INTERFACE REQUIREMENTS DOCUMENT (IRD)

for

NATIONAL POLAR-ORBITING OPERATIONAL ENVIRONMENTAL SATELLITE SYSTEM (NPOESS) SPACECRAFT AND SENSORS

Prepared by

Associate Directorate for Acquisition
NPOESS Integrated Program Office

Version 3

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Integrated Program Office
Silver Spring MD 20910

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NATIONAL POLAR-ORBITING OPERATIONAL ENVIRONMENTAL SATELLITE SYSTEM (NPOESS) SPACECRAFT SENSORS

Approved by

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Associate Director for Acquisition

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Change Record

CCB Approved Changes:

This version of the IRD contains changes resulting from comments, SWIG inputs, and directives including formal and informal CCB directives (CCBDs). Formal CCB action against the SRDs also affected the IRD. The following formal CCBD changes were made against the indicated IRD version and are included in this current version of the IRD.

CCBD #	Items	IRD Version
98079	1	2J
98101	3 thru 5	2J
99005	1 thru 3	2J
99006	1 thru 18	2J
99008	1 thru 3	2J
99010	1 thru 6	2J
99013	1 thru 3	2J

Changes for Informal CCBDs:

The following informal CCBD changes were made against the indicated IRD version and are included in this current version of the IRD. Because the IRD is not under formal CCB control these CCBDs have not been distributed outside the IPO.

CCBD # - Date	Items	IRD Version
A - 2/23/99	1	2J
B - 2/23/99	1	2J
C - 2/23/99	1 thru 3	2J
D - 2/23/99	1	2J
E - 2/23/99	1	2J
F - 2/23/99	1 thru 7	2J

Other Changes:

The majority of the changes to version 2 of the IRD are a result of working group meetings with the IPO and ATSP contractors and internal IPO coordination. There is no formal tracking of these changes.

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1.0 SCOPE

1.1 Introduction

The purpose of this Interface Requirements Document (IRD) is twofold. The first is to establish a baseline for interface requirements between the National Polar-orbiting Operational Environmental Satellite System (NPOESS) spacecraft and sensors. Second, it serves as a core building block on which the sensor-to-spacecraft interface can be designed.

The spacecraft-to-sensor interface requirements are broken down into four primary groups: mechanical, power, data, and thermal. A notional diagram of the top-level functional interfaces for any sensor is shown in Figure 1. In addition environmental, software, testing, contamination, launch environment, and safety requirements are defined.

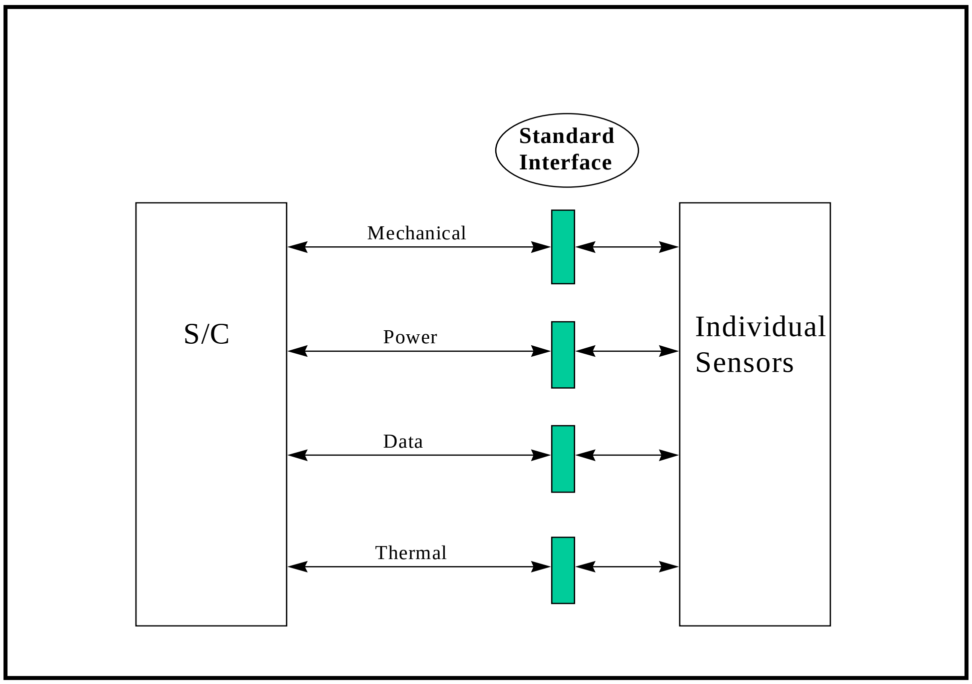

Figure 1. Notional Spacecraft-To-Sensor Functional Interfaces.

This document is intended to provide the basic interface requirements between the sensors, which will be developed first and a spacecraft that will be designed later in the program.

IRD1.1-1
The integrating contractor and the sensor contractors shall each meet their respective interface requirements as defined in this document. Sensor Requirement Documents (SRD) establish the allocation of the system requirements to different NPOESS sensors, and define the sensors' requirements, as well as sensor unique interface requirements.

IRD1.1-2
After award of the spacecraft integration contract, each sensor developer and the integrating contractor shall jointly write an Interface Control Document (ICD) which defines the details of the sensor-to-spacecraft interface and sensor accommodation information.

1.2 System Overview

The U.S. Government currently operates and maintains two polar-orbiting meteorological satellite programs. The U.S. Air Force (USAF) operates the military's Defense Meteorological Satellite Program (DMSP), while the National Oceanic and Atmospheric Administration (NOAA) operates the Polar-orbiting Operational Environmental Satellite (POES) program.

The DoD predecessor program to NPOESS was the DMSP Block 6. Upon completion of Concept Studies started in 1988, two Risk Reduction contracts were awarded in July 1991 to define a military next-generation satellite system (including the Space, Command, Control, and Communications (C^3), and User Segments) to provide meteorological, oceanographic, and solar-environmental support to all DoD users. The purpose of the Risk Reduction effort was to develop preliminary system designs and perform key demonstrations for the baseline system.

The comparable Department of Commerce (DOC) program was the POES Follow-On program, also known as the O, P, Q acquisition. Phase A (advanced study phase) for these satellites was initiated in 1991. Some of the initial design characteristics were common interfaces with the European meteorological operational (METOP) program, growth room to accommodate selected, proven Earth Observation System (EOS) instruments, and a three year design life. The O, P, Q plan was subsequently changed, and the decision was made to procure two additional spacecraft, called N and N-prime based on the Television Infrared Orbital Satellite (TIROS) K, L, M spacecraft design. A contract for N and N-prime spacecraft was awarded in December 1994.

In February 1993, the Committee for Science, Space and Technology requested DoD and NOAA to begin looking at opportunities to integrate the DMSP and POES programs and investigate the use of technologies developed by the EOS program. A tri-agency study with DoD, NOAA, and NASA was initiated in June 1993 at the request of Congress. Convergence was also an initiative of the National Performance Review. The result of this tri-agency study was an agreement to develop a converged operational polar-orbiting environmental satellite system with a transition period beginning in the late 1990s leading to a fully converged system by the mid 2000s. This agreement was formalized by the Office of Science and Technology Policy (OSTP) with the Implementation Plan for a Converged Polar-orbiting Environmental Satellite System, dated May 2, 1994. On May 5, 1994, a Presidential Decision Directive (PDD/NSTC-2) was signed, directing DoD and DOC to converge their independent operational polar-orbiting environmental satellite systems into a single, integrated system. This decision, as part of a National Performance Review recommendation, was expected to save the U.S. Government up to an estimated $300 million in FY94-FY99 with additional savings expected after FY99. A tri-agency Memorandum of Agreement (MOA), dated May 26, 1995 specifies the roles, responsibilities and agreements between the agencies.

The NPOESS Program is required to provide, for a period of at least 10 years after Initial Operational Capability (IOC), a remote sensing capability to acquire, receive at ground terminals, and disseminate to processing centers, global and regional environmental imagery and specialized meteorological, climatic, terrestrial, oceanographic, solar-geophysical, and other data in support of DOC/NOAA mission requirements, and DoD peacetime and wartime missions.

It is anticipated that operational data will be collected by satellites flying in sun-synchronous near-polar orbits at approximately 833 km altitude with the following nominal nodal crossing times - 0530, 0930, and 1330. Satellites in the 0530 and 1330 orbits are considered U.S. assets and will be developed, acquired, deployed, and operated by the U.S. Satellites in the 0930 orbit are European assets and will be developed, acquired, deployed, and operated by the Europeans. Pending an international agreement, sensors will be exchanged between the United States Government (USG) and the European Organization for the Exploitation of Meteorological Satellites (EUMETSAT). Under this arrangement, called the Joint Polar System (JPS), some USG sensors will fly on EUMETSAT satellites (designated METOP). In this way, the USG and EUMETSAT requirements will be met jointly by NPOESS satellites and METOP satellites beginning with METOP-3.

It is anticipated that the NPOESS spacecraft will be launched using a medium launch vehicle (MLV or EELV) class of booster. The NPOESS program is comprised of four segments: 1) Space; 2) Launch Support; 3) C^3; and, 4) Interface Data Processor (IDP). Standardization (which includes compatibility, interoperability, interchangeability, and commonality) of DoD, DOC, and NASA systems, components, and interfaces is a primary goal of NPOESS.

1.3 Document Overview

This document comprises five sections.

a. Scope
b. Applicable Documents
c. Requirements
d. Testing Provisions
e. Notes

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2.0 APPLICABLE DOCUMENTS

IRD2.0-1
The following documents of the exact issue shown form a part of this IRD to the extent specified herein. In the event of conflict between the documents referenced and the contents of this IRD, the latter shall be the superseding requirement.

2.1 Compliance Documents

CCSDS 203.0-B-1 January 87	CCSDS Recommendations for Space Data System Standards. Telecommand, Part 3: Data Management Service, Architectural Definition, Issue 1
CCSDS 701.0-B-2 October 1989	Consultative Committee for Space Data Systems (CCSDS) Recommendations for Advanced Orbiting Systems (AOS), Networks and Data Links: Architectural Specification
EWR 127-1 31 March 95	Range Safety Requirements, Eastern and Western Range (EWR)
MIL-A-83577B February 88	Assemblies, Moving Mechanical, for Space and Launch Vehicles
MIL-STD-461D January 93	Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference
MIL-STD-462D January 93	Measurements of Electromagnetic Interference Characteristics
MIL-STD-882c January 93	System Safety Program Requirements
MIL-STD-975 Cancellation notice 3, 5 May 98	NASA Standard Electrical, Electronic and Electromechanical (EEE) Parts List
MIL-STD-1522A 18 May 84 Notice 2: 20 Nov 86	Standard General Requirements for Safe Design and Operation of Pressurized Missile and Space Systems
MIL-STD-1540C September 94	Test Requirements for Space Vehicles (Tailored)
MIL-STD-1541A 30 December 87	Electromagnetic Compatibility Requirements for Space Systems
MIL-STD-1547A December 92	Electronic Parts, Materials, and Processes for Spacecraft and Launch Vehicles
SAE-AS-1773	Fiber Optics Mechanization of an Aircraft Internal Time Division Command/Response Multiplex Data Bus

