B). Future AOs will specify the risk class of the instrument.

### 4.3 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 the Instrument’s mission is “Complete.”

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<sup>1</sup> “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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#### 4.4 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.

#### 4.5 Data Guidelines

##### 4.5.1 Assumptions

The HPIG data assume the following regarding the Host Spacecraft:

- 1) During the pairing 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.

##### 4.5.2 Data Interface

###### 4.5.2.1 *Command and telemetry*

**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.

###### 4.5.2.2 *Science*

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

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.

##### 4.5.3 Data Accommodation

###### 4.5.3.1 *Command and telemetry*

**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.

###### 4.5.3.2 *Science*

**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).

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#### 4.5.4 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 Host Spacecraft. This is consistent with the direct-to-transponder science data interface. A spacecraft needs only enough buffer capacity to relay Instrument telemetry. Fewer resource impacts on the spacecraft maximize Instrument hosting opportunities.

#### 4.5.5 Command and Telemetry Dictionary

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

Rationale: 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.*).

Rationale: 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.

#### 4.5.6 SAFE mode

**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 Host Spacecraft resources (*e.g.* power).

Note: Please see Appendix D for a discussion of the notional instrument mode scheme referenced in this document.

##### 4.5.6.1 *Command (SAFE mode)*

**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.

#### 4.5.7 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 Host Spacecraft Manufacturer, Satellite Operator, and ground operations team to devise and operate Fault Detection Isolation, and Recovery (FDIR) procedures, crucial for on-orbit operations.

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#### 4.5.8 Command (Acknowledgement)

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

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

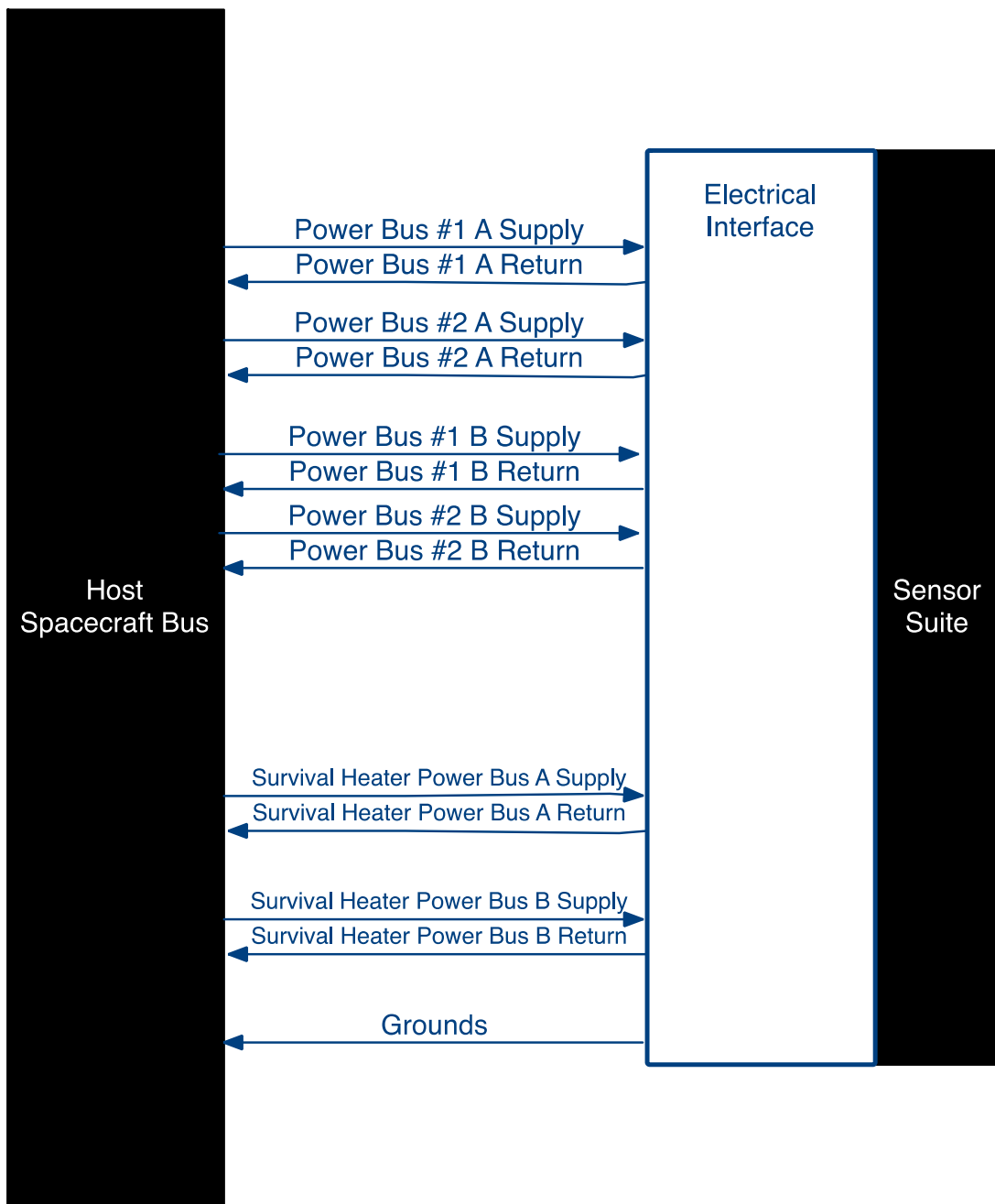
## 4.6 Electrical Power System

### 4.6.1 Assumptions

The HPIG electrical power assume the following regarding the Host Spacecraft:

- 1) During the pairing 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) The Host Spacecraft will supply to the Instrument EPS regulated electrical power within the range of  $28 \pm 3$  VDC, including ripple and normal transients as defined below, and power distribution losses due to switching, fusing, harness and connectors.
- 3) 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 4-1 shows a pictorial representation of this architecture.





**Figure 4-1: Host Spacecraft-Instrument Electrical Interface (Depicted with the optional Instrument side redundant Power Bus B interface)**

- 4) The Host Spacecraft will energize the Survival Heater Power at approximately 30% (or possibly higher, as negotiated with the host provider) of the AP [GEO] in accordance with the mission timeline documented in the EICD.

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- 5) The Host Spacecraft Manufacturer will supply a definition of the maximum source impedance by frequency band. Table 4-1 provides an example of this definition.

**Table 4-1: Example of Power Source Impedance Function**

Frequency	Maximum Source Impedance [ $\Omega$ ]
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

- 6) The Host Spacecraft Manufacturer will furnish all Host Spacecraft and Host Spacecraft-to-Instrument harnessing.
- 7) 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.
- 8) The Host Spacecraft will protect its own electrical power system via overcurrent protection devices on its side of the interface.
- 9) 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.
- 10) 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. The Host Spacecraft will continue to provide Survival Heater Power, but may remove Survival Heater Power if conditions deteriorate significantly.
- 11) 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.

#### 4.6.2 Grounding

**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.

##### 4.6.2.1 *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.

#### 4.6.3 Electrical Power Return

**The Instrument electrical power return should be via dedicated conductor(s).**

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Rationale: The Instrument Electrical Power System (EPS) should return electrical power via electrical harness or ground strap to reduce the potential to introduce stray currents or ground loop currents into the Instrument, Host Spacecraft, or other payloads.

#### 4.6.4 Accommodation

**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'<sup>2</sup> buses can accommodate, and requiring a power level less than this increases the likelihood of finding a suitable Host Spacecraft.

#### 4.6.5 Voltage

**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).

#### 4.6.6 Resistance power

**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 +6/-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.

#### 4.6.7 Power Bus Interface

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

- 12) 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.

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<sup>2</sup> 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 publicly-announced satellite operator contracts.

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13) 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.

#### 4.6.8 Survival Heater Bus Interface

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

14) Rationale: The Survival Heaters, which are elements of the Thermal subsystem, require power to heat certain instrument components during off-nominal scenarios when the Power Buses are not fully energized. See separate HPIG Best Practices document for more discussion about survival heaters.

#### 4.6.9 Bonding

**The Instrument bonding should comply with NASA-STD-4003 or equivalent.**

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

#### 4.6.10 Mitigation of In-Space Charging Effects

**The Instrument should comply with NASA-HDBK-4002A or equivalent 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 Host Spacecraft components and external surfaces related to space plasmas and high-energy electrons and the consequences of that charge buildup.

#### 4.6.11 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.

##### *4.6.11.1 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.

#### 4.6.12 EPS Accommodation

This section specifies the characteristics, connections, and control of the Host 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.

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#### 4.6.12.1 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.

#### 4.6.12.2 Instrument Power Harness

**Instrument power harnesses should be sized to the largest possible current value as specified by 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.

#### 4.6.12.3 Allocation of Instrument Power

**The EPS should draw no more power from the Host Spacecraft in each Instrument mode than defined in Table 4-2.**

Rationale: The 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 E.

**Table 4-2: Instrument Power Allocation**

Mode	GEO Average (W)
Off/ Survival	0/90
Activation	300
Safe	150
Operation	300

#### 4.6.12.4 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.

#### 4.6.12.5 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).**

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

#### 4.6.12.6 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. Ideally, the Instrument should power up in the minimum power draw state of the OFF/SURVIVAL Mode and then transition into the minimum power draw state of the INITIALIZATION Mode. The +28 VDC is inclusive of nominal voltage transients of  $\pm 3$  VDC for GEO Instruments.

#### 4.6.12.7 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.

## 4.7 Mechanical Interface

### 4.7.1 Assumptions

The HPIG mechanical interfaces assume the following regarding the Host Spacecraft:

- 1) 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. (It should be noted that FOV requests are best accommodated during the initial configuration of the host. Therefore, FOV may be a limiting factor in determining which host spacecraft is a viable candidate for your payload.)
- 3) The Host Spacecraft Manufacturer will furnish all instrument mounting fasteners.
- 4) The Host Spacecraft Manufacturer will provide a glint analysis that demonstrates that no reflected light impinges onto the Instrument FOV, if requested by the Instrument Developer.
- 5) The Host Spacecraft Manufacturer will furnish the combined structural dynamics analysis results to the Instrument Developer.

### 4.7.2 Mechanical Interface

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

Rationale: Assessments of the responses to the *CII RFI for GEO Hosted Payload Opportunities* indicate nadir-deck mounting of hosted payloads can be accommodated. Alternative mechanical

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interface locations or kinematic mounts are not prohibited by this guidance but may increase interface complexity.

#### 4.7.3 Accommodation

**The Instrument mass should be less than or equal to 150 kg.**

##### 4.7.3.1 *Mass*

Rationale: Based on broad survey of industry hosted payload accommodations, instrument mass of approximately 50kg and below would **provide maximum** opportunity for finding a host. Opportunities above 150kg may exist but **provide only a minimum** hosting probability.

##### 4.7.3.2 *Volume*

**The Instrument and all of its components should remain within a volume of 1 m<sup>3</sup> during all phases of flight.**

Rationale: Based on broad survey of industry hosted payload accommodations, instrument volume of approximately 0.1 m<sup>3</sup> and below would **provide maximum** opportunity for finding a host. Opportunities above 1m<sup>3</sup> may exist but **provide only a minimum** hosting probability.