2.2 Reference Documents

40 CFR Parts 1500-1508	National Environmental Policy Act Specifications
ASTM E595-93 15-June-93	Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials for Outgassing in a Vacuum Environment, American Society for Testing and Materials (ASTM)
ANSI STD X3/159-1989	Programming Language - C (withdrawn and replaced by ANSI/ISO 9899-1990)
DD 1494	Air Force Frequency Allocation Form?
DOD-E-83578A May 96	Explosive Ordnance for Space Vehicles, General Specifications for
DOD-W-83575 04-June-98	General Specification for Wiring Harness, Space Vehicle, Design and Testing
DOD HDBK-263	Electrostatic Discharge Control Handbook for Protection of Electrical and Electronic Parts. Assemblies and Equipment
EELV SIS	Evolved Expendable Launch Vehicle Standard Interface Specification
FED-STD-209E 11 September 92	Airborne Particulate Cleanliness Classes in Cleanrooms and Clean Zones
ICD GPS 060 2 June 86	GPS User Equipment, Precise Time and Time Interval (PTTI) Interface
ISO/TC 209 (ISO/DIS 14644-1)	Cleanrooms and Associated Controlled Environments
MAN91-2010002	Explosive Safety Standards
MIL-C-24308 26 January 89 Amendment: June 93 Supplement 1: 5/93	General Specification for Connectors, Electric, Rectangular, Non-Environmental, Miniature, Polarized Shell, Rack and Panel
MIL-C-38999	Connectors, Electrical
MIL-STD-498 5 December 94	Software Development and Documentation
MIL-STD-1246C 11 April 94	Product Cleanliness Levels and Contamination Control Program
MIL-STD-1543B October 88	Reliability Program Requirements for Space and Missile Systems
MIL-STD-1546B	Parts, Materials and Processes Control Program for Space and Launch Vehicles
MIL-STD-1629	Procedures for Performing a Failure Mode, Effects and Criticality Analysis
MIL-STD-1815A Cancellation notice 17 June 97	Ada Programming Language (ANSI/Mil Std 1815A-1993), replaced by Information Technology - Programming Languages - Ada, (ANSI/ISO/IEC 8652-1995), April 95

3.0 REQUIREMENTS

The requirements stated in this document are not of equal importance or weight. The following three paragraphs define the weighting factors incorporated in this specification.

a. *Shall* designates the most important weighting level; that is, mandatory. Any deviations from these contractually imposed mandatory requirements require the approval of the contracting officer.
b. Should designates requirements requested by the government and are not mandatory. Unless required by other contract provisions, noncompliance with the *should* requirements does not require approval of the contracting officer.
c. *Will* designates the lowest weighting level. These *will* requirements designate the intent of the government and are often stated as examples of acceptable designs, items and practices. Unless required by other contract provisions, noncompliance with the *will* requirements does not require approval of the contracting officer and does not require documented technical substantiation.

The values specified in the IRD are one sigma unless specified otherwise.

3.1 Mechanical Requirements

IRD3.1-1
All requirements specified in Section 3.1 shall be met at the mechanical interface and shall be consistent with the SRD allocations.

3.1.1 Sensor Envelopes

3.1.1.1 Sensor Launch Mode Envelope

IRD3.1.1.1-1
Sensor components in the launch configuration shall be contained within the sensor launch mode envelope as allocated within the SRD.

3.1.1.2 Sensor On-Orbit Envelope

IRD3.1.1.2-1
Sensor components in the on-orbit configuration shall be contained within the sensor on-orbit envelope as allocated in the SRD.

IRD3.1.1.2-2
For a sensor with mechanisms that cause a change in the external envelope or external surfaces of the sensor, the initial and final configurations, as well as the swept volumes, shall be documented in the sensor-spacecraft ICD.

3.1.1.3 Sensor Envelope Documentation

IRD3.1.1.3-1
The sensor component envelope (including thermal blankets) shall be documented in the sensor-spacecraft ICD by engineering drawings with a set of “not to exceed” dimensions.

IRD3.1.1.3-2
The sensor developer shall ensure that the "swept" or deployed volume is verified accounting for all distortions and misalignments.

3.1.1.4 Stowed and Critical Clearances

IRD3.1.1.4-1
The integrating contractor is responsible for defining available sensor volume and making sure the satellite fits within the dynamic envelope of the launch vehicle’s fairing. This is controlled with the satellite-to-launch vehicle Interface Control Document (ICD). Both the integrating contractor and the sensor developer must work together to insure that the stowed, deploying, and final deployed positions of the sensor shall clear all obstacles including obstacles on the spacecraft, other sensors, and the launch vehicle.

IRD3.1.1.4-2
If the sensor is to be deployed, all obstacles shall be cleared in the stowed, deploying, and final deployed positions.

IRD3.1.1.4-3
If the sensor has moving assemblies, all obstacles shall be cleared within the region of motion.

IRD3.1.1.4-4
As a baseline, a 2.5 cm. clearance between the sensor and surrounding structure shall be maintained.

IRD3.1.1.4-5
A critical clearance analysis shall be implemented to identify areas where the clearance rule may be violated, accounting for miscellaneous support hardware such as wire bundles and thermal blankets, deflections due to launch loads, launch vibrations, 1-g sag, thermal distortions, and misalignments, with all identified areas tracked in a critical clearance document.

3.1.2 Fields of View

IRD3.1.2-1
All sensor fields of view shall be documented in the sensor-spacecraft ICD.

3.1.3 Mass Properties

3.1.3.1 Sensor Mass Allocation

IRD3.1.3.1-1
The sensor mass for the delivered sensor hardware shall be less than or equal to the maximum value allocated for that sensor in the applicable SRD.

3.1.3.2 Sensor Mass Documentation

IRD3.1.3.2-1
The mass of the sensor shall be documented in the sensor-spacecraft ICD and shall be measured to +/- 0.1 Kg.

3.1.3.3 Sensor Mass Variability

3.1.3.3.1 Sensor Mass Variability Documentation

IRD3.1.3.3.1-1
Sensor mass expulsion rates and substances, if any, shall be documented in the sensor-spacecraft ICD.

3.1.3.3.2 Center of Mass Location

IRD3.1.3.3.2-1
The stowed and deployed center of mass of each sensor component shall be measured and reported to +/- 6 mm, referenced to the sensor coordinate axes as documented in the sensor-spacecraft ICD.

3.1.3.4 Moments of Inertia

3.1.3.4.1 Moments of Inertia Measurement

IRD3.1.3.4.1-1
The sensor moments of inertia shall be defined using the satellite axis convention passing through the sensor center of mass.

3.1.3.4.2 Moments of Inertia Accuracy

IRD3.1.3.4.2-1
Moments of inertia values shall be accurate to within +/- 5%.

3.1.3.4.3 Moments of Inertia Documentation

IRD3.1.3.4.3-1
The moments of inertia of each separately mounted component of the sensor shall be documented in the sensor-spacecraft ICD, referenced to the sensor coordinate axes.

3.1.3.4.4 Moments of Inertia Variation Documentation

IRD3.1.3.4.4-1
If the sensor contains movable masses, expendable masses, or deployables, the moments of the inertia values for each configuration shall be documented in the sensor-spacecraft ICD.

3.1.4 Mounting

3.1.4.1 Mounting Method

IRD3.1.4.1-1
The mounting method shall accommodate manufacturing tolerance, structural, and thermal distortions.

IRD3.1.4.1-2
The method by which each sensor component is mounted to the spacecraft shall be defined in the sensor-spacecraft ICD.

3.1.4.2 Mounting Interface

3.1.4.2.1 Mounting Interface Documentation

IRD3.1.4.2.1-1
The spacecraft mounting interface for each sensor component shall be documented in the sensor-spacecraft ICD.

3.1.4.2.2 Mounting Hole Coordinates and Dimensions

IRD3.1.4.2.2-1
Coordinates and dimensions of the holes for mounting hardware shall be specified at the mechanical interface and defined in the sensor-spacecraft ICD.

3.1.4.3 Mounting Hardware

3.1.4.3.1 Mounting Hardware Provider

IRD3.1.4.3.1-1
The integrating contractor shall provide all sensor mounting hardware including secondary structures.

3.1.4.3.2 Mounting Hardware Documentation

IRD3.1.4.3.2-1
Sensor mounting hardware shall be defined and documented in the sensor-spacecraft ICD.

3.1.4.3.3 Mounting Surface Requirements

IRD3.1.4.3.3-1
Finish and flatness requirements for the mounting surfaces shall be specified by the integrating contractor and documented in the sensor-spacecraft ICD.

3.1.4.4 Mounting Location and Documentation

IRD3.1.4.4-1
The integrating contractor working with the sensor contractor shall determine the location of the sensor on the spacecraft.

IRD3.1.4.4-2
This location shall be documented in the sensor-spacecraft ICD.

3.1.4.5 Drill Templates

3.1.4.5.1 Drill Template Usage

IRD3.1.4.5.1-1
If drill templates are used for simple planar interfaces, then sensor equipment, spacecraft, and test fixture interfaces shall be drilled using templates.

3.1.4.5.2 Drill Template Fabrication Requirements

IRD3.1.4.5.2-1
The drill template fabrication, functional requirements (e.g. material, use of inserts, etc.), and orientation information shall be provided by the sensor contractor.

3.1.4.5.3 Drill Template Provider

IRD3.1.4.5.3-1
The sensor provider shall provide the drill template to the integrating contractor. The drill template shall include appropriate alignment, orientation, and location reference information.

3.1.5 Alignment

3.1.5.1 Alignment References

IRD3.1.5.1-1
The sensor contractor shall provide a sensor alignment reference composed of an alignment cube and a mounting surface datum.

IRD3.1.5.1-2
The spacecraft shall provide a spacecraft alignment reference.

IRD3.1.5.1-3
Both the sensor alignment cube and the spacecraft alignment reference shall be viewable from two orthogonal directions.

3.1.5.2 Alignment Responsibilities

IRD3.1.5.2-1
The sensor contractors shall be responsible for measuring the alignment angles between the sensor boresight (line-of-sight), if applicable, and the sensor alignment reference.

IRD3.1.5.2-2
The integrating contractor shall be responsible for aligning the sensor alignment reference to the spacecraft attitude reference.

3.1.5.3 Alignment Control

IRD3.1.5.3-1
The integrating contractor shall control the alignment of the sensor alignment reference with respect to the spacecraft attitude reference to within 0.25 degrees (3 sigma).

3.1.5.4 Alignment Knowledge

3.1.5.4.1 Measurement Uncertainty

IRD3.1.5.4-1
The integrating contractor shall measure the alignment between the sensor alignment reference and the spacecraft attitude reference.

IRD3.1.5.4-2
The rms uncertainty in the alignment knowledge shall be less than 25 arcsec per axis.

IRD3.1.5.4-3
This uncertainty shall include (if applicable), but not be limited to, measurement uncertainties, alignment shifts due to vibration environments in both ground processing and launch, uncompensated gravity effects, hygroscopic effects of composite materials, and component removal and replacement.

3.1.5.4.2 Structural Thermal Distortion Uncertainty

IRD3.1.5.4.2-1
The integrating contractor shall limit the rms uncertainty in the alignment between the sensor alignment reference and spacecraft attitude reference caused by structural thermal distortion due to the on-orbit thermal environment to be less than 10 arcsec per axis.

3.1.5.5 Spacecraft Attitude Reference

IRD3.1.5.5-1
The spacecraft shall use the attitude reference frame as defined by Section 3.1.5.1 to determine satellite pointing.

3.1.5.5.1 Attitude Reference Knowledge

IRD3.1.5.5.1-1
The spacecraft shall supply a three-axis attitude of the spacecraft attitude reference for ground processing.

IRD3.1.5.5.1-2
The supplied attitude shall be time-tagged and possess an angular rms accuracy per axis of 10 arcsec over a bandwidth of 0 to 10 Hz.

3.1.5.5.2 High Frequency Attitude Reference Errors

IRD3.1.5.5.2-1
The rms of all components of the attitude error of the spacecraft attitude reference with a frequency greater than 10 Hz shall be less than 5 arcsec per axis.

3.1.5.5.3 Attitude Reference Control

IRD3.1.5.5.3-1
The rms of the attitude reference control error over a bandwidth of 0 to 10 Hz shall be less than 0.01 deg per axis.

3.1.5.5.4 Attitude Reference Rate Error

IRD3.1.5.5.4-1
The rate error of the attitude reference frame shall be less than 0.03 deg/sec during all mission data collection periods.

3.1.5.6 Position Knowledge

IRD3.1.5.6-1
The spacecraft shall provide a spacecraft position estimate with a rms uncertainty of 25/25/25 meters for radial/in-track/cross-track components.

3.1.6 General Structural Design Requirements

IRD3.1.6-1
A-basis material allowables shall be used for design of metallic elements. An A-basis allowable is defined as a value where 99 percent of a population of values is expected to equal or exceed the allowable, with a confidence of 95 percent.

IRD3.1.6-2
B-basis material allowables shall be used for design of composite structures. A B-basis allowable is defined as a value where 90 percent of a population of values is expected to equal or exceed the allowable, with a confidence of 95 percent.

IRD3.1.6-3
Materials for the space equipment shall be selected for low outgassing, using NASA SP-R-0 022A (NASA JSC) as a guide, and resistance to the effects of incident radiation.

IRD3.1.6-4
Materials shall be selected that have demonstrated their suitability for the intended application.

IRD3.1.6-5
Materials shall be corrosion resistant or be suitably treated to resist corrosion when subjected to the specified environments.

IRD3.1.6-6
Where practicable, fungus inert materials shall be used.

IRD3.1.6-7
Class I Ozone Depleting Substances (ODS) shall not be used in the design, test, manufacture, integration and assembly, handling, transportation, operations, maintenance, or disposal of the sensor.

IRD3.1.6-8
Use of Class II ODS and Emergency Planning and Community Right to Know Act (EPCRA) Section 313 chemicals shall be identified and either eliminated or minimized, justified, and controlled.

IRD3.1.6-9
A Hazardous Materials Management Program shall be de eloped in accordance with NAS 411.

3.1.6.1 Structural Support

IRD3.1.6.1-1
The spacecraft shall provide structural support for the sensor such that the loads transmitted across the interface into the sensor do not exceed interface limit loads to be determined by the integrating contractor.

IRD3.1.6.1-2
The sensor and interface equipment shall be designed to design load factors determined by launch vehicle acceleration levels. A survey of typical launch vehicle environments (accelerations, frequencies, temperatures, etc.) is included in Section 3.9

3.1.6.2 Sensor Structural Dynamics

IRD3.1.6.2-1
When the sensor is in its launch-locked configuration, the fundamental natural frequency of the sensor shall be 50 Hz or greater, axial and lateral.

IRD3.1.6.2-2
For a deployable, the spacecraft integrating contractor shall specify a deployed frequency such that the sensor will not saturate the satellite’s control capability.

IRD3.1.6.2-3
The lowest natural frequency for a deployed sensor shall be greater than 6 Hz.

IRD3.1.6.2-4
The sensor contractor shall ensure that the sensor dynamic characteristics and control capability (e.g. a gimbaled sensor) will meet the requirements specified for the deployed frequency.

3.1.6.3 Interface Design Limit Loads Requirements

IRD3.1.6.3-1
The flight hardware shall be capable of withstanding all worst-case load conditions to which it may be exposed during ground (handling and transportation), pre-launch, launch, and on-orbit operations.

IRD3.1.6.3-2
Positive structural margins of safety shall be maintained so that the sensor can meet all of its design requirements after being subjected to the worst case loads combination.

IRD3.1.6.3-3
In those cases involving maintenance of sensor critical components for on-orbit operations, the precision elastic limit shall be used for structural materials. Table 1 shows the design and test options that are recommended for structures.

IRD3.1.6.3-4
The following minimum design factors of safety in Table 1 shall be applied to all loading conditions:

TABLE 1 FACTORS OF SAFETY

Design/Test Options	Factors of Safety (Yield)	Factors of Safety (Ultimate)	Test Level Factors
1. Dedicated test article	1.10	1.25	1.25
2. Test on flight article	1.25	1.40	1.25
3. Proof test each flight article	1.10	1.25	1.1
4. No-static test	1.60	2.00	N/A

IRD3.1.6.3-5
Note: The level of required analysis increases significantly with increased option number. For the no-static-test option, a detailed and comprehensive structural analysis is required and shall be available for review by the integrating contractor and Integrated Program Office (IPO).

IRD3.1.6.3-6
The dedicated test article option is a qualification test article that will be subjected to the maximum expected loads times a test level factor of 1.25. The test on flight article option refers to a protoqualification on the structure at the same test level factor. All composite structures and structural bonded joints shall be proof tested regardless of safety factor, but a metallic structure is usually qualified such that each unit will not have to be tested, or it is protoqualed. The no-static test option allows the capability of the structure to be determined via purely analytical methods, with the analytical models not being verified by test, but verified by the integrator/government for accuracy.

3.1.6.4 Combined Structural Dynamics Analysis

3.1.6.4.1 Combined Structural Dynamics Analysis Responsibility

IRD3.1.6.4-1
The integrating contractor shall be responsible for the combined structural dynamics analysis of the spacecraft bus and the sensors.

3.1.6.4.2 Combined Structural/Dynamic Analysis

IRD3.1.6.4.2-1
All models shall be exchanged in NASA Structural Analysis (NASTRAN) bulk data format. A test-verified model is preferred when available, and is required if the sensor lowest frequency is less than 50 Hz as shown by analysis.

3.1.6.4.3 Combined Structural Dynamics Analysis Results

IRD3.1.6.4.3-1
The integrating contractor shall provide the combined structural dynamics analysis results to both the customer program office and the sensor contractors.

3.1.6.4.4 Coupled Loads Analysis Results

IRD3.1.6.4.4-1
The launch vehicle/spacecraft coupled loads analysis will be performed by the launch vehicle contractor. The integrating contractor shall be responsible for providing the results of the launch vehicle/spacecraft coupled loads analysis in a standard format to the sensor contractors.

3.1.6.4.5 Structural Analyses

IRD3.1.6.4.5-1
A structural analysis using maximum equivalent loads shall be conducted by the sensor developer on all sensors**.**

IRD3.1.6.4.5-2
In addition, those sensors with modes under 50 Hz (as shown by the model) shall have a full modal survey test completed in a base fixed configuration to obtain all mode shapes and frequencies to correlate the dynamics model.

IRD3.1.6.4.5-3
An analysis using static loads shall be performed if those loads exceed the maximum equivalent values.

IRD3.1.6.4.5-4
The integrating contractor shall provide mission-specific information for maximum equivalent loads to the sensor developer for his static load analyses.

3.1.6.5 Pressurized System Design

IRD3.1.6.5-1
Sensors with pressurized systems shall follow the requirements in accordance with EWR 127-1 for the design of pressurized systems and using MIL-STD-1522 for guidance.

IRD3.1.6.5-2
Factors of safety for pressure loads shall be determined individually for each pressure vessel, based on tests to establish material characteristics and an analysis of life requirements and other environmental exposure.

IRD3.1.6.5-3
Proof and burst pressure factors shall be established at levels that ensure structural integrity, structural life, and safety throughout all phases. The values listed in Table 2 below are to be considered as limiting lower bounds.

TABLE 2 FACTORS OF SAFETY FOR PRESSURIZED COMPONENTS

Component	Design Ultimate	Acceptance (Proof)	Qualification (Burst)
Pneumatic Vessels (SVE)[a]	2.00	1.50 [b]	2.00 [b]
Pneumatic Vessels (GSE)[a]	4.00	2.00[b]	4.00
Lines, Fittings, and Hoses			
Less than 3.81 cm diameter	4.00	2.00 [b]	4.00 [b]
3.81 cm diameter and larger	1.50	1.10 [b]	1.50 [b]
Other Pressurized Components	2.50	2.00 [b]	2.50 [b]

See Notes below.

Notes:	
a	Factors of safety shown are minimum values applicable to metallic pressure vessels for which ductile fracture mode is predicted via a combination of stress and fracture mechanics analyses. Design of metallic pressure vessels for which brittle fracture mode is predicted by these analyses should be in accordance with fracture mechanics methodology, wherein the proof factor as well as the design ultimate factor of safety shall be established to provide a minimum of four times the specified service life against mission requirements. In addition, a fracture control program should be established to prevent structural failure due to the initiation or propagation of flaws or crack-like defects during fabrication, testing, and service life.
b	No measurable (TBR) yielding is permitted at acceptance (proof) test pressure and no rupture at qualification pressure.

3.1.7 Sensor Mass Model

The requirement for mass models will be determined prior to sensor preliminary design review (PDR).

3.1.8 Mechanisms and Deployables

IRD3.1.8-1
Sensor developers shall use the design and test guidelines provided in MIL-A-83577, to increase reliability of Moving Mechanical Assembly (MMA's) and facilitate integration and test activities.

3.1.8.1 Actuating Devices

IRD3.1.8.1-1
Non-explosive actuators shall be preferred over pyrotechnic devices wherever practicable in order to minimize shock loads. A fast release requirement can preclude this design option (paraffin actuators are too slow to release).

IRD3.1.8.1-2
Actuating circuitry shall be two-fault tolerant to unanticipated deployment or release.

3.1.8.2 Sensor Disturbance Allocations

IRD3.1.8.2-1
The integrating contractor shall provide estimates of allowable disturbance torques, vibration, and end-of-travel or latch-up loads to the sensor developer.

3.1.8.3 Sensor Mechanisms

IRD3.1.8.3-1
All sensor mechanisms which require restraint during launch shall be caged during launch without requiring power to maintain the caged condition.

3.1.8.4 Uncompensated Momentum

IRD3.1.8.4-1
Each sensor having movable components shall not exceed an uncompensated momentum contribution of ±0.5 N-m-sec per axis.

IRD3.1.8.4-2
The uncompensated momentum contribution of the sensor shall be documented in the sensor-spacecraft ICD.

3.1.9 Sensor Disturbance Allocations

3.1.9.1 Periodic Disturbance Torque Limits

IRD3.1.9.1-1
A Fourier decomposition of the sum of the sensor-induced periodic disturbance torques shall be used to produce the corresponding magnitude spectrum.

IRD3.1.9.1-2
For each sensor, the periodic disturbance torque magnitude shall be in the acceptance region of Figure 2 for all frequencies. The transition points for Figure 2 are shown in Table 3.

IRD3.1.9.1-3
The sensor-spacecraft ICD shall specify the measured/predicted disturbance torque contributions to the spacecraft, if any.