#### 4.7.4 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.

#### 4.7.5 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.

#### 4.7.6 Moveable Masses

**The Instrument should compensate for the momentum associated with the repetitive movement of large masses, relative to the mass of the Host Spacecraft.**

Rationale: This prevents moveable masses from disturbing the operation of the Host Spacecraft or other payloads. This will generally not apply to items deploying during startup/initiation of operations, and the applicability of the guideline will be negotiated with the Host Spacecraft Manufacturer and/or Satellite Operator during pairing.

#### 4.7.7 Minimum Fixed-Base Frequency

**The Instrument should have a fixed-base frequency greater than 100 Hz.**

Rationale: Based on broad survey of industry hosted payload accommodations, this minimum fixed-based frequency meets the composite guidance of a majority of the responding (GEO) Host

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Spacecraft manufacturers. Opportunities below 100 Hz may exist but decrease hosting possibilities. To some extent, the Instrument will affect the Host Spacecraft frequency depending on the payload's mass and mounting location. Host Spacecraft Manufacturers may negotiate for a greater fixed-based frequency for hosted payloads until the maturity of the instrument can support Coupled Loads Analysis.

## 4.8 Thermal Guidelines

### 4.8.1 Assumptions

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

- 1) During the pairing 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 its own side of the interface from the Integration through Disposal portions of its lifecycle.
- 3) The Host Spacecraft Manufacturer will be responsible for thermal hardware used to close out the interfaces between the Instrument and Host Spacecraft, such as closeout Multi-layer Insulation (MLI).

### 4.8.2 Thermal Interface

**The Instrument should be thermally isolated from the Host Spacecraft.**

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.

### 4.8.3 Thermal Design at the Mechanical Interface

**The Instrument thermal design should be decoupled from the Host Spacecraft at the mechanical interface between the spacecraft and neighboring payloads to the maximum practical extent.**

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.

### 4.8.4 Conductive Heat Transfer

**The conductive heat transfer at the Instrument-Host Spacecraft mechanical interface should be less than 15 W/m<sup>2</sup> or 4 W, whichever is greater.**

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

### 4.8.5 Radiative Heat Transfer

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



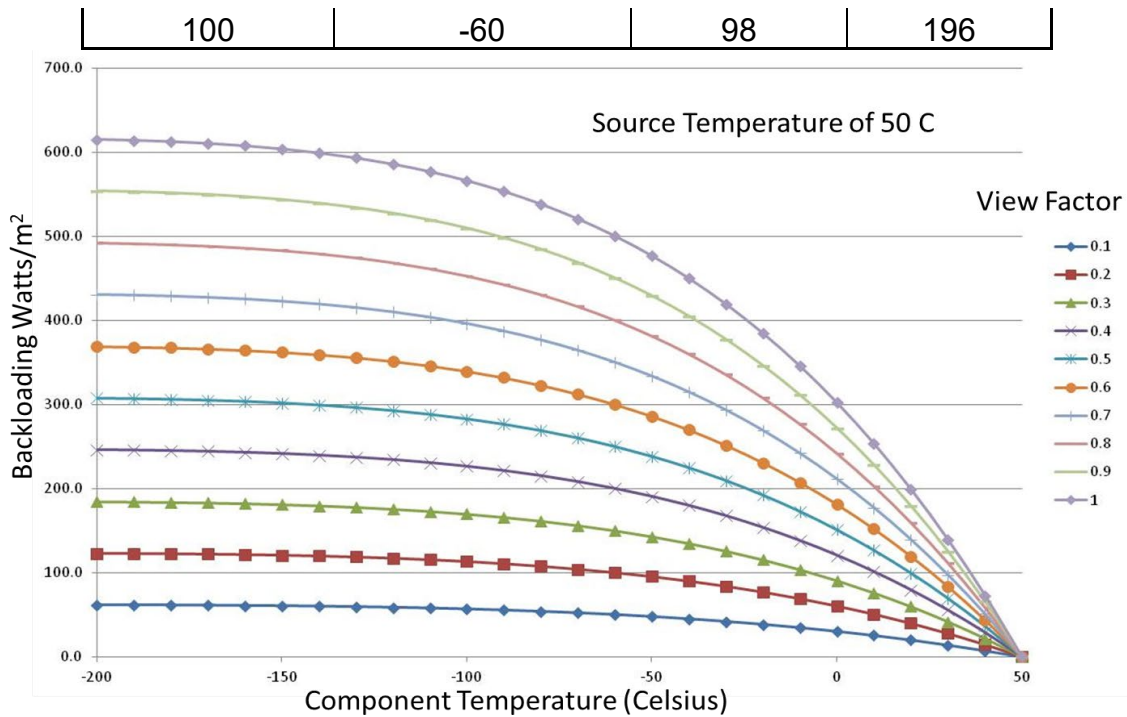
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**Rationale:**

- 1) 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.
- 2) Hosted payload with science instruments requiring radiators operating at cold temperatures (below 25 °C) should consider the backloading and field of view blockage from the warm parts of the spacecraft on the radiators. Although this configuration details will be unknown prior to Instrument-to-Host pairing, the Hosted Payload should assess its sensitivity to some degree of blockage. For GEO spacecraft, radiators are likely to be co-located on North or South facing panels where solar arrays are co-located. Solar array and antennas can impose significant backloading if the radiator has any view of them (see Table 4-3 and Figure 4-2).

**Table 4-3: Worst-case Backloading on Payload Radiator**

S/C Source Temp., °C	Payload Radiator Temp., °C	Load, W/m2 VF=0.1	Load, W/m2 VF=0.2
50	50	0	0
50	40	7	15
50	30	14	28
50	20	20	40
50	10	25	51
50	0	30	60
50	-10	35	69
50	-20	38	77
50	-30	42	84
50	-40	45	90
50	-50	48	95
50	-60	50	100
100	50	48	96
100	40	55	110
100	30	62	124
100	20	68	136
100	10	73	147
100	0	78	156
100	-10	82	165
100	-20	86	173
100	-30	90	180
100	-40	93	186
100	-50	96	191



**Figure 4-2: Worst-case Backloading on Payload Radiator**

4.8.6 Temperature Maintenance Responsibility

**The Instrument should maintain its own instrument temperature requirements.**

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

4.8.7 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.

4.8.8 Thermal Control Hardware Responsibility

**The Instrument Provider should provide and install all Instrument thermal control hardware including blankets (except MLI closeout blankets to be provided by the Host Spacecraft), 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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## 4.9 Attitude Control

### 4.9.1 Attitude Control System Pointing Accommodation

**The Instrument  $3\sigma$  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.

### 4.9.2 Attitude Determination System Pointing Knowledge Accommodation

**The Instrument  $3\sigma$  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.

### 4.9.3 Payload Pointing Stability Accommodation

**The Instrument short term ( $\geq 0.1$  Hz)  $3\sigma$  pointing stability required should be greater than or equal to 110 seconds of arc/second (0.03 degrees/second) in each Host Spacecraft axis.**

**The Instrument long term (Diurnal)  $3\sigma$  pointing stability required should be greater than or equal to 440 seconds of arc (0.12 degrees/second) in each Host Spacecraft axis.**

Rationale: Host Spacecraft pointing stability varies significantly both by manufacturer and by bus configuration. In order to maximize the probability of pairing with an available HPO, 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 ( $\geq 0.1$  Hz) pointing stability available for secondary hosted payloads is greater than or equal to 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 greater than or equal to 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.

## 4.10 Instrument Models

**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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## 4.11 Environmental Guidelines

### 4.11.1 Assumptions

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

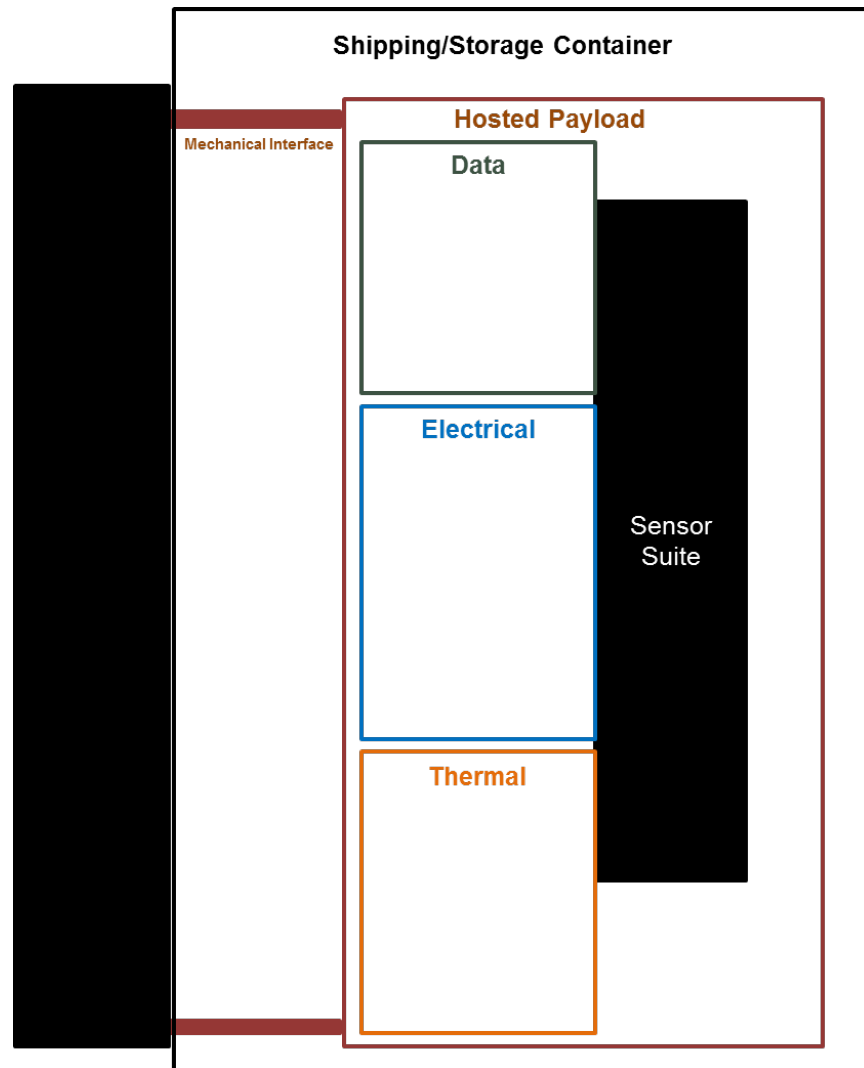
- 1) During the pairing process, the Host Spacecraft Manufacturer/Systems Integrator and the Instrument Developer will negotiate detailed parameters of the environmental interface. The Environmental Requirements Document (ERD) will record those parameters and decisions.

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 E.

### 4.11.2 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 Host Spacecraft Manufacturer/Systems Integrator. The Instrument is dormant and attached mechanically to its container (see Figure 4-3).