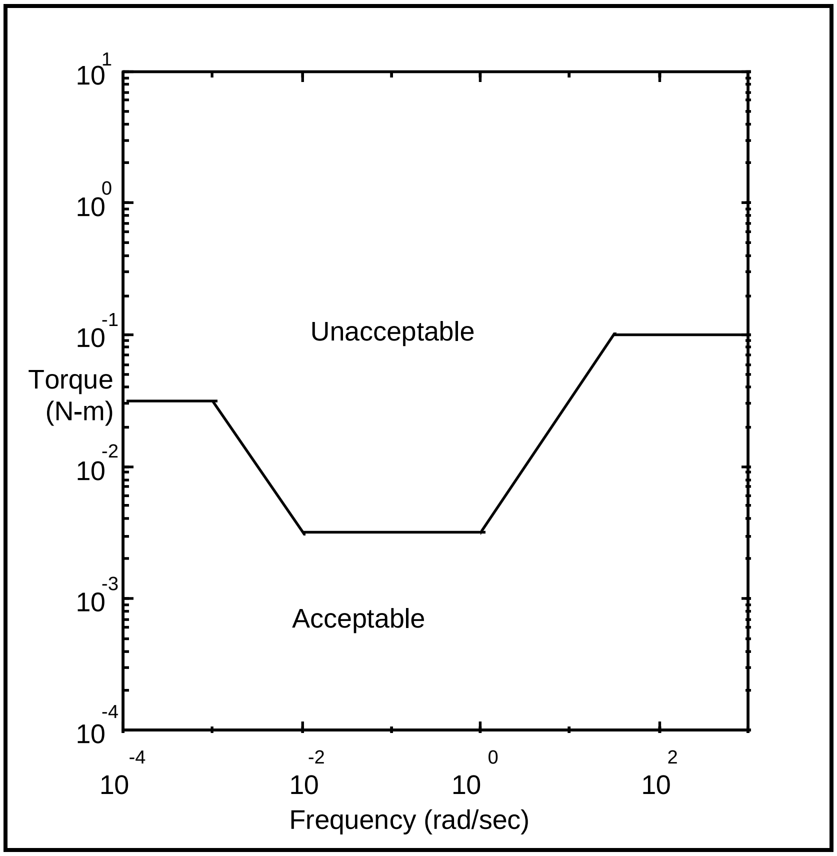

Figure 2. Allowable Transmitted Torque

TABLE 3. ALLOWABLE TRANSMITTED TORQUE TRANSITION POINTS

Frequency (rad/sec)	Torque (N-m)
< 0.001	0.03
0.01	0.003
1.0	0.003
> 33.3	0.1

3.1.9.2 Constant Disturbance Torque Limits

IRD3.1.9.2-1
Sensor-induced constant disturbances of the same polarity, separated by more than 200 seconds, shall not exceed the torque limit defined in Figure 3 if the duration of application is in excess of 10 seconds. For constant torques of 10 seconds duration or less, the impulse limit is 0.04 N-m-sec. For constant torques of 400 seconds duration or more, the torque limit is maintained at the 400 second limit shown in Figure 3.

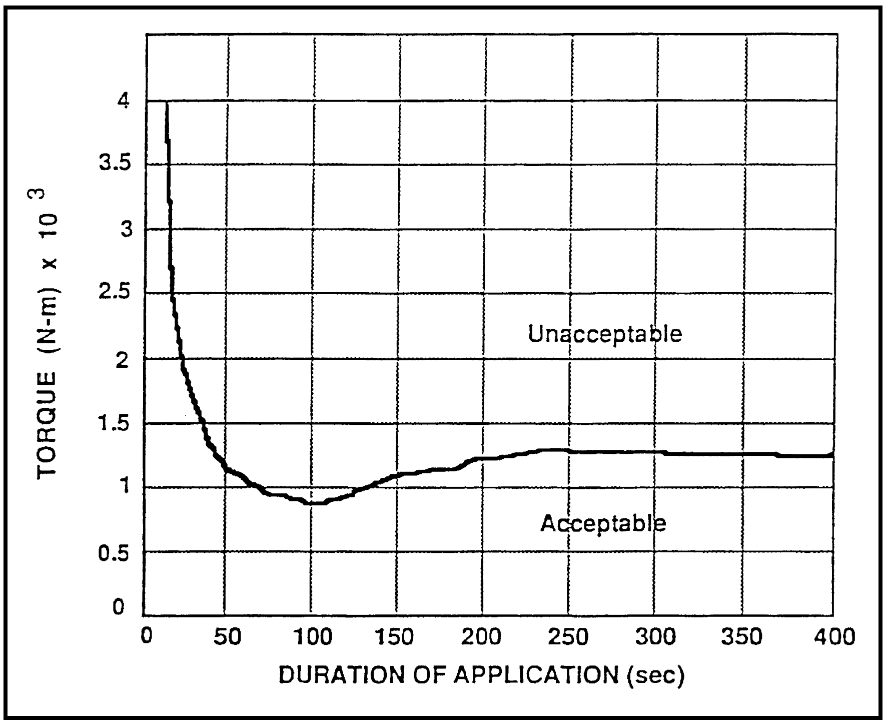

Figure 3. Constant Torque vs. Duration of Application

3.1.9.3 Torque Profile Documentation

IRD3.1.9.3-1
The actual sensor torque versus time profile shall be documented in the sensor-spacecraft ICD.

3.1.9.4 Thrust Direction Definition

IRD3.1.9.4-1
The magnitude and direction of net thrust resulting from the expulsion of expendables by the sensor shall be documented in the sensor-spacecraft ICD.

3.1.10 Magnetics

IRD3.1.10-1
Avoid using large quantities of magnetic materials where possible. If magnets are inherent to the sensor design, early estimates of magnetic fields and residual magnetic dipole moments shall be provided to the integrating contractor.

IRD3.1.10-2
If the uncompensated magnetic moment of a sensor exceeds 0.0005 ampere-turn-meter-square (0.5 pole-cm) per kilogram of mass, a full magnetic survey shall be conducted and corrective action taken by the sensor contractor. This allocation is done on a per box level and may not be reallocated between sensor boxes.

3.1.11 Access

3.1.11.1 Access Identification

IRD3.1.11.1-1
Access requirements shall be documented in the sensor-spacecraft ICD.

3.1.11.2 General Access

IRD3.1.11.2-1
All items to be installed, removed, or replaced at the satellite level shall be accessible without disassembly of the unit.

3.1.12 Handling Fixtures

IRD3.1.12-1
The sensor contractor shall provide proof tested handling fixtures for each component.

IRD3.1.12-2
Handling fixtures shall be designed to 5 times limit load for ultimate and 3 times limit load for yield.

IRD3.1.12-3
Handling fixtures shall be tested to 2 times working load.

3.1.13 Mounting Orientation

IRD3.1.13-1
Sensors shall be capable of being mounted to the spacecraft with the spacecraft in the horizontal or vertical position.

3.1.14 Sensor to Spacecraft Integration and Test Mounting

IRD3.1.14-1
Sensors shall be capable of being mounted or removed without removal of other sensors or components.

3.1.15 Non-Flight Equipment

IRD3.1.15-1
All non-flight items to be installed and/or removed prior to flight shall be identified in the sensor-spacecraft ICD.

3.2 Thermal Requirements

3.2.1 Sensor Thermal Design

IRD3.2.1-1
All interface requirements specified in Section 3.2 shall be met at the mechanical interface.

The sensor thermal design shall provide for:

IRD3.2.1-2
a. Maintaining the sensor within operating and survival temperature limits,

IRD3.2.1-3
b. Maintaining the sensor at the minimum turn-on temperature via survival power,

IRD3.2.1-4
c. Thermal decoupling of the sensor from the spacecraft.

IRD3.2.1-5
d. The sensor shall take no more than 60 minutes to return to a normal steady-state temperature from a safehold mode assuming no anomalies.

3.2.2 Thermal Isolation to Spacecraft

IRD3.2.2-1
The spacecraft shall not be used primarily as a heat source or sink (i.e., the sensor design should maximize thermal isolation).

IRD3.2.2-2
Sensor components shall be designed to maintain the sensor within its allowable temperature limits.

IRD3.2.2-3
The thermal control units shall be mounted on the sensor, where possible, or insulated in order to minimize thermal load to the spacecraft.

3.2.3 Heat Transfer

3.2.3.1 Heat Transfer to Spacecraft

The sensors should be as thermally isolated as possible to minimize heat transfer to the spacecraft and other adjacent sensors. To maintain flexibility in the placement of sensors on the spacecraft, it is necessary to limit both the total heat transfer and the heat flux. The heat transfer is primarily via conduction through the baseplate. However, radiative limits are also required to facilitate the placement of sensor units near other sensors.

IRD3.2.3.1-1
The conducted heat transfer between the sensor baseplate and the spacecraft shall not exceed 10 watts maximum.

IRD3.2.3.1-2
Deleted

IRD3.2.3.1-3
The radiative heat transfer from the sensor to any adjacent unit shall not exceed 2 watts maximum, excluding radiators.

3.2.3.2 Radiation

IRD3.2.3.2-1
Incident radiation between the spacecraft and a sensor on any given surface shall be minimized.

IRD3.2.3.2-2
The integrating contractor shall provide the radiative loads to the sensor.

IRD3.2.3.2-3
The environmental fluxes, as shown in Table 4 below, shall add solar, albedo and earth infrared (IR) hot fluxes for the hot case analysis and cold fluxes for the cold case analysis.

TABLE 4. WORSE-CASE HOT AND COLD ENVIRONMENTS

	Hot Case		Cold Case	
	BTU/hr-ft^2	W/m^2	BTU/hr-ft^2	W/m^2
Solar Radiation	444	1400	415	1308
Albedo	172	542	86	271
Earth IR Radiation	83	262	60	189

3.2.4 Temperature Ranges

3.2.4.1 Spacecraft Temperature Range

IRD3.2.4.1-1
For planning and preliminary design purposes, the temperature of the interface between the payload and the spacecraft shall be initially assumed to range from:

a. -10 °C to +40 °C during normal operations
b. -20 °C to +50 °C during survival modes

3.2.4.2 Sensor Temperature Range

IRD3.2.4.2-1
Temperature limits for sensor components during ground test and orbital operations shall be documented in the sensor-spacecraft ICD.

IRD3.2.4.2-2
Operating, non-operating, survival and turn-on temperature requirements shall be included.

3.2.4.3 Thermal Uncertainty Margins

IRD3.2.4.3-1
Thermal uncertainty margins used during the design and validation shall be applied to determine acceptance ranges per MIL-STD-1540C. If heaters are employed, a 25% heater control authority can be used in place of the thermal uncertainty margin.

IRD3.2.4.3-2
Protoqualification ranges shall be calculated by adding an additional margin of ±5 °C.

3.2.5 Temperature Monitoring

3.2.5.1 Mechanical Mounting Interface Temperature Monitoring

IRD3.2.5.1-1
The spacecraft shall monitor and report in the spacecraft telemetry the temperature of the spacecraft at the sensor mechanical mounting interfaces.

3.2.5.2 Sensor Temperature Monitoring

IRD3.2.5.2-1
All critical sensor temperatures shall be measured and reported in the health and status telemetry data.

3.2.5.3 Temperature Sensor Locations

IRD3.2.5.3-1
The location of all sensor and mounting interface temperature sensors shall be documented in the sensor-spacecraft ICD.

3.2.6 Thermal Control Design

3.2.6.1 Survival Heater Design

IRD3.2.6.1-1
Sensors shall use survival heaters to maintain temperature at the safe turn-on level.

IRD3.2.6.1-2
Operational heaters inside the sensor shall be controlled by the sensor when sensor operational power is present.

IRD3.2.6.1-3
Electrical power for survival heaters shall be provided by the spacecraft and accommodate at least two strings, a primary and secondary string, of sensor survival heaters.

IRD3.2.6.1-4
Survival heater circuits shall not exceed 1 ampere per string. More than one string may be required.

IRD3.2.6.1-5
Survival heater circuits shall be provided directly to thermostatically controlled heaters on the sensor side.

IRD3.2.6.1-6
Sensor survival heaters shall be capable of operation when the sensor operational power is not present, but the survival heater power voltage is present.

IRD3.2.6.1-7
The interface shall also have the capability to accommodate up to five passive analog temperature measurements per interface.

IRD3.2.6.1-8
Each passive analog temperature measurement shall be redundant.

IRD3.2.6.1-9
These analog lines are separate and in addition to any state of health (SOH) input being transmitted over the serial data bus interface and are intended to provide insight during periods when the sensor Operational Power is not present; therefore, excitation of passive analog temperature measurement devices shall be provided by the spacecraft.