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**Figure 4-3: Shipping / Storage Environment**

*4.11.2.1 Documentation*

**The ERD will document the maximum allowable 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.

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#### *4.11.2.2 Instrument Configuration*

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

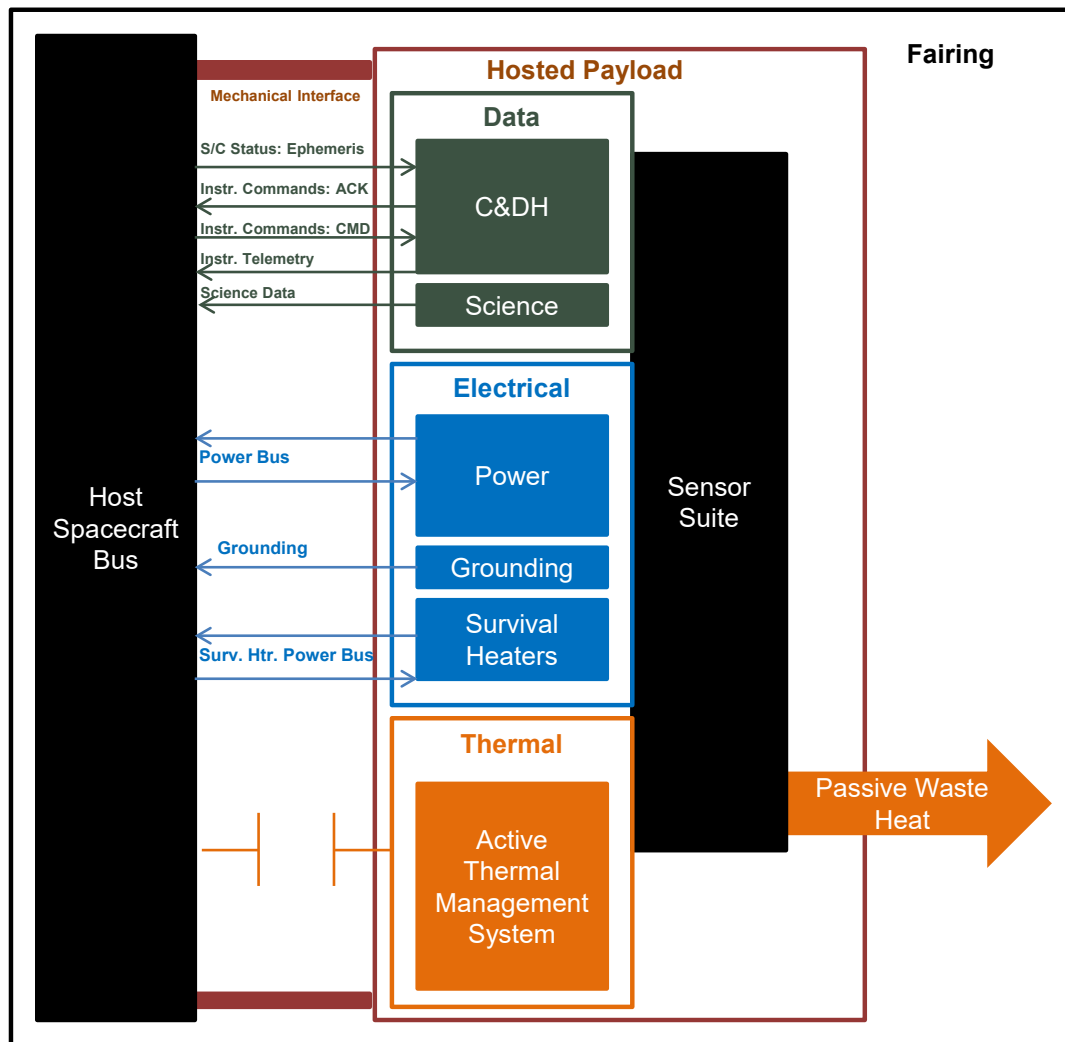
Rationale: Specifying the configuration of the Instrument during shipping/storage drives the volume requirements for the container as well as any associated support equipment and required services.

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

#### 4.11.3 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 Host 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 spacecraft bus and verify that system performs as designed throughout various environmental and dynamics regimes. The Instrument may be attached to the spacecraft bus or to various ground support equipment that transmits power, thermal conditioning, and diagnostic data (see Figure 4-4).

The instrument should be designed to minimize integrated tests with the spacecraft during the system level I&T phase. This is especially important during test activities in the environmental chambers. To the extent practical for the instrument, all performance testing should be performed prior to arrival at the spacecraft facility. Interface compatibility should be tested and the instrument should be powered down for the majority of spacecraft system level activities. This approach is to minimize schedule, cost, and complexity with the host.



**Figure 4-4: Integration and Test Environment**

#### 4.11.3.1 Documentation

**The ERD will document the maximum allowable 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 ERD may either record these data explicitly or refer to a negotiated Test and Evaluation Master Plan (TEMP).

#### 4.11.3.2 Instrument Configuration

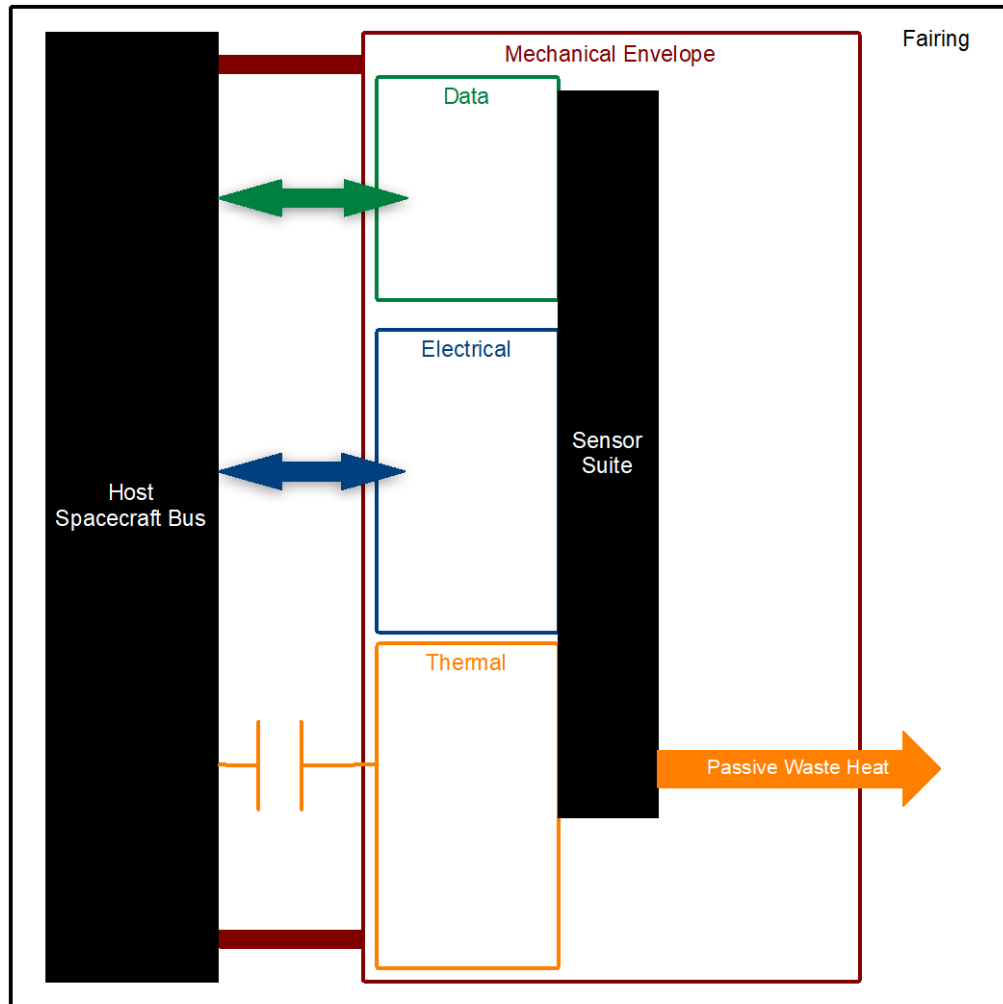
**The ERD will document the configuration and operational mode of the Instrument during the Integration and Test phase.**

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

#### 4.11.4 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 4-5).



**Figure 4-5: Launch Environment**

##### 4.11.4.1 Documentation

**The ERD will document the maximum allowable 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.



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#### 4.11.4.2 Instrument Configuration

**The ERD 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.

#### 4.11.4.3 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, damage, 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 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)*.

#### 4.11.4.4 Quasi-Static Acceleration

**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 4-4.**

**Table 4-4: Mass Acceleration Curve Design Load Limits**

Mass [kg]	Limit Load [g](any direction)
1	68.0
5	49.0
10	39.8
20	31.2
40	23.8
60	20.2
80	17.8
100	16.2
125	14.7
150	13.5
175	12.6
200 or Greater	12.0

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft or other payloads. This guidance represents the

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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, and the loads shown in Table 4-4 should be updated based on a launch vehicle specific set of MAC loads or the results of coupled loads analysis when this information becomes available.

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.

#### 4.11.4.5 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 4-5.**

**Table 4-5: Sinusoidal Vibration Environment**

Frequency (Hz)	Amplitude	
	Flight Level	Qual/Protoflight
5 – 20	12.7 mm (double amplitude)	16 mm (double amplitude)
20 – 100	10.0 g	12.5 g
Qual/Protoflight Sweep Rate: From 5 to 100 Hz at 4 octaves/minute except from 40 to 55 Hz at 6 Hz/min		
Flight Level Sweep Rate: From 5 to 100 Hz at 4 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, damage, or inducing degraded performance of or damage to the Host Spacecraft or other payloads. Table 4-5 provides a generic sine environment for the preliminary design of components and subsystems. The sine sweep vibration levels shown in Table 4-5 are defined at the hardware mounting interface. This guidance represents the need to be compatible with the coupled dynamics loads that will be experienced during ground processing and launch ascent.

#### 4.11.4.6 Random Vibration

[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 4-6.

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.

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**Table 4-6: Random Vibration Environment (derived from GEVS-SE, Table 2.4-4)**

Zone/Assembly	Frequency (Hz)	Protoflight / Qualification	Acceptance
Instrument	20	0.026 g <sup>2</sup> /Hz	0.013 g <sup>2</sup> /Hz
	20 – 50	+6 dB/octave	+6 dB/octave
	50 - 800	0.16 g <sup>2</sup> /Hz	0.08 g <sup>2</sup> /Hz
	800 - 2000	-6 dB/octave	-6 dB/octave
	2000	0.026 g <sup>2</sup> /Hz	0.013 g <sup>2</sup> /Hz
	Overall	14.1 g <sub>rms</sub>	10.0 g <sub>rms</sub>

Table 4-6 represents the random vibration environment for instruments with mass less than or equal to 25 kg and having resonant frequencies greater than 80 Hz. 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  $\pm 6$  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<sup>2</sup>/Hz at 20 Hz and at 2000 Hz for qualification testing and an ASD of 0.005 g<sup>2</sup>/Hz at 20 Hz and at 2000 Hz for acceptance testing.
- 3) The random vibration levels given in Table 3-6 should be updated based on test data or acoustic analysis of the payload once the launch vehicle specific acoustic environment has been defined
- 4) Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft 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. Based upon CII analysis of the following sources of performance data: the *The CII Guidelines Document, Revision A, GEVS-SE* and the *USAF HoPS studies data*, an overall protoflight/qual level of 23.1 g (rms) would maximize hosting opportunity.

#### 4.11.4.7 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 4-7.**

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**Table 4-7: Acoustic Noise Environment**

<b>1/3 Octave Band Center Frequency (Hz)"</b>	<b>Design/Qual/Protoflight (dB w/ 20 µPa reference)"</b>	<b>Acceptance (dB w/ 20 µPa reference)"</b>
20	129.5	126.5
25	130.7	127.7
31.5	130.0	127.0
40	131.5	128.5
50	133.0	130.0
63	134.5	131.5
80	135.5	132.5
100	136.0	133.0
125	136.8	133.8
160	136.7	133.7
200	136.0	133.0
250	136.0	133.0
315	136.0	133.0
400	134.0	131.0
500	132.0	129.0
630	131.4	128.4
800	131.6	128.6
1000	129.9	126.9
1250	126.1	123.1
1600	121.3	118.3
2000	119.5	116.5
2500	118.0	115.0
3150	116.1	113.1
4000	115.5	112.5
5000	114.8	111.8
6300	114.0	111.0
8000	113.0	110.0
10000	112.1	109.1

Rationale: Acoustic design guidelines are an envelope of a number of the common launch vehicles. This acoustic environment should be used for preliminary design of components and subsystems if a specific launch vehicle has not been defined. While all hardware should be assessed for sensitivity to direct acoustic impingement, unless the component or subsystem has structure which is light-weight and has large surface area (typically a surface to weight ratio of > 150 in<sup>2</sup>/lb), it is expected that the random environment specified in Section 4.11.4.6 will be the dominant high-frequency loading condition rather than the acoustic environment defined in Table 4-7.

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.

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#### 4.11.4.8 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 4-8.**

**Table 4-8: Shock Response Spectrum (Q=10)**

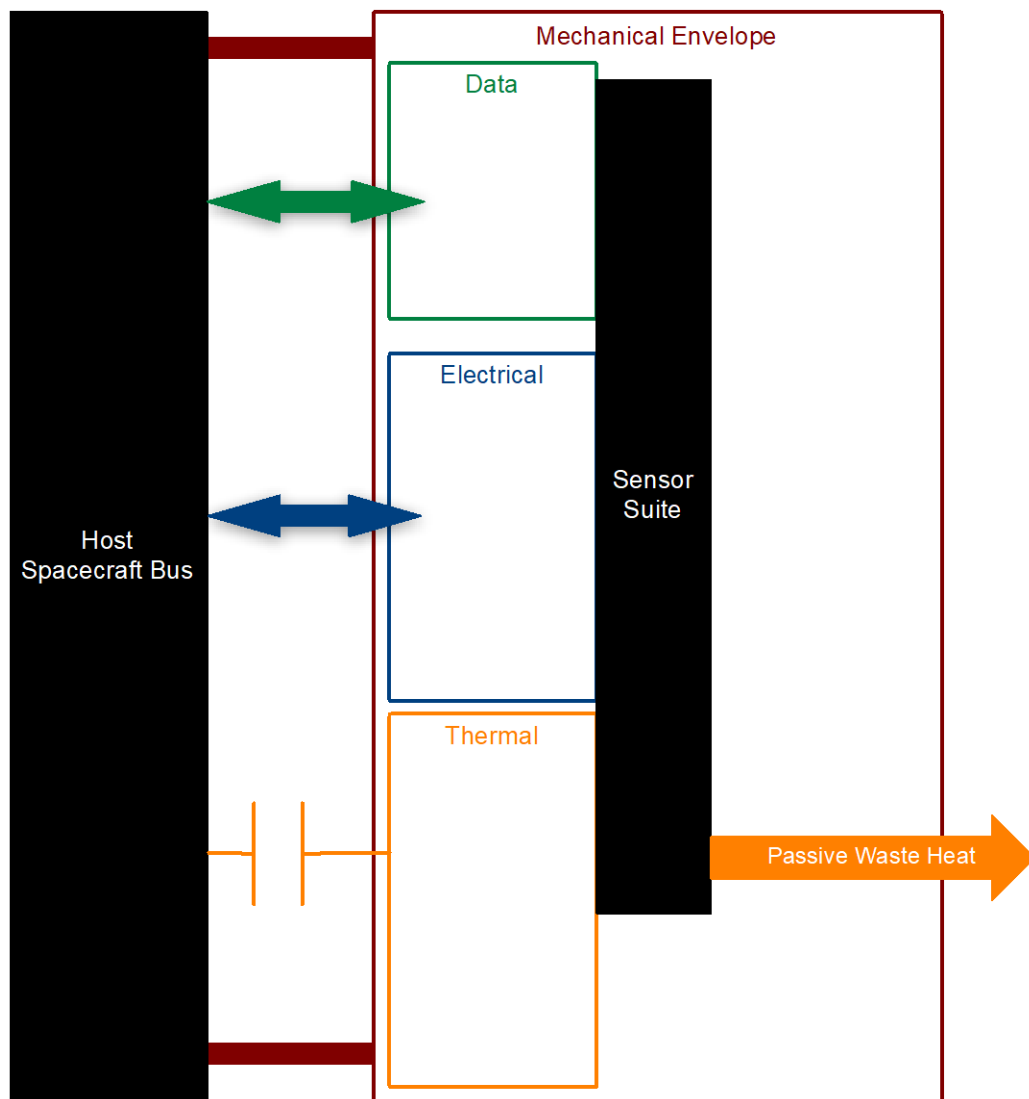
Frequency (Hz)	Acceptance Level (g)	Protoflight/Qualification (g)
100	160	224
630	1000	1400
10000	1000	1400
	The shock levels given assume that a component is located at least 60 cm (2 ft) from a shock source	The shock levels given assume that a component is located at least 60 cm (2 ft) from a shock source

Rationale: The Instrument must be able to withstand conditions typical of the AI&T, launch and on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft 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. Table 4-8 provides a generic shock environment that may be used for hardware design until the mission specific shock environments can be defined. Based on broad survey of industry hosted payload accommodations, designing for higher shock levels (up to 5000 g for 1600 Hz and 10000 Hz) would maximize opportunity for finding a host. After pairing, the levels shown in Table 4-8 should be updated once all payload shock sources have been defined.

#### 4.11.5 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 4-6).

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**Figure 4-6: Operational Environment**

The GEO guidelines are based upon a zero degree inclination, 35786 km altitude circular orbit.

*4.11.5.1 Orbital Acceleration*

**The Instrument should function according to its operational specifications after being subjected to a maximum spacecraft-induced acceleration of 0.15g.**

Rationale: The Instrument in its operational configuration 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 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.

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The GEO guidelines are based upon a zero degree inclination, 35786 km altitude circular orbit.

#### 4.11.5.2 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  $1.3 \times 10^{-15}$  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, damage, or inducing degraded performance of or damage to the Host Spacecraft 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*.

#### 4.11.5.3 Thermal Environment

**The Instrument should function according to its operational specifications after being subjected to a thermal environment characterized by Table 4-9.**

**Table 4-9: Thermal Radiation Environment**

Domain	Solar Flux [W/m <sup>2</sup> ]	Earth IR (Long Wave) [W/m <sup>2</sup> ]	Earth Albedo
GEO	1290 to 1420	5.5	2.5-7.2W/sq.m

Rationale: The Instrument must be able to withstand conditions typical of the on-orbit environment without suffering degraded performance, damage, or inducing degraded performance of or damage to the Host Spacecraft 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. The Host Spacecraft Manufacturer will document the expected Free Molecular Heating rate seen by the exposed surface of the payload during the launch ascent in the TICD. This guidance defines the solar flux over the entire spectrum. In the UV portion of the spectrum ( $\lambda \leq 300$  nm), the solar flux is approximately 118 W/m<sup>2</sup> and the integrated photon flux is approximately  $2.28 \times 10^{15}$  photons/cm/sec. Reference NASA TM4527 for additional detail regarding the UV spectrum and associated photon flux.

#### 4.11.6 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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#### 4.11.6.1 Total Ionizing Dose

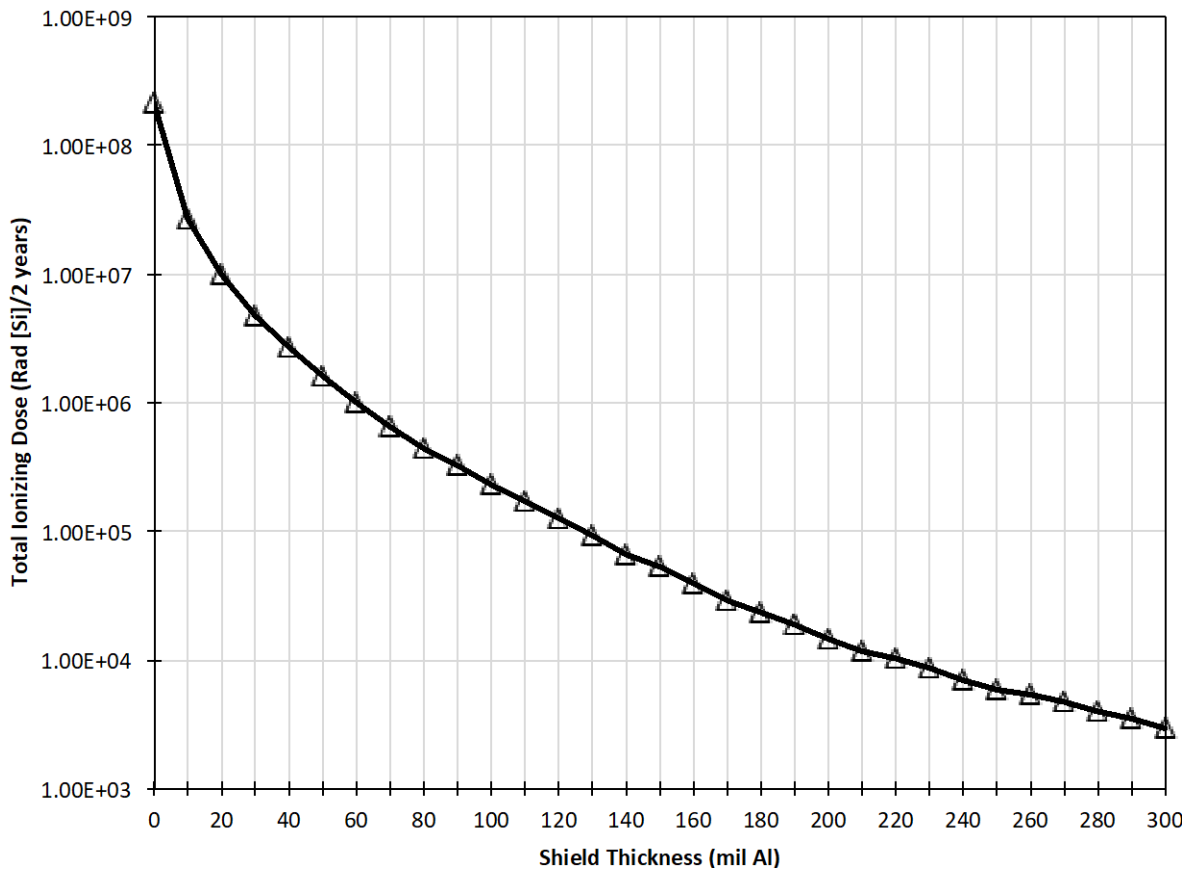
**The Instrument should function according to its operational specifications during and after exposure to the Total Ionizing Dose (TID) radiation environment based upon the specified mission orbit over the specified mission lifetime.**

Table 4-10 shows the expected total ionizing dose for an 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 4-7 plots the same data in graphical form. The data contain no margin or uncertainty factors.