IRD3.2.6.1-10
The electrical connection between these passive analog temperature measurement devices and any sensor electronics, chassis, or ground shall be isolated to at least 1 megohm resistance.

IRD3.2.6.1-11
Electrical connections to these dedicated passive analog temperature measurement devices shall be made only through an electrical connector interfacing the sensor with the spacecraft.

IRD3.2.6.1-12
The spacecraft shall accommodate passive analog measurements derived from the spacecraft excited measurement devices with characteristics as follows:

Current Source Excitation: 1 mA ± 5%
Input Impedance Range: 100 ohms to 10 kiloohms

IRD3.2.6.1-13
When the sensor operational power is present, the sensor shall be responsible for distributing power and controlling the operational heaters inside the sensor.

IRD3.2.6.1-14
When the sensor operational power is not present, but Survival Heater Power is present, the survival heaters shall nominally be controlled from within the sensor using thermostats internal to the sensor.

IRD3.2.6.1-15
Redundant thermostats shall be used.

3.2.6.2 Thermal Control Hardware

IRD3.2.6.2-1
Thermal control hardware shall be documented in the sensor-spacecraft ICD. The responsibility for providing the thermal control hardware is defined in Table 5

TABLE 5. THERMAL CONTROL HARDWARE RESPONSIBILITY

Hardware	Responsibility
Survival Heaters	Sensor Provider
Sensor, Thermal Control Hardware, including blankets, louvers, and heat pipes	Sensor Provider
Thermal insulation Blankets to Interface between the Sensor Thermal Blankets and the Spacecraft Thermal Blankets	Spacecraft Provider

3.2.6.3 Multilayer Insulation

IRD3.2.6.3-1
Multilayer Insulation (MLI) used in thermal control design shall have the following provisions: venting, interfacing with spacecraft thermal control surfaces, and electrical grounding to prevent Electro Static Discharge (ESD).

3.2.6.4 Other Considerations

IRD3.2.6.4-1
Thermal control surfaces shall be cleanable to visibly clean or better.

IRD3.2.6.4-2
Any sealed or closed system such as heat pipes, thermal control enclosures or fluid loops shall be analyzed to demonstrate that a safety hazard does not exist.

3.2.6.5 Ambient Tests

IRD3.2.6.5-1
The sensor shall be capable of being functionally tested in an ambient environment, to the maximum extent possible.

IRD3.2.6.5-2
Constraints related to this testing capability shall be documented in the sensor-spacecraft ICD.

3.3 Electrical Power Requirements

3.3.1 Electrical Interfaces

IRD3.3.1-1
The electrical interfaces (Figure 4) shall include the following:

a. Operational Power Interface
b. Survival Heater Power Bus
c. Pulse Command Interface
d. Command and Telemetry Data Bus
e. Grounding Interface
f. Test Point Interface

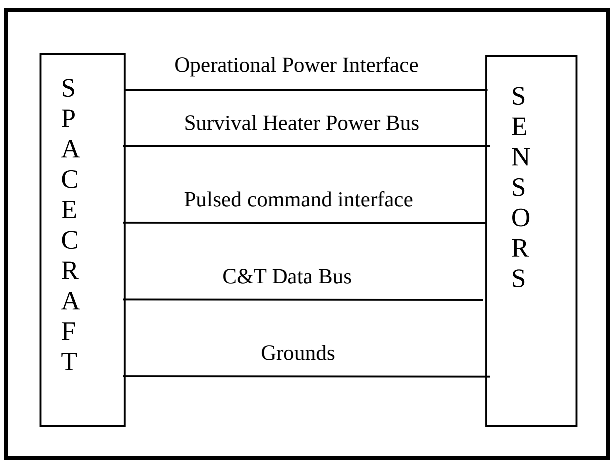

Figure 4. Spacecraft-Sensor Electrical Interfaces

3.3.2 Electrical Voltage

3.3.2.1 Primary Sensor Voltage

IRD3.3.2.1-1
The electrical power supplied at the sensor interface shall be regulated 28 $\pm$ 6.0 Vdc.

3.3.2.2 Voltage Ripple

IRD3.3.2.2-1
The power source-generated and load-induced ripple, including repetitive spikes, shall not exceed 1.0 volt peak-to-peak as measured over the bandwidth of 30 Hz to 1.0 kHz, and 0.5 volt peak-to-peak from 1.0 kHz to 10 MHz when the power system is delivering the maximum rated current into loads.

3.3.2.3 Reflected Ripple

IRD3.3.2.3-1
Loads shall not produce reflected ripple greater than the limits of MIL-STD-461D, CE101 (Figure CE101-4, curve #1) and CE102. CE101 and CE102 maximum levels apply to loads that are 15 amps/450 watts and greater.

IRD3.3.2.3-2
The maximum emissions shall be reduced by 1 dB microamp for each dB amperes reduction in the average current. Measurements will be made at the sensor interface.

3.3.2.4 Transients

IRD3.3.2.4-1
Positive and negative voltage surges shall decay to within steady state limits in less than 5 and 100 milliseconds, respectively.

IRD3.3.2.4-2
All spacecraft and sensor components shall remain undamaged when subjected to step changes of the input voltage from 0% to 140% and from 120% to 0% of the nominal load voltage (28 volts). The step changes, exclusive of spikes, are the instantaneous surge amplitudes produced by load switching and the clearing of faults on the space-vehicle power bus.

3.3.2.5 Undervoltage Protection

IRD3.3.2.5-1
The spacecraft shall be able to remove bus power to all sensors if the bus voltage drop below 22 volts.

IRD3.3.2.5-2
Control heaters shall also be turned off during these occurrences. This does not apply to survival heaters.

3.3.2.6 Spacecraft Power Bus Impedance

IRD3.3.2.6-1
The spacecraft bus impedance at the interface looking back into the source shall be less than 100 milliohms resistive and 10 micro-henries inductive.

3.3.3 Prime Electrical Power

IRD3.3.3-1
Two types of power shall be supplied to each sensor: survival heater power and operational power.

IRD3.3.3-2
The sensor contractor, in the sensor-spacecraft ICD, shall document the nominal and maximum load currents for each power interface line.

3.3.3.1 Survival Heater Power

IRD3.3.3.1-1
Direct bus connection shall be through a spacecraft fuse for 50 watt heater maximum.

IRD3.3.3.1-2
To ensure fault-tolerant usage of this bus, the sensor shall have two thermostats arranged to ensure circuit is single fault tolerant.

3.3.3.2 Operational Power

Two supply service types shall be provided:

IRD3.3.3.2-1

a. A 0-5 ampere steady-state power connection is made through a fuse and relay switch in the spacecraft. Inrush current at initial power application to the sensor suite is no more than 10-times the maximum steady-state current, defined as when all sensor are operational. The inrush current should decay to within 10% of the maximum steady-state current in less than 3 milliseconds. Inrush current at equipment commanded turn-on is 4-times the maximum steady-state current and should decay to less than 2% of the maximum steady-state current in less than 100 milliseconds.

IRD3.3.3.2-2

b. A 5-20 ampere steady-state power connection is made through a fuse and relay switch in the spacecraft. Inrush current at initial power application to the sensor suite is no more than 4-times the maximum steady-state current. This inrush current should decay to within 10% of the maximum steady-state current in less than 3 milliseconds. Inrush current at equipment commanded turn-on is no more than 4-times the maximum steady-state current and should decay to within 10% of the maximum steady-state current in less than 20 milliseconds.

3.3.4 Grounds, Returns, and References

3.3.4.1 Prime Power Grounding

IRD3.3.4.1-1
Prime power and return shall be isolated from sensor chassis and structure by more than 1 Meg ohm. Prime power return is tied to vehicle structure at the vehicle single point ground.

3.3.4.2 Power Leads and Signal Returns

IRD3.3.4.2-1
Command returns shall be referenced to prime power return.

IRD3.3.4.2-2
Telemetry returns shall be referenced to sensor chassis.

3.3.4.3 Power Harnesses

3.3.4.3.1 ElectroMagnetic Interference/Compatibility (EMI/EMC) Considerations.

IRD3.3.4.3.1-1
Data and telemetry signals shall be segregated and routed from any power circuitry via a separate connector.

IRD3.3.4.3.1-2
The characteristics of the power bus with respect to power returns shall be as specified in the Electromagnetic Compatibility (EMC) Control Plan.

3.3.4.3.2 Fault Isolation

IRD3.3.4.3.2-1
Prime power fault isolation shall be included only on the spacecraft side of the interface.

3.3.4.3.3 Electrical Connectors

IRD3.3.4.3.3-1
The electrical connectors shall be either MIL-STD-975 or MIL-STD-1547 approved versions of MIL-C-38999 or MIL-C-24308 connectors.

IRD3.3.4.3.3-2
Primary and redundant connectors shall be differentiated by clearly marking all units and cables.

IRD3.3.4.3.3-3
The sensor contractor shall provide two sets of interface mating connectors to the integrating contractor for each sensor.

3.3.4.3.4 Wiring

IRD3.3.4.3.4-1
All interface wiring shall comply with MIL-STD-975 and MIL-STD-1547A.

3.3.4.3.5 Power Cabling

IRD3.3.4.3.5-1
Power interface wiring, supply and return wires shall be twisted to reduce electro-magnetic contribution.

3.3.4.3.6 Signal Cabling

IRD3.3.4.3.6-1
All signal interface wiring shall be twisted shielded pairs with the shield being terminated circumferentially on the connector backshells.

3.3.4.3.7 Electromagnetic Interference Filtering of Satellite Power

IRD3.3.4.3.7-1
The sensor shall have EMI input filters installed on the sensor side of the power interface. This does not apply to the survival heater circuits which are controlled on the spacecraft side of the interface.

IRD3.3.4.3.7-2
The filters shall provide both common-mode and differential-mode filtering capable of meeting EMC requirements per MIL-STD-1541A.

IRD3.3.4.3.7-3
The filters shall be designed to withstand and suppress electrical transients.

3.3.5 Test Points Location

IRD3.3.5-1
Test points shall be accessible when the unit is mounted on the spacecraft.

IRD3.3.5-2
Test points shall be tolerant to short circuits and a polarity reversal of power.

IRD3.3.5-3
All test points shall be terminated in their connector cap.

3.3.6 Spacecraft/Sensor Interface Simulator

IRD3.3.6-1
The sensor contractor shall provide a simulator to the integrating contractor for initial interface testing.

IRD3.3.6-2
The simulator shall, as a minimum, have the same mechanical characteristics (geometry, mass, c.g., mounting holes, connectors, etc.) as the real sensor-to-spacecraft interfaces.

IRD3.3.6-3
In addition, an electrical interface simulator, having corresponding connectors and pin assignment, shall be provided.

3.4 Command and Data Handling (C&DH) Requirements

3.4.1 Functional and Performance Description

The spacecraft C&DH subsystem will perform the following functions:

- Collect all mission data, satellite health and status data, and receive and process ground commands and memory loads.
- Format and process the collected data for both real-time transmission and onboard storage. Transfer mission data to the communications subsystem for real-time and stored data transmission.
- Maintain a data base of allowable limits for the satellite critical parameters to establish nominal and maximum/minimum values for monitoring status and health.
- Monitor all the satellite health and status telemetry and issue commands to the subsystems for appropriate action.
- Receives demodulated uplink commands and memory loads from the Communication subsystem, and transmits them to the appropriate destination.
- Provide time marks, time codes, and satellite ephemeris data to each of the sensors.
- Initiate power down or self-recover mode if required to maintain spacecraft power and ground communication.