**Table 4-10: [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





**Figure 4-7: 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 shielding. 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 in Section 3.10.5.5 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.

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Note that with the advent of all electric propulsion spacecraft, significant amounts of time may be spent during the spiral orbital transfer to GEO and this must be accounted for in the development of the TID and other orbit sensitive environments.

#### 4.11.6.2 Particle Fluxes

**The Instrument should function according to specification in the operational environment when exposed to the particle fluxes defined by Table 4-11.**

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 4-11: Particle fluxes in GEO w/ 100 mils of Aluminum Shielding**

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

#### 4.11.6.3 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 4-12 and Figure 4-8 provide a conservative micrometeoroid flux environment.

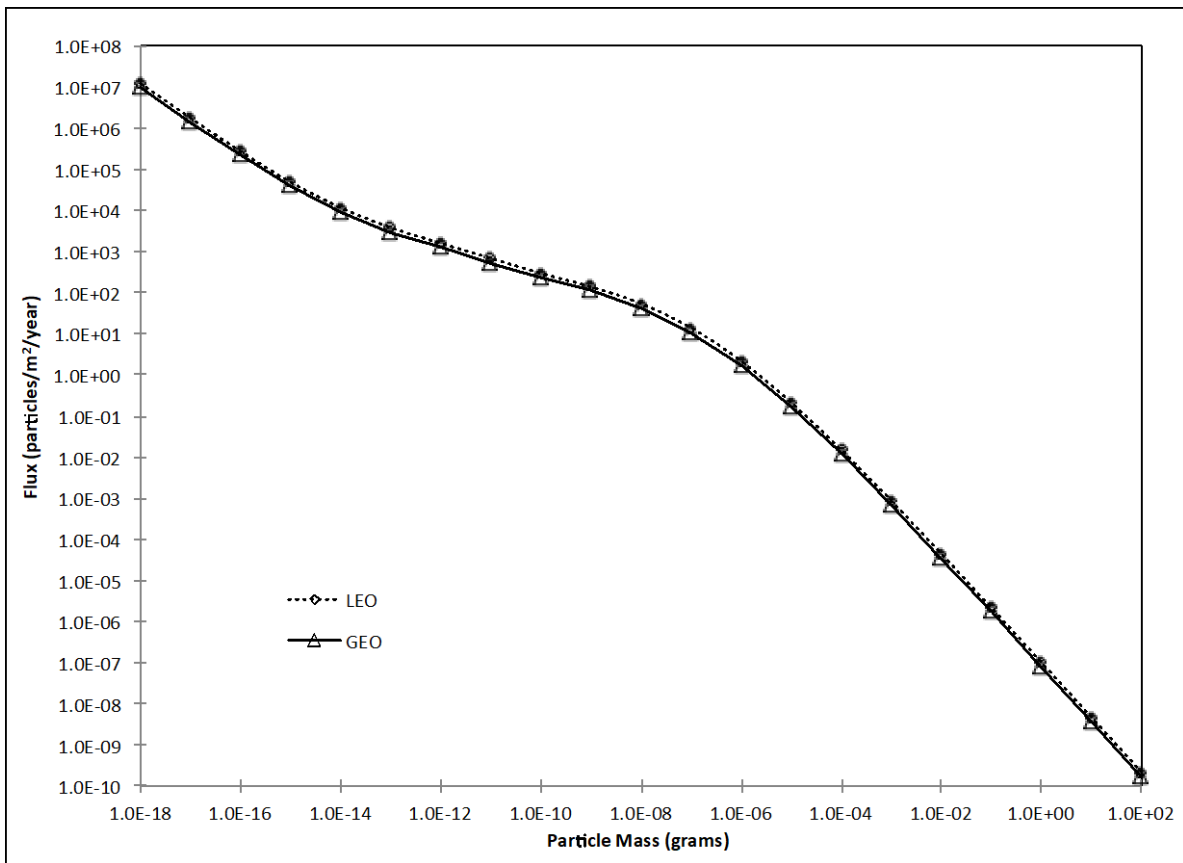
Rationale: Impacts from micrometeoroids may cause permanently degraded performance or damage to the hosted payload instrument. This guidance provides estimates of the worst-case scenarios of micrometeoroid particle size and associated flux over the GEO domains. The data come from the Grün flux model assuming a meteoroid mean velocity of 20 km/s and a constant average particle density of 2.5 g/cm<sup>3</sup>.

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 4-12: Worst-case Micrometeoroid Environment**

Particle mass [g]	Particle diameter [cm]	Flux (particles/m <sup>2</sup> /year)	
		LEO	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



**Figure 4-8: Worst-case Micrometeoroid Environment**

*4.11.6.4 Artificial Space Debris*

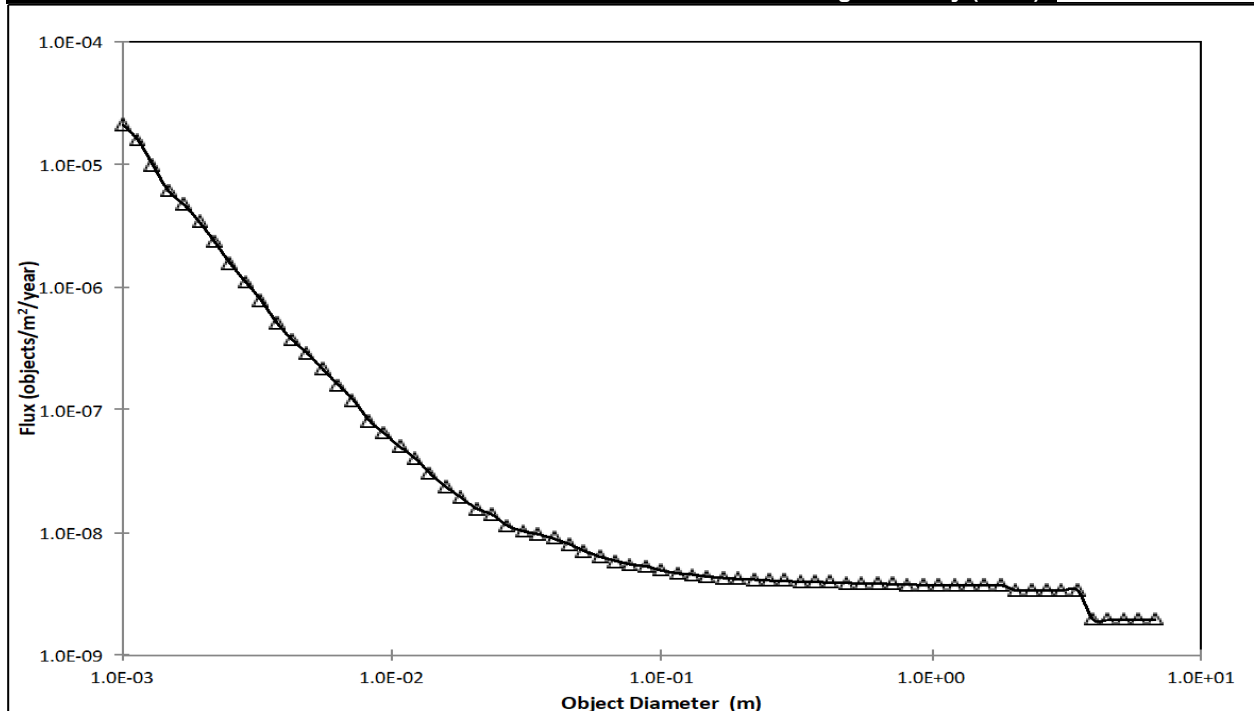
**The Instrument Developer should perform a probability analysis to determine the type and amount of shielding to mitigate the fluence of artificial space debris in the expected mission orbit over the primary mission.**

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Table 4-13 and Figure 4-9 provide conservative artificial space debris flux environments for GEO.

**Table 4-13: [GEO] Worst-case Artificial Space Debris Environment**

Object Diameter [m]	Flux [objects/m <sup>2</sup> /year]	Object Diameter [m]	Flux [objects/m <sup>2</sup> /year]	Object Diameter [m]	Flux [objects/m <sup>2</sup> /year]
1.0000E-03	2.0880E-05	2.0620E-02	1.5630E-08	4.25179E-01	3.9300E-09
1.1410E-03	1.5880E-05	2.3520E-02	1.4020E-08	4.84969E-01	3.8970E-09
1.3010E-03	9.7470E-06	2.6827E-02	1.1350E-08	5.53168E-01	3.8570E-09
1.4840E-03	6.0620E-06	3.0599E-02	1.0290E-08	6.30957E-01	3.8300E-09
1.6930E-03	4.7030E-06	3.4903E-02	9.7410E-09	7.19686E-01	3.8170E-09
1.9310E-03	3.3890E-06	3.9811E-02	8.9250E-09	8.20891E-01	3.7660E-09
2.2020E-03	2.3270E-06	4.5409E-02	8.0740E-09	9.36329E-01	3.7520E-09
2.5120E-03	1.5570E-06	5.1795E-02	7.0630E-09	1.06800E+00	3.7380E-09
2.8650E-03	1.1020E-06	5.9078E-02	6.3620E-09	1.21819E+00	3.7380E-09
3.2680E-03	7.8160E-07	6.7386E-02	5.8890E-09	1.38949E+00	3.7380E-09
3.7280E-03	5.1680E-07	7.6862E-02	5.5220E-09	1.58489E+00	3.7380E-09
4.2520E-03	3.7360E-07	8.7671E-02	5.3070E-09	1.80777E+00	3.7380E-09
4.8500E-03	2.8860E-07	1.0000E-01	4.9120E-09	2.06199E+00	3.3850E-09
5.5320E-03	2.1560E-07	1.14062E-01	4.6650E-09	2.35195E+00	3.3850E-09
6.3100E-03	1.6020E-07	1.30103E-01	4.5600E-09	2.68270E+00	3.3850E-09
7.1970E-03	1.2030E-07	1.48398E-01	4.3940E-09	3.05995E+00	3.3800E-09
8.2090E-03	8.2150E-08	1.69267E-01	4.2740E-09	3.49025E+00	3.3780E-09
9.3630E-03	6.4250E-08	1.93070E-01	4.1830E-09	3.98107E+00	1.9520E-09
1.0680E-02	5.0020E-08	2.20220E-01	4.1470E-09	4.54091E+00	1.9500E-09
1.2182E-02	4.0540E-08	2.51189E-01	4.0820E-09	5.17948E+00	1.9490E-09
1.3895E-02	3.0030E-08	2.86512E-01	4.0290E-09	5.90784E+00	1.9480E-09
1.5849E-02	2.3630E-08	3.26803E-01	3.9930E-09	6.73863E+00	1.9480E-09
1.8078E-02	1.9200E-08	3.72759E-01	3.9600E-09	7.68625E+00	1.3690E-13
<b>Average Velocity (km/s)</b>					<b>1.3333</b>



**Figure 4-9: [GEO] Worst-case Artificial Space Debris Environment**

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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 GEO domains during the Calendar Year 2015 epoch. Expected artificial space debris flux increases over time as more hardware is launched into orbit.

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. The premier and overriding guidance is that the Instrument will “do no harm” to the Host Spacecraft or other payloads. This implies that the Instrument will not generate orbital debris.

#### *4.11.6.5 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 GEO are negligible and are only significant for GEO-bound Instruments that spend extended times in LEO prior to GEO transfer. Instrument Developers should conservatively estimate the atomic oxygen environment for their Instrument’s specified orbit(s), orbital lifetime and launch date relative to the solar cycle. One source for predictive models is the Community Coordinated Modeling Center (CCMC) at <http://ccmc.gsfc.nasa.gov/index.php>

#### 4.11.7 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.**

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.

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

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## 5 ACRONYMS

AI&T	Assembly, Integration, and Test
AP	Average Power
ASD	Acceleration Spectral Density
AWG	American Wire Gauge
C&DH	Command and Data Handling
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
ERD	Environmental Requirements Document
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
HPIG	Hosted Payload Interface Guide
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



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LVDS	Low Voltage Differential Signaling
MAC	Mass Acceleration Curve
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
PPS	Pulse Per Second
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
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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## 7 UNITS OF MEASURE AND METRIC PREFIXES

**Table 7-1: Units of Measure**

<b>Abbreviation</b>	<b>Unit</b>
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 $\equiv$ 0.01 J/(kg of Silicon)
s	second
T	tesla
Torr	torr
V	volt
$\Omega$	ohm

**Table 7-2: Metric Prefixes**

<b>Prefix</b>	<b>Meaning</b>
M	mega ( $10^6$ )
k	kilo ( $10^3$ )
d	deci ( $10^{-1}$ )
c	centi ( $10^{-2}$ )
m	milli ( $10^{-3}$ )
$\mu$	micro ( $10^{-6}$ )
n	nano ( $10^{-9}$ )
p	pico ( $10^{-12}$ )

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## Appendix A Lessons Learned

### A.1 OVERVIEW

#### A.1.1 Introduction and Scope

When dealing with hosted payloads, it is unlikely that the hosting bus will be able to provide all the required interfaces or provide the performance necessary to satisfy the payload's mission requirements. As a result, each hosted payload will need to coordinate solutions with the spacecraft and primary payload teams to ensure mission success.

This appendix describes lessons learned from previous hosted payload studies and flight missions. Each section will provide an overview of the mission for context, and lessons learned related to the noted topics. The intent is to focus on the interface designs as called out in the HPIG, and not issues related to programmatic elements unless the issue caused a large cost or schedule delta.

The main body of the HPIG is meant to describe *guidelines* rather than prescriptive requirements, and thus deviation from those guidelines is to be expected. This Appendix, however, highlights where deviations were negotiated between bus and instrument developers, or instances where deviations were outright problematic.

### A.2 LESSONS LEARNED

#### A.2.1 NASA Tropospheric Emissions: Monitoring of Pollution (TEMPO)

The TEMPO instrument was competitively selected as part of NASA's Earth Venture Instrument program. TEMPO will be the first hosted payload to ride on a commercial satellite in GEO synchronous orbit with a scheduled delivery date of September 2017. Several commercial communication satellites are expected to be suitable to host the TEMPO instrument and selection of the host spacecraft contractor is pending.

TEMPO is a dispersive spectrometer that measures the pollution of North America hourly and at high spatial resolution. TEMPO spectroscopic measurements in the ultraviolet and visible wavelengths provide a tropospheric measurement suite that includes the key elements of tropospheric air pollution chemistry. Measurements are from geostationary orbit, to capture the inherent high variability in the diurnal cycle of emissions and chemistry. A small spatial footprint resolves pollution sources at a sub-urban scale. TEMPO quantifies and tracks the evolution of aerosol loading providing near-real-time air quality products that will be made available publicly to allow near-real time air quality management.

The TEMPO instrument is a NASA Designated Risk Class C payload (Med Priority, Med Risk, Less than 2 years Primary Mission Timeline).

### A.2.2 Summary of Lessons learned – Thermal Control / Thermal Interfaces

During the instrument design phase ensure that a system level thermal model is developed that includes a representative host spacecraft bus in order to verify instrument thermal performance.

The TEMPO project planned to utilize a heritage radiator design from a similar instrument to ensure the Focal Plane Subassembly (FPS) and structures were maintained at their optimal temperatures. The radiator rejects waste heat to space through a two-stage radiator. This radiator passively cools the FPA and uses trim heaters to control focal plane temperature. The other radiator stage rejects the remaining instrument waste heat. Maximum incident thermal backload on the instrument radiator was not expected to exceed 25 W/m<sup>2</sup> at any time during mission.

Thermal Desktop simulation of representative GEO Com Sats indicated that this class of spacecraft is not able to satisfy the 25 W/m<sup>2</sup> backload requirement due to excessive radiation from the bus solar arrays as show in Figure A-1 and Table A-1.

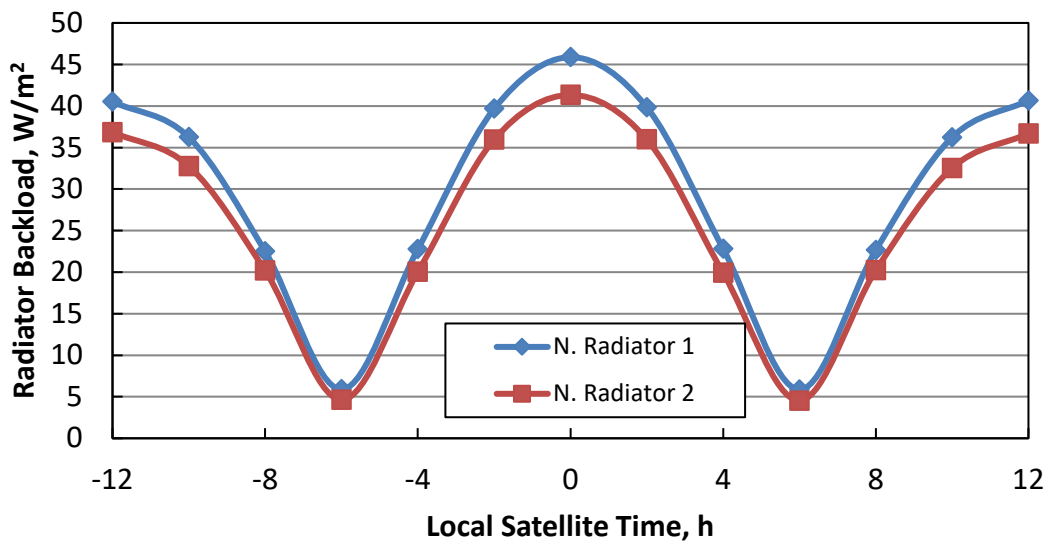


Figure A-1: Baseline S/C configurations cannot satisfy 25 W/m<sup>2</sup> backload limit

Table A-1: Representative Baseline S/C Configuration Thermal Backload

S/C	South-facing radiators (Winter Solstice)	North-facing radiators (Summer Solstice)
S/C1	48 W/m <sup>2</sup>	44 W/m <sup>2</sup>
S/C2	79 W/m <sup>2</sup>	---
S/C3	49 W/m <sup>2</sup>	46 W/m <sup>2</sup>



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Rationale for Change –

Per the HPIG (Section 4.8.6) The Instrument should maintain its own instrument temperature requirements. As a thermally isolated payload, the Instrument has to manage its own thermal properties without support from the Host Spacecraft. Section 4.8.5 addresses radiative heat transfer between the payload and spacecraft.

It was not feasible to change the TEMPO instrument radiator design at this phase of the project so the instrument heat rejection responsibility was moved to the spacecraft side of the interface where the spacecraft bus will provide dedicated area on the S/C bus radiator to support the payloads temperature requirements.

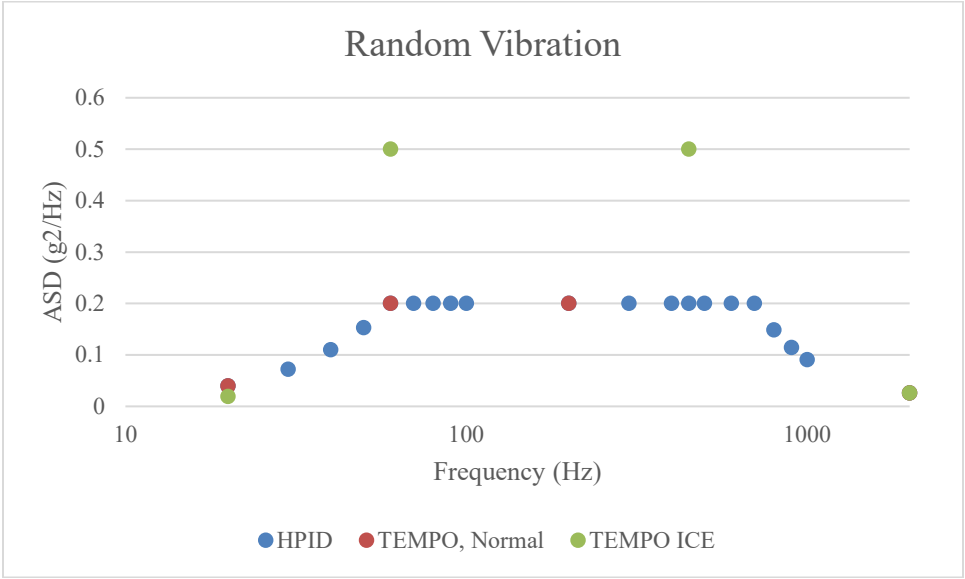
### A.2.3 TEMPO ERD

There are several instances where the TEMPO ERD specifies testing levels or environments that are not congruent with the HPIG. Most of these instances were minor discrepancies where the payload was not being tested to the levels mentioned in the HPIG, and were negotiated with the spacecraft bus manufacturers as necessary. These are noted with corresponding HPIG section in Table A-2.

**Table A-2: Negotiated TEMPO ERD Tests and Corresponding HPIG Sections**

Concern	HPIG Section
<b>Sine Vibration</b>	4.11.4.5
<b>Acoustics</b>	4.11.4.7
<b>Shock</b>	4.11.4.8
<b>Launch Pressure Profile</b>	4.11.4.3
<b>Meteoroid and Space Debris</b>	4.11.6.3

In other instances, more detailed discussions between the instrument provider and bus manufacturers were had to ensure the Do No Harm specification is met. For instance, the Interface Control Electronics (ICE) are sensitive to vibration, and the instrument developers requested that they be tested at a lower level than the spacecraft provider designated to be the appropriate level for the environment. The appropriate random vibration testing level is captured in Figure A-2.



**Figure A-2: Random Vibration Testing Level**

**A.2.4 Pointing Requirements**

The TEMPO instrument operates with strict pointing requirements to ensure accurate measurements. After selection, it was apparent that the spacecraft gyros did not meet the pointing needs of the TEMPO payload, and additional rework was needed.

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## **Appendix B Hosted Payload Concept of Operations**

### **B.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.

#### **B.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.

#### **B.1.2 Document Scope**

The purpose of this CONOPS document is to give an overview of LEO and GEO satellite 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.

### **B.2 COMMON INSTRUMENT INTERFACE PHILOSOPHY**

This CONOPS supports the “Do No Harm” concept as described in Section 2.