3.4.2 Satellite Modes

IRD3.4.2-1
The satellite shall implement the following common modes as a minimum:
- OFF Mode
- OPERATIONAL Mode
- SAFE HOLD Mode
- AUTONOMOUS Mode
- DIAGNOSTICS Mode

IRD3.4.2-2
The satellite shall be designed to handle any mode sequence without damage to the satellite.

3.4.2.1 OFF Mode

IRD3.4.2.1-1
In the OFF Mode, no power shall be supplied to the satellite.

3.4.2.2 OPERATIONAL Mode

IRD3.4.2.2-1
The sensors shall have one or more OPERATIONAL Modes for collecting data as defined in the applicable sensor-spacecraft ICDs.

IRD3.4.2.2-2
The sensor shall be fully operational in this mode.

3.4.2.3 SAFE HOLD Mode

IRD3.4.2.3-1
In the case of an anomalous satellite event, it may be necessary to enter the SAFE HOLD Mode to protect the satellite. The C&DH shall be capable of re-configuring the satellite to a safe condition.

IRD3.4.2.3-2
Ground intervention shall be required to return to the OPERATIONAL Mode.

3.4.2.4 AUTONOMOUS Mode

IRD3.4.2.4-1
In the AUTONOMOUS mode, the satellite shall be capable of operating for a minimum of 21 days with an objective of 60 days without additional commands from the ground.

3.4.2.5 DIAGNOSTICS Mode

IRD3.4.2.5-1
DIAGNOSTIC mode shall include housekeeping, troubleshooting, testing, and software updates.

3.4.2.6 Mode Documentation

IRD3.4.2.6-1
Additional detail on the various satellite modes shall be defined in the sensor-spacecraft ICDs.

IRD3.4.2.6-2
SAFE HOLD re-configuration commands shall be defined in the sensor-spacecraft ICDs.

3.4.3 General Electrical Interface Requirements

Unless specified otherwise, the following requirements apply to all C&DH electrical interfaces.

3.4.3.1 Interface Conductors

IRD3.4.3.1-1
All C&DH subsystem interface conductors and circuits shall be compliant with the EMI/EMC specifications in Section 3.7.4.6.

3.4.3.2 Interface Circuitry Isolation

IRD3.4.3.2-1
The sensor shall maintain electrical isolation of greater than 100 kilo-ohms between the primary and redundant interface circuitry within the sensor front end.

3.4.3.3 Interface Fault Tolerance

IRD3.4.3.3-1
The sensor and spacecraft bus shall be tolerant of a single fault occurring in a signal interface circuit on either side of the interface.

3.4.4 Command and Telemetry (C&T) Data Bus Requirements

3.4.4.1 Bus Functions

The C&T Data Bus requirements listed below can be satisfied by a combination of shared buses (such as MIL-STD-1553B and AS-1773 or equivalent), or, if necessary, a dedicated connection (such as EIA RS-422). The choice of a particular data bus for a sensor will depend on the data rate for that sensor. [Note: the use of a dedicated connection may require deviation from the requirements listed below.] Data rates for the sensors and spacecraft subsystems may require multiple data buses

IRD3.4.4.1-1
The C&T Data bus (Figure 5) shall be used to provide the following functions:

a. Spacecraft to sensor transfers consisting of:
   - real time commands
   - stored commands
   - memory loads
   - time code data
   - ancillary data
   - satellite ephemeris data

b. Sensor/remote terminal to spacecraft transfers consisting of:
   - sensor health and status telemetry
   - sensor diagnostic data
   - mission data
   - time code data (GPSOS only)
   - satellite ephemeris data (GPSOS only)

3.4.4.2 Bus Type

IRD3.4.4.2-1
Each C&T data bus shall be dual standby redundant, and shall fully comply with the requirements of SAE AS-1773 or Mil-STD-1553B, Notice 2, all sections.

3.4.4.2.1 Data Bus Sampling Rate

IRD3.4.4.2.1-1
The combined rate at which the spacecraft transmits commands, samples telemetry and collects mission data to/from the sensors, the maximum duration of a data transfer cycle and the minimum time gap between transfer cycles shall comply with the interfacing bus specification.

IRD3.4.4.2.1-2
The bus sampling rates for each sensor shall meet the sensor functional requirements as identified in the Sensor Requirements Document.

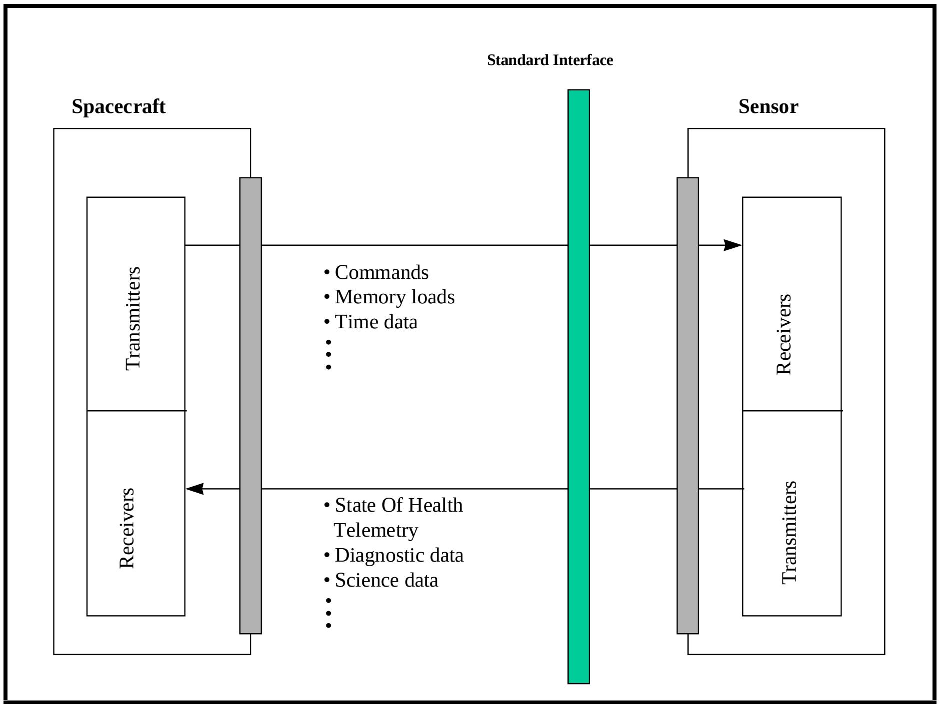

Figure 5. Data Transfer Interface

3.4.4.2.2 Data Packetization

IRD3.4.4.2.2-1
All telemetry and mission data to be transferred to the spacecraft C&DH via the data bus shall be packetized using the CCSDS Path Protocol Data Unit format defined in CCSDS 701.0-B-2.

3.4.4.3 Bus Configuration

IRD3.4.4.3-1
The spacecraft C&DH shall perform the Bus Controller (BC) function for the data bus to send data to and collect data from the sensors (Figure 6).

IRD3.4.4.3-2
The sensors shall interface with one of the data buses to exchange data with the C&DH.

IRD3.4.4.3-3
The RT shall physically reside in the sensor unless special arrangements are made with the spacecraft contractor and documented in the ICD.

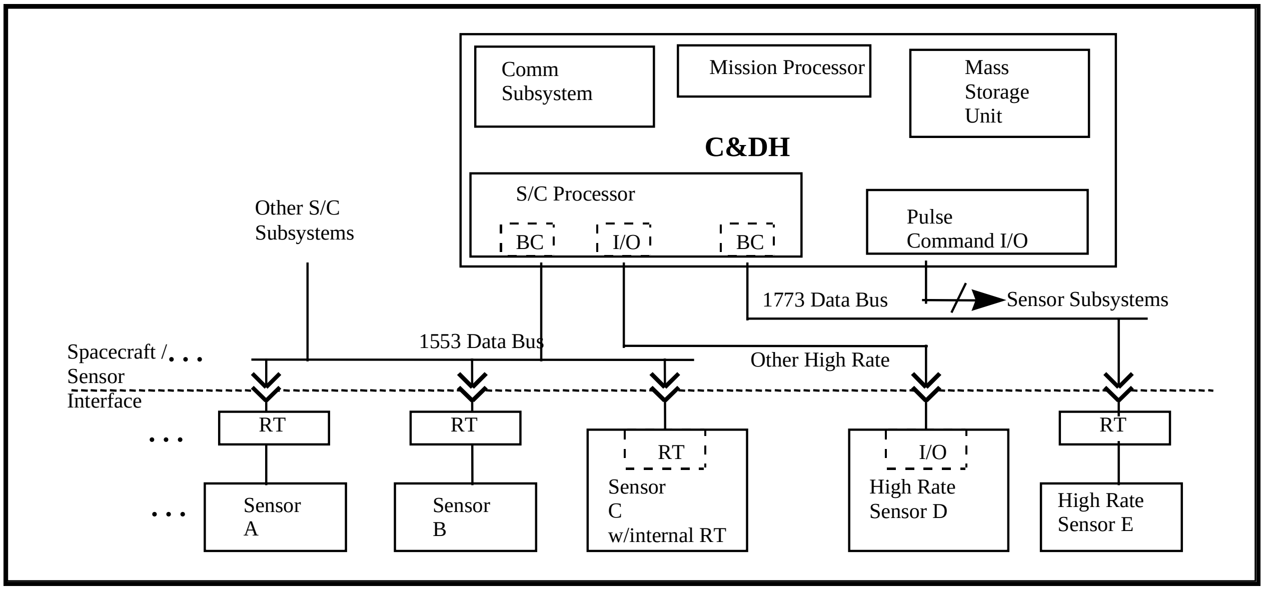

Figure 6. Notional Command and Data Handling Interface Topology

3.4.4.4 General Bus Characteristics

3.4.4.4.1 Electrical/Optical Interface

The spacecraft contractor shall define the bus implementation characteristics not defined in AS-1773. These characteristics will be documented in the sensor-spacecraft ICD.

IRD3.4.4.4.1-1
The electrical or fiberoptic interface of the C&T Data buses shall comply with the requirements of MIL-STD-1553B, Notice 2, all sections or other specification designated in the spacecraft-sensor ICD.

IRD3.4.4.4.1-2
Each electrical/optical interface between the sensor and the data bus shall be dual redundant.

IRD3.4.4.4.1-3
Each functionally distinct RT shall be dual redundant.

IRD3.4.4.4.1-4
Each bus interface shall be individually electrically/optically coupled to both the primary and the redundant data buses.

IRD3.4.4.4.1-5
No single failure in the data bus electrical/optical interface circuit on either the sensor side of the interface or the spacecraft data bus side of the interface shall cause the sensor to lose the capability to communicate with either the primary or the redundant data buses.

IRD3.4.4.4.1-6
For those sensors requiring the AS-1773 bus connection, the sensor contractor shall be responsible for providing an RT which fully compiles with the implementation parameters defined in the sensor-spacecraft ICD and with SAE AS-1773.

3.4.4.4.2 Data Bus Monitoring

IRD3.4.4.4.2-1
The Bus Controller shall have the capability to monitor the bus status and switch to a redundant bus so that no sensor or single data bus failure shall prevent the Bus Controller from maintaining data flow over the Data Bus.

3.4.4.5 Sensor Commands and Memory Load

3.4.4.5.1 Command Types

IRD3.4.4.5.1-1
The spacecraft shall deliver the following data to the specified sensor RT-receive sub-addresses by conducting single BC to RT Transfers or single RT to RT Transfers (from a spacecraft RT to an sensor RT).

IRD3.4.4.5.1-2
The C&DH shall have the capability to issue on/off pulse commands to the sensors via a redundant hardwire interface.