### **B.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.

#### **B.3.1 General Information**

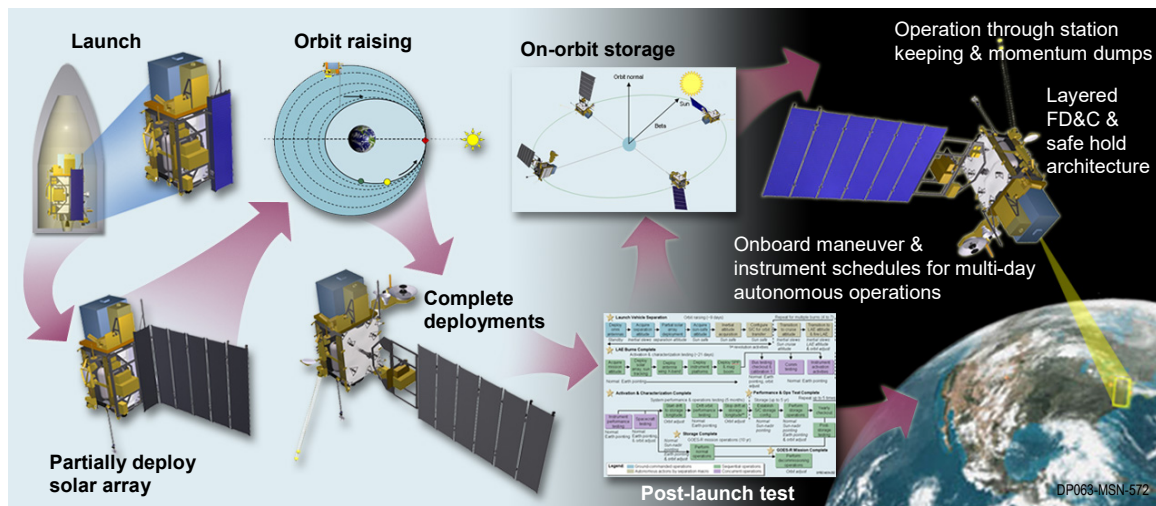
[LEO] Nominal Orbit: The Host Spacecraft will operate in a Low Earth Orbit with an altitude between 350 and 2000 kilometers with eccentricity less than 1 and inclination between zero and 180°, inclusive (see Section 3.1). LEO orbital periods are approximately 90 minutes.

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[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°.

### B.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 B-1.



**Figure B-1: Summary of Transition to Normal Operations**

#### Launch and Ascent

During this phase, the Host Spacecraft 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 Host Spacecraft 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 Host Spacecraft, 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 communication 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.

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[LEO] The Host Spacecraft 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 communication 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

[LEO] An on-orbit storage location may be used if the Host Spacecraft is part of a constellation where the current operational spacecraft has not yet been decommissioned. The Host Spacecraft may inject into this location to perform the checkout of itself and 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 Host Spacecraft checkout process will begin. Each subsystem will be fully powered and checked out in a systematic manor. Once the Host Spacecraft is successfully checked-out and operational, its communication payload checkout begins, also in a systematic manor. When both the Host Spacecraft and its communications payload are successfully checked-out, the owner/operator will transition to normal operations.

#### Normal Operations

The Host Spacecraft 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 Instrument 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.

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## 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 Host Spacecraft 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 Instrument will be powered back on.

## Instrument Safehold

The Instrument will transition to this mode due to one of two reasons, either due to a Host Spacecraft failure or an Instrument failure.

In the event the Instrument experiences a failure of some sort, it must 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. 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, the Instrument must be capable of safely decommissioning itself via ground commands.

During the end of life phase, the Instrument will be completely unpowered, unless survival heaters are required to ensure Host Spacecraft safety. 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 of these operations, the Host Spacecraft will consider the Instrument as a simple mass model that does not affect operations.

## De-commissioning

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At the end of the Host Spacecraft’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 Spacecraft maneuvers may span several days to relocate where it will be powered down and its mission life ended.

## B.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. If this occurs, the Satellite Operator may or may not inform the Instrument operators prior to suspension of operations.

### B.4.1 Instrument Modes of Operation

Table B-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 B-1: 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
<i>Note: Host Spacecraft controls Instrument power.</i>							

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 E.

#### Off/Survival Mode

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

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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 Host 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 Host Spacecraft host and would result in 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).

#### 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 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 Host 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 Host 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

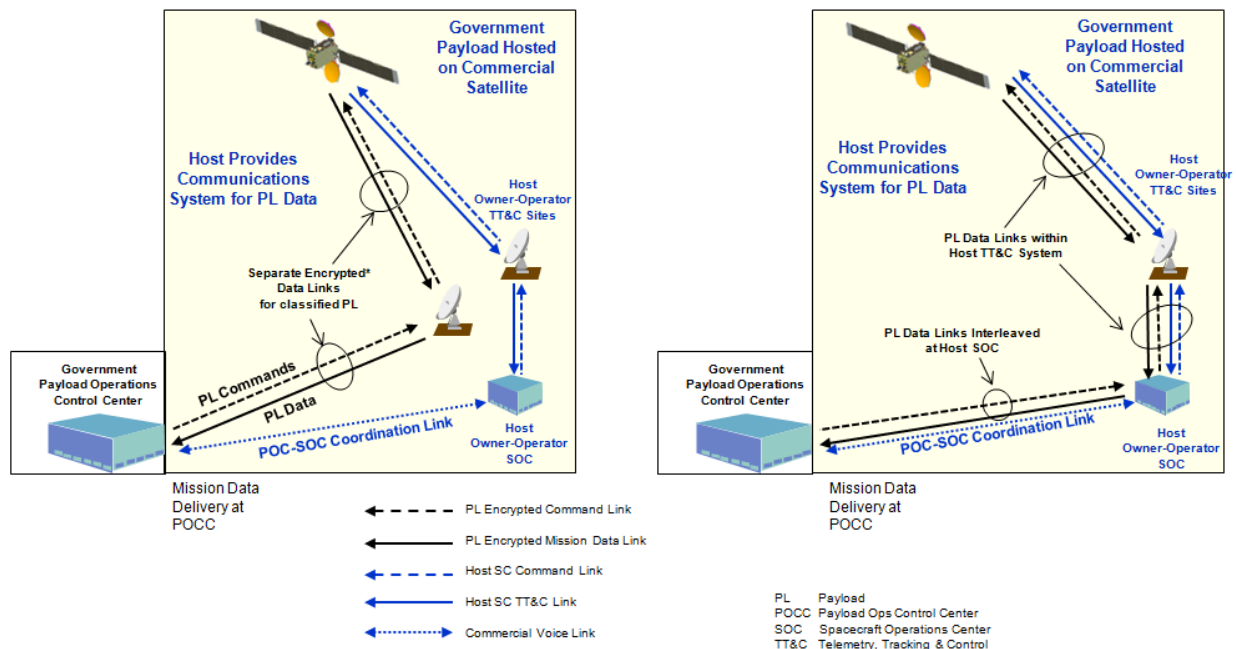


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the Instrument is in Safe mode, the data collected and transmitted to the HPOC should be limited to health and status information only. Nominal transitions into the Safe mode are either from the Initialization mode or the Operation mode. Nominal transitions out of the Safe mode are into either the Off/Survival mode or the Operation mode.

### B.4.2 Hosted Payload Commanding and Data Flow

The reference architecture for a typical hosted payload mission is depicted in Figure B-2 below. Variations to this architecture may be implemented for specific missions depending on the mission specific requirements and associated payload concept of operations.



**Figure B-2: Notional Hosted Payload Mission Architecture**

Several options are available to transport hosted payload command and telemetry data through the host satellite and the host’s ground data network depending on the required bandwidth of the payload data and the capability of the host TT&C system.

- Command and control of the hosted payload can be implemented through a dedicated communications path (left figure) for high bandwidth payload data or through an embedded (right figure) link through the host Telemetry, Tracking, and Command (TT&C) system if the payload data can be multiplexed within the host’s TT&C systems,
- Transmission of payload data through the host spacecraft and ground segment to the Government’s POCC,
- Operations support by the commercial Spacecraft Operations Center (SOC) to the POCC will require coordinated operations plans that clearly define the support and coordination

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required and regularly updated to ensure the objectives of the commercial and hosted payload missions are satisfied,

These data flow options can support a variety of payload command and control approaches from all payload commanding performed at the POCC, a mix of commanding from both the POCC and the SOC, to all commanding to be provided by the commercial SOC. An appropriate approach should be selected based on the needs of the hosted payload and the capabilities of the host space and ground systems.

The reference mission architecture is intended to support different types of Government payload missions. This architecture provides payloads the flexibility to implement cost effective solutions for payload command and control that balance the needs of the payload with the capabilities available from the commercial host spacecraft and ground systems.

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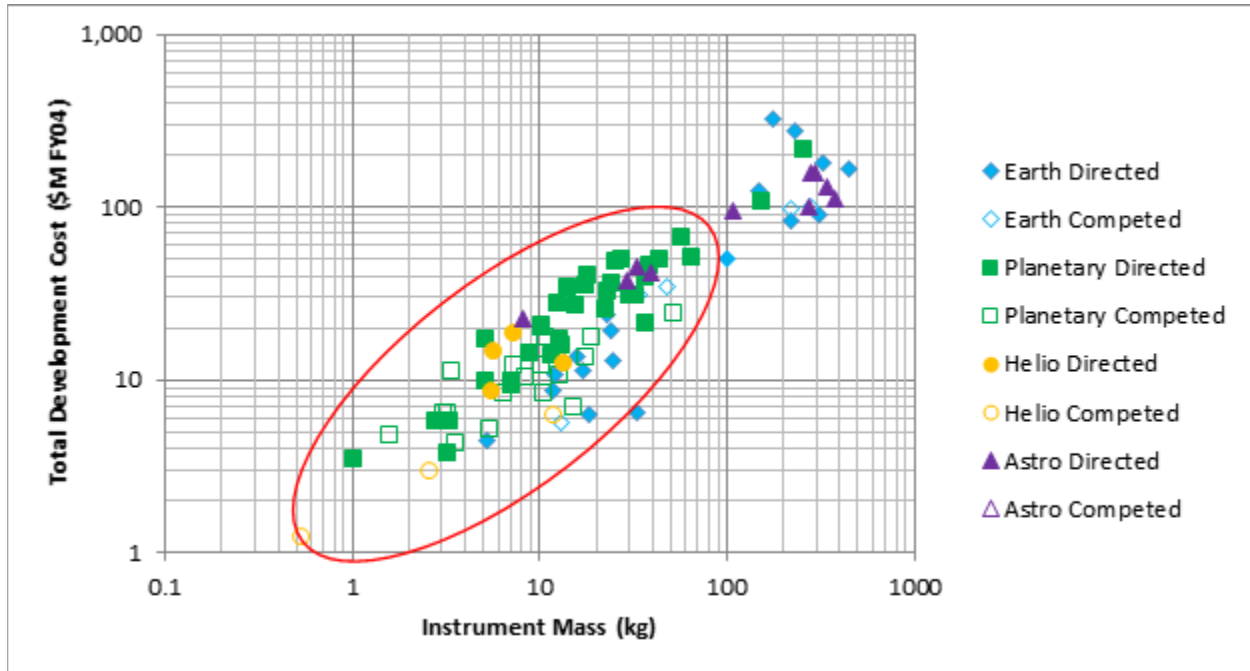
## Appendix C Analysis for LEO Guidelines

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 C-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 C-1: Distribution of NICM Instruments Among Science Mission Directorate Divisions**

SMD Division	Directed	Completed	Non-NASA	Total
Earth	18	5	5	28
Planetary	35	18	1	54
Heliophysics	5	3	1	9
Astrophysics	10	1	0	11
<b>Total</b>	<b>68</b>	<b>27</b>	<b>7</b>	<b>102</b>

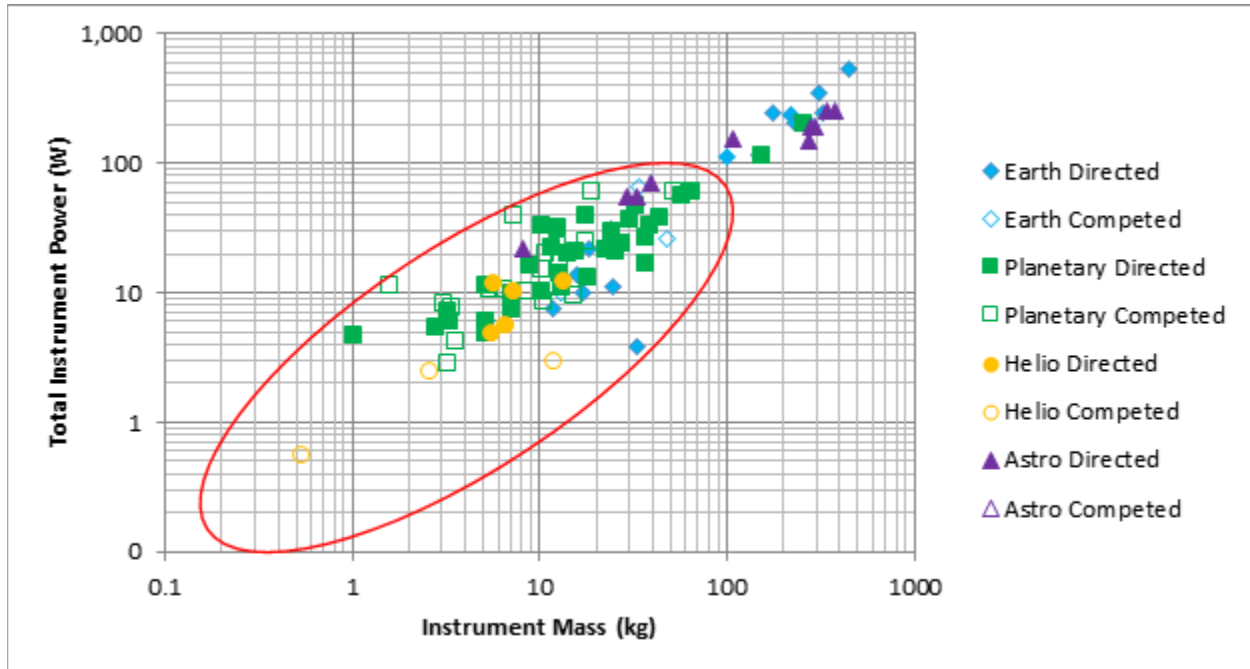
In analyzing the data, one may easily conclude that the development cost of an instrument is a function of multiple parameters such as: mass, power, data rate, year built, SMD division and acquisition strategy. With further analysis, it is clear that these parameters are not independent of each other 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 C-1.



**Figure C-1: Instrument Mass vs. Development Cost**

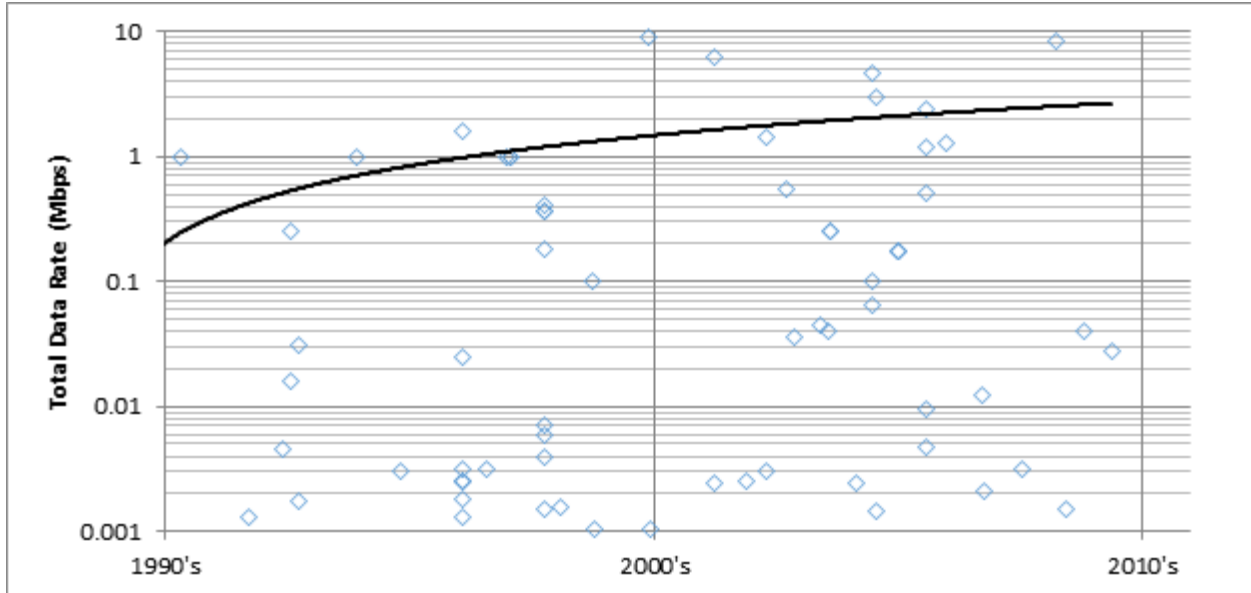
In further examination of the data, specifically the Earth Science instruments that are outside the ellipse in Figure C-1, the specific instrument details indicate that they were primary instruments that drove the mission requirements. This is certainly the case for the Aura mission with the MLS and 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 pairing with an HPO.

Figure C-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 C-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 C-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*, although some instruments may 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. This clearly illustrates the need to pair an Instrument to a compatible HPO as early as possible. 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 C-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.

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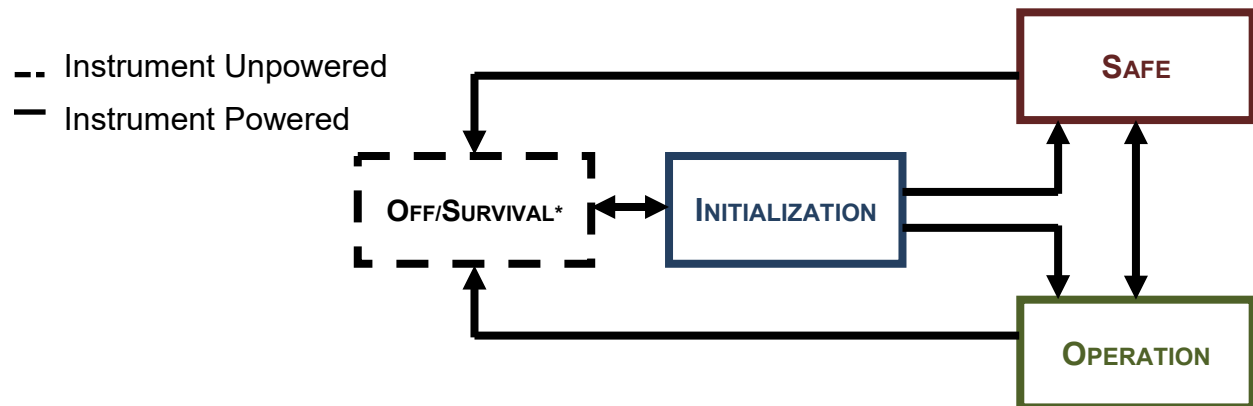
## Appendix D Instrument Modes

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.

### D.1 MODE GUIDELINES

#### Basic Modes

Instruments should function in four basic modes of operation: OFF/SURVIVAL, INITIALIZATION, OPERATION, and SAFE (see Figure E-1). Within any mode, the Instrument may define additional sub-modes specific to their operation (e.g. STANDBY, DIAGNOSTIC, MEASUREMENT, etc.).



**Figure E-1: Instrument Mode Transitions**

#### OFF/SURVIVAL Mode, Survival Heater OFF State

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 Host 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.

#### OFF/SURVIVAL Mode, Survival Heater ON State

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

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#### *Post-Launch Instrument Survival Circuit Initiation*

The Host 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 Host 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 to the Host 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 Host Spacecraft and ground operations team.



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### *Resources*

When in OPERATION mode, the Instrument should be supported by all allocated Host Spacecraft resources.

### 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 Host 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 Host Spacecraft when it has completed a transition to SAFE mode.

## **D.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 E-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 Host Spacecraft should transition the Instrument to OFF/SURVIVAL mode in the event of a severe Spacecraft emergency.

#### *Instrument Operational Power*

The Host 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 Host Spacecraft or ground operations team to transition to OPERATION (or SAFE) mode.

### SAFE Mode Transitions

#### *Command Trigger*

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

### OPERATION Mode Transitions

#### *Trigger*

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

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## Appendix E Examples of Data Deliverables for Verification

This section provides the types of data items that will, be required for interface verification with the host spacecraft. This is not an exhaustive list and is not necessarily all-inclusive but provides examples of data that is usually required. This example list applies to the hosted payload (instrument)-to-host spacecraft interfaces only. Additional verification for the hosted payload (instrument) itself will be required.

- Compliance matrix
- Thermal analysis
- Mechanical Analysis
- Failure Modes and Effects Analysis
- Mass Properties Report
- Safety/hazard Analyses and Report
- Qualification and Acceptance Test Report
- Qualification Certification (and associated analysis)
- End Item Data Package

Each host spacecraft project may have an individual list that may differ from the examples cited above. The Hosted Payload Developer should make every attempt to ascertain the actual interface verification requirements as soon as practical from the host spacecraft developer.

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## **Appendix F Examples of Payload-Provided Hardware and Associated Tasks**

Hosted payload providers should be prepared to provide additional hardware that support their payload, in addition to several tasks associated with their specific hardware. Examples cited below:

- Payload providers should supply any required shipping containers, shipping materials, accelerometers, temperature, pressure and humidity sensors and contamination monitors to monitor and record the environment during payload transportation, storage and shipping to the Host Spacecraft integration site.
- Payload suppliers are responsible for all shipment, insurance, tax, and import/export fee costs for delivery (flight and non-flight hardware) to the Host Spacecraft integration site.
- Payload providers should provide all necessary information and support for any Export Control application, license, evaluation, and agreement required to support the payload movement as required throughout the integrated space vehicle mission phases.
- Payload suppliers are responsible for unpacking and incoming inspection & test (flight, non-flight hardware, GSE) prior to acceptance by the Host Spacecraft.
- Payload suppliers are responsible for providing, maintaining and performing certification, calibration, maintenance, and archiving tasks for all payload unique GSE.
- Payload suppliers are responsible for any intra-payload (not connected to any Commodity Bus interface) flight and non-flight harnessing and cables.
- Payload suppliers should provide non-flight harnessing, thermal treatments and structural elements as required for any pre-shipment testing.
- Payload suppliers should provide storage requirements, safe-to-mate procedures; functional testing procedures, scripts, expected results, constraints and sensitivities as related to the payload tasks to be performed at the vehicle level.
- Payload suppliers should plan for long-term storage of payloads, as required, by the host spacecraft.