IRD3.4.4.5.1-3
The sensor shall be capable of accepting pulse and serial commands, via a redundant hardwire interface, with the characteristics specified:

<u>Pulse Command</u>

a. Logic 0	TBS
b. Logic 1	TBS
c. Load Capacitance	TBS
d. Pulse Width	TBS
e. Voltage Rise Time	TBS
f. Voltage Fall Time	TBS
g. Noise Immunity	TBS
h. Inductive Spike Suppression	TBS

<u>Serial Command *(TBR)*</u>

IRD3.4.4.5.1-4
The serial command input shall consist of Non-return to zero (NRZ) data, clock and envelope signals.

IRD3.4.4.5.1-5
The RT to Sensor serial command transfer shall consist of a three-wire interface. Characteristics of the interface are *TBR*.

3.4.4.5.2 Packetization for Commands and Memory Loads

IRD3.4.4.5.2-1
Unless otherwise specified, all commands and memory loads delivered to the sensor shall be formatted in accordance with the CCSDS Telecommand packet defined in CCSDS 203.0-B-1 and communicated over the data bus.

3.4.4.5.3 Documentation

IRD3.4.4.5.3-1
All sensor commands and memory load packet descriptions shall be documented in the sensor-spacecraft ICD.

3.4.4.5.4 Critical Commands

IRD3.4.4.5.4-1
Initiation of critical or hazardous functions shall use, as a minimum, separate enable and execute commands to the commanded unit, to prevent inadvertent execution of critical commands.

IRD3.4.4.5.4-2
Enables shall be disabled by the sensor after the critical function is commanded or after (TBD) seconds.

IRD3.4.4.5.4-3
Critical or hazardous functions shall be identified in the sensor-spacecraft ICD.

3.4.4.5.4.1 Squib Driver Function

IRD3.4.4.5.4.1-1
The C&DH subsystem shall command the sensor squib functions through a dedicated interface. Sensor squib functions are defined as signals to fire pyrotechnic devices such as tie-downs, bolt cutters, pin-pullers, separation nuts and squib valves.

3.4.4.6 Synchronization and Time Code Data

IRD3.4.4.6-1
The spacecraft shall provide synchronization and time code data signals to the sensors via the Data Bus.

IRD3.4.4.6-2
The format of the vehicle time code words shall be based on the universal time coordinated (UTC) time representation.

IRD3.4.4.6-3
On-board absolute correlation of time shall be 1 millisecond or better with a correlation to 1 microsecond as a goal.

IRD3.4.4.6-4
Time representation shall be transmitted over the data bus once per second.

3.4.4.7 Health and Status Telemetry and Diagnostic Data

IRD3.4.4.7-1
The spacecraft and sensors shall provide housekeeping telemetry in one or more telemetry formats associated with any mode.

IRD3.4.4.7-2
The details of the telemetry formats shall be documented in the sensor-spacecraft ICD.

3.4.4.7.1 Telemetry Data Overview

IRD3.4.4.7.1-1
The spacecraft computer within the C&DH shall hold in memory the Health and Status Telemetry formats and be capable of accepting telemetry formats from the ground.

IRD3.4.4.7.1-2
The capability to select, by command, one of the fixed formats and dwell modes shall be provided.

IRD3.4.4.7.1-3
The spacecraft shall collect the selected format's telemetry data by conducting a sequence of bus data transfers. The collected data includes the following types: unconditioned analog, conditioned analog, unconditioned bilevel, conditioned bilevel, and serial digital.

IRD3.4.4.7.1-4
All critical telemetry channels shall be redundant *(TBR)*.

3.4.4.7.2 Health and Status Telemetry Data

IRD3.4.4.7.2-1
Sensor health and status telemetry data shall include housekeeping data required for sensor status and health monitoring at the Ground Control Center. Sensor health and status telemetry includes:

- Sensor mode and configuration
- Sensor temperatures
- Sensor power supply current and voltage
- Relay status, scan mirror rotation and other rotating mechanism rates
- Other telemetry data required to support sensor performance evaluation

3.4.4.7.3 Telemetry Diagnostic Data

IRD3.4.4.7.3-1
During sensor anomaly resolution, the spacecraft C&DH shall have the capability to dwell on particular telemetry measurands within the selected telemetry format in support of ground diagnostic investigation of the sensor anomaly. The spacecraft C&DH will accommodate the specified dwell mode by accommodating a sensor telemetry sample rate of up to 10 samples/sec.

IRD3.4.4.7.3-2
Dwell capability shall be a ground initiated process.

IRD3.4.4.7.3-3
The sensor shall provide sufficient telemetry to diagnose failures to the lowest switchable level.

3.5 Contamination

IRD3.5-1
A contamination control program shall be developed and implemented as part of the sensor to spacecraft interface.

IRD3.5-2
The applicable contamination interface requirements shall be included in the sensor-spacecraft ICD.

IRD3.5-3
A system level contamination plan shall be developed by the integrating contractor.

IRD3.5-4
Sensor and other sensor contamination and cleanliness requirements shall be considered along with the spacecraft requirements and the resulting contamination budget included in the system level contamination plan and the sensor-spacecraft ICD.

3.5.1 Contamination Control Requirements

IRD3.5.1-1
The integrating and sensor contractors shall perform independent contamination analyses to identify, locate and size components sensitive to contamination and assess, calculate or measure the maximum allowable particulate and molecular film (nonvolatile residue, or NVR) contamination consistent with top level mission performance and lifetime specifications.

IRD3.5.1-2
The resultant contamination requirements shall be documented in the sensor-spacecraft ICD

3.5.2 Sensor Sources of Contamination

IRD3.5.2-1
Sensor contractors shall identify and characterize all sources of contamination, particulate and molecular, that can be emitted from the sensor.

IRD3.5.2-2
At a minimum, the characterization of particulate and molecular contamination shall include the material name, the amount, the emission rate, and its location.

IRD3.5.2-3
The extent to which molecular outgassing products have access to exterior surfaces shall also be considered.

IRD3.5.2-4
Data from the ASTM E-595 test for percent total mass loss (%TML) and percent collected volatile condensable material (%CVCM) shall be used.

IRD3.5.2-5
The outgassing chemical species and any tendency to photodeposit in an ultraviolet (UV) or energetic particle radiation environment shall be identified and quantified.

IRD3.5.2-6
Material outgassing is an issue that shall be coordinated between the sensor developer and the satellite contractor.

IRD3.5.2-7
Materials and coatings known to flake or outgas, such as cadmium and zinc plating, shall not be used. Materials with the following properties are recommended: Total Mass Loss (TML) less than 1.0%; production of Collected Volatile Condensable Material (CVCM) less than 0.1% when tested under conditions of ASTM E595-93 or equivalent. Composite materials are an exception.

3.5.3 Sensor Venting

IRD3.5.3-1
Sensor contractors shall define the location, size, path and operation time of vents in the sensors.

IRD3.5.3-2
This information shall be defined in the sensor-spacecraft ICD.

3.5.4 Sensor Purge Requirements

IRD3.5.4-1
Spacecraft and sensor purge requirements, including type of purge gas (e.g. dry air or GN_2), flow rate, gas purity specifications, filtration and desiccant requirements, and the acceptable limits for purge interruption, shall be provided and documented in the sensor-spacecraft ICD.

3.5.5 Sensor Inspection and Cleaning During I&T

IRD3.5.5-1
Inspections and cleaning by the spacecraft and sensor contractors during Integration and Test (I&T) and access requirements shall be coordinated and defined in the sensor-spacecraft ICD.

3.5.6 Spacecraft Contractor Supplied Analysis Inputs

IRD3.5.6-1
As part of the contamination control analysis and sensor-spacecraft ICD development, the integrating contractor shall perform a particulate and molecular plume flowfield analysis for all spacecraft thrusters.

IRD3.5.6-2
The molecular and particulate analyses shall provide molecular flux quantities into instrument apertures at the time they are opened on-orbit and total predicted depositions at end-of-life onto all sensitive surfaces.

IRD3.5.6-3
The plume analyses shall include the identity and quantity of each chemical species emitted and provide sufficient dynamic information to determine the final deposition amount on all sensitive surfaces.

IRD3.5.6-4
Margin shall be given for an additional contribution from the payload fairing and launch vehicle thrusters fired after the payload fairing has been jettisoned.

IRD3.5.6-5
The integrating contractor shall also identify and characterize all sources of contamination, particulate or molecular, that can be emitted from the spacecraft.

IRD3.5.6-6
The spacecraft design shall incorporate deployment systems that minimize particle release or completely eliminate the escape of actuating materials.

IRD3.5.6-7
At a minimum, the characterization of molecular and particulate contamination shall include the material name, the amount, the emission rate, and its location.

IRD3.5.6-8
Data from the ASTM E-595 test for percent total mass loss (%TML) and percent collected volatile condensable material (%CVCM) shall be used.

IRD3.5.6-9
The outgassing chemical species and any tendency to photodeposit in a UV or energetic particle radiation environment shall be identified and quantified.

3.5.7 Atomic Oxygen Contamination

IRD3.5.7-1
The integrating and sensor contractors shall consider the effects of atomic oxygen in the space environment. spacecraft and sensor materials selection should minimize the generation of particulate and molecular film contamination via interaction with atomic oxygen.

3.5.8 Facility Environmental Requirements

IRD3.5.8-1
The integrating and sensor contractors shall describe the required integration and test environments using the definitions of FED-STD-209E or ISO/TC-209.

IRD3.5.8-2
The sensors shall be integrated with the spacecraft in a Class 10,000 cleanroom environment and maintained in that environment as much as possible during the integration and test flow.

IRD3.5.8-3
The facility requirements shall be documented in the sensor-spacecraft ICD and should include air cleanliness, air flow and recirculation rates, temperature and humidity, and tolerance for out-of-spec conditions (i.e., intermittent spikes) as a minimum.

IRD3.5.8-4
Requirements shall include verification by standard testing methods to be performed at regular, specified intervals.

3.5.9 GSE Cleanliness Requirements

IRD3.5.9-1
Integrating and sensor contractors shall document the need for contamination control of all Ground Support Equipment (GSE) entering cleanrooms.

IRD3.5.9-2
In addition, all GSE used inside thermal/vacuum chambers shall be cleaned and verified as vacuum compatible.

3.6 Software and GSE Requirements

3.6.1 Software Programming Language Requirements

The sensor software provider should implement all software using standard Ada (MIL-STD-1815A), C (ANSI STD X3/159-1989), or C++.

3.6.2 Sensor Flight Software Requirements

3.6.2.1 Sensor Flight Software Version Control

IRD3.6.2.1-1
All software and firmware shall be implemented with an internal identifier (embedded in the executable program) that can be included in the sensor engineering data.

IRD3.6.2.1-2
This identifier shall be keyed to the configuration management process so that the exact version of software and firmware residing in the sensor can be determined at any time.

3.6.2.2 Sensor Flight Software Loading

IRD3.6.2.2-1
Loading of the sensor microprocessor via the spacecraft uplink command processor (spacecraft commanding link) shall take no longer than 10 minutes following hardware reset or power-up.

3.6.2.3 Sensor Flight Software On-Orbit Installation and Verification

IRD3.6.2.3-1
Flight software shall be designed so that complete or partial revisions, consistent with C^3 constraints, can be installed and verified on-orbit.

3.6.3 Sensor Ground Support Equipment (GSE) Software Requirements

IRD3.6.3-1
Commands that can potentially damage hardware or cause injury to personnel shall require test operator authorization prior to being sent to the sensor for execution.

3.6.4 Sensor GSE to Spacecraft I&T GSE Interface

IRD3.6.4-1
The electrical sensor GSE shall interface with the spacecraft electrical GSE via a local area network (for example, ETHERNET) and/or point-to-point links if necessary for spacecraft level testing.

IRD3.6.4-2
GSE shall receive sensor telemetry via the spacecraft electrical GSE.

IRD3.6.4-3
Commanding of the sensor shall be from the spacecraft electrical GSE.

3.7 Environmental Requirements

3.7.1 Total Ionizing Dose Environment

IRD3.7.1-1
The sensor and the spacecraft shall be capable of meeting the proton and electron total dose levels for a 7-year mission given in Table 6 below.

IRD3.7.1-2
Two times the total dose shall be used to provide a design margin factor of two (Note: aE+N=a x 10^N, e.g. 3.264E+06 = 3.264 x 10^6; one mil is 10^{-3} inch). The depth dose curve is provided in Figure 7, and the total mission trapped proton and electron fluences are provided in Figure 8 and Figure 9 (the 2x margin has not been applied).

TABLE 6. TRAPPED PROTON/ELECTRON DOSAGE.

SHIELDING Mils (Al)	Trapped Protons Rad(Si)/7 Yr	Trapped Electrons Rad(Si)/7 Yr	Solar Flare Protons Rad(Si)/7 yr	Total Rad(Si)/7 yr
100	6.50 E03	1.81 E04	1.56E3	2.62 E04
200	4.79 E03	2.06 E03	7.12E2	7.56 E03
400	3.67 E03	6.76 E01	4.06E2	4.14 E03
600	3.05 E03	4.35 E01	0.00E0	3.09 E03
1000	2.25 E03	3.04 E01	0.00E0	2.28 E03

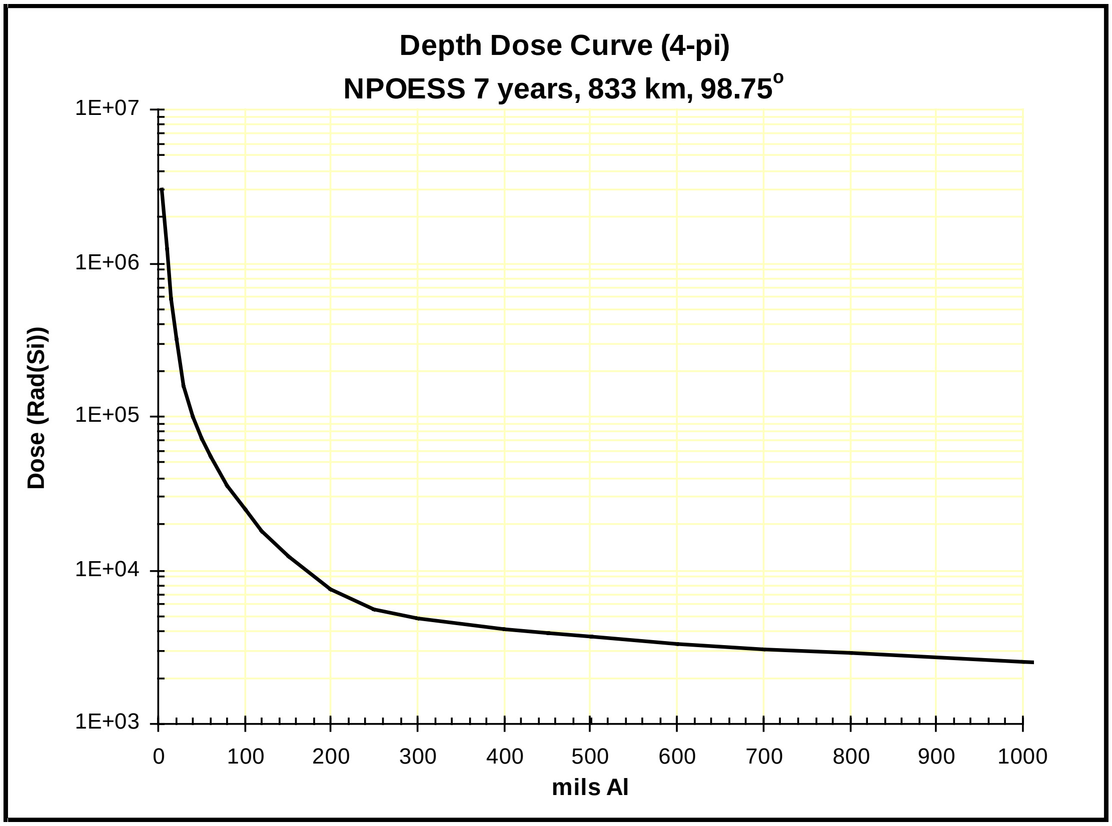

Figure 7. Depth Dose Curve

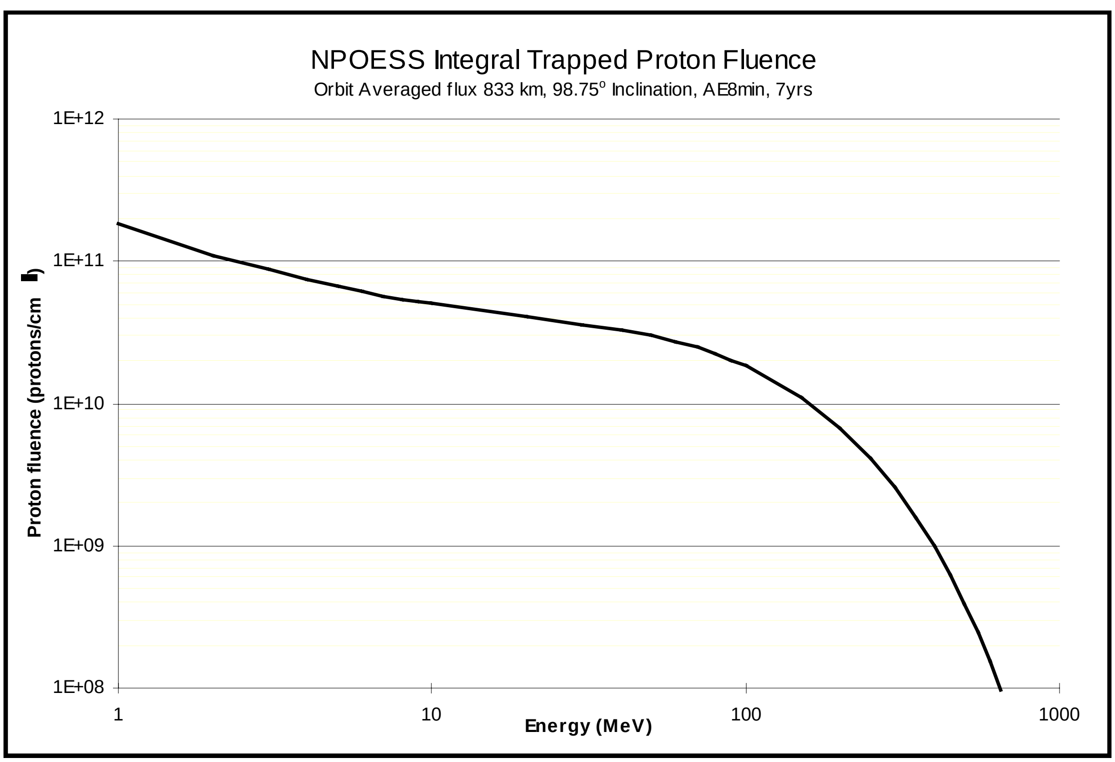

Figure 8. Integral Trapped Proton Fluence

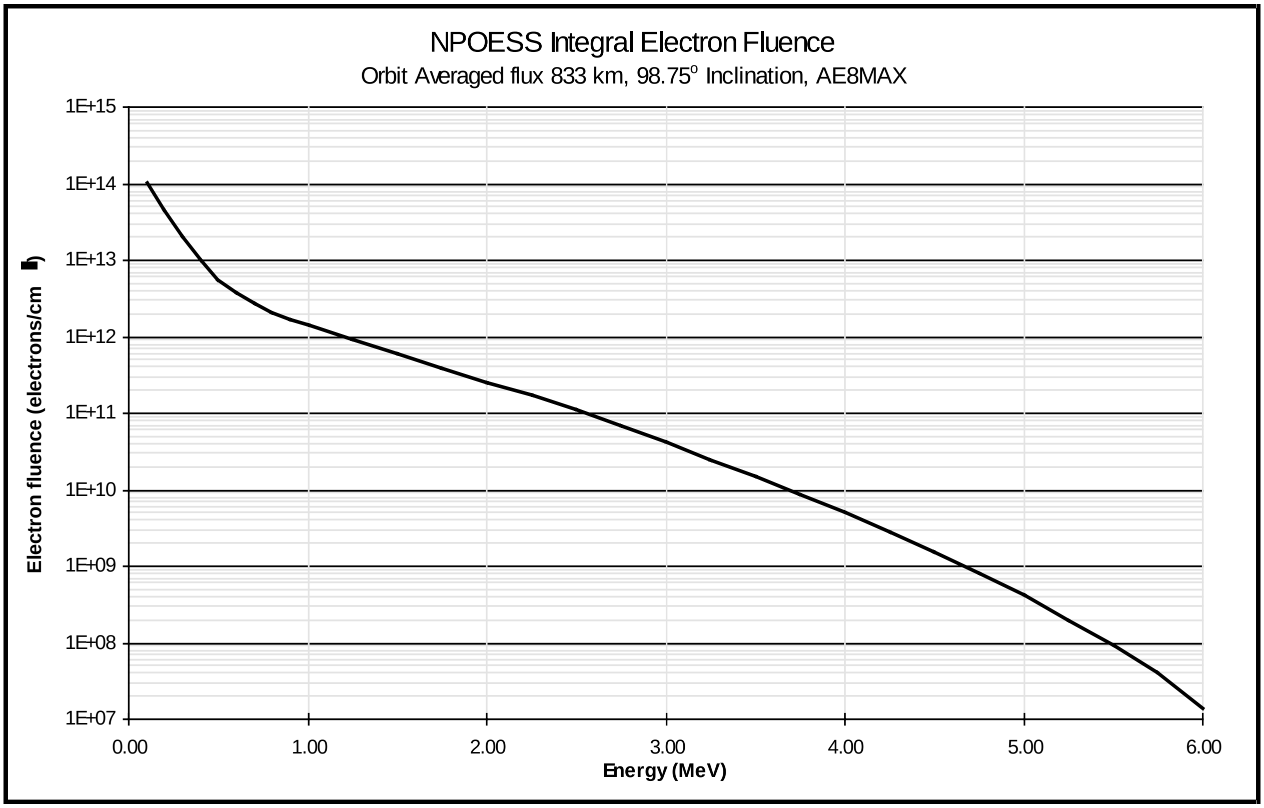

Figure 9. Integral Electron Fluence

3.7.2 Cosmic Ray and High Energy Proton Environment

3.7.2.1 Single Events Radiation Environment

IRD3.7.2.1-1
The sensor and the spacecraft shall be capable of meeting all performance requirements in the Cosmic Ray and High Energy Proton Radiation Environment specified in 3.7.2.1.1 and 3.7.2.1.2.

IRD3.7.2.1-2
Predictions of single events (i.e. single event latch-up, single event upset and single event burn-out) induced by galactic cosmic ray ions and high energy protons shall be performed separately and the results added.

3.7.2.1.1 Galactic Cosmic Ray Linear Energy Transfer (LET) Spectrum

IRD3.7.2.1.1-1
The integral galactic cosmic ray linear energy transfer spectrum in Figure 10 shall be used for prediction of ion-induced single events.

IRD3.7.2.1.1-2
The sensor and the spacecraft shall be capable of meeting all performance requirements in the background (90% worst case) environment.

IRD3.7.2.1.1-3
The sensor and the spacecraft and shall survive the heavy ion peak fluxes from a large solar flare (CREME96 Peak).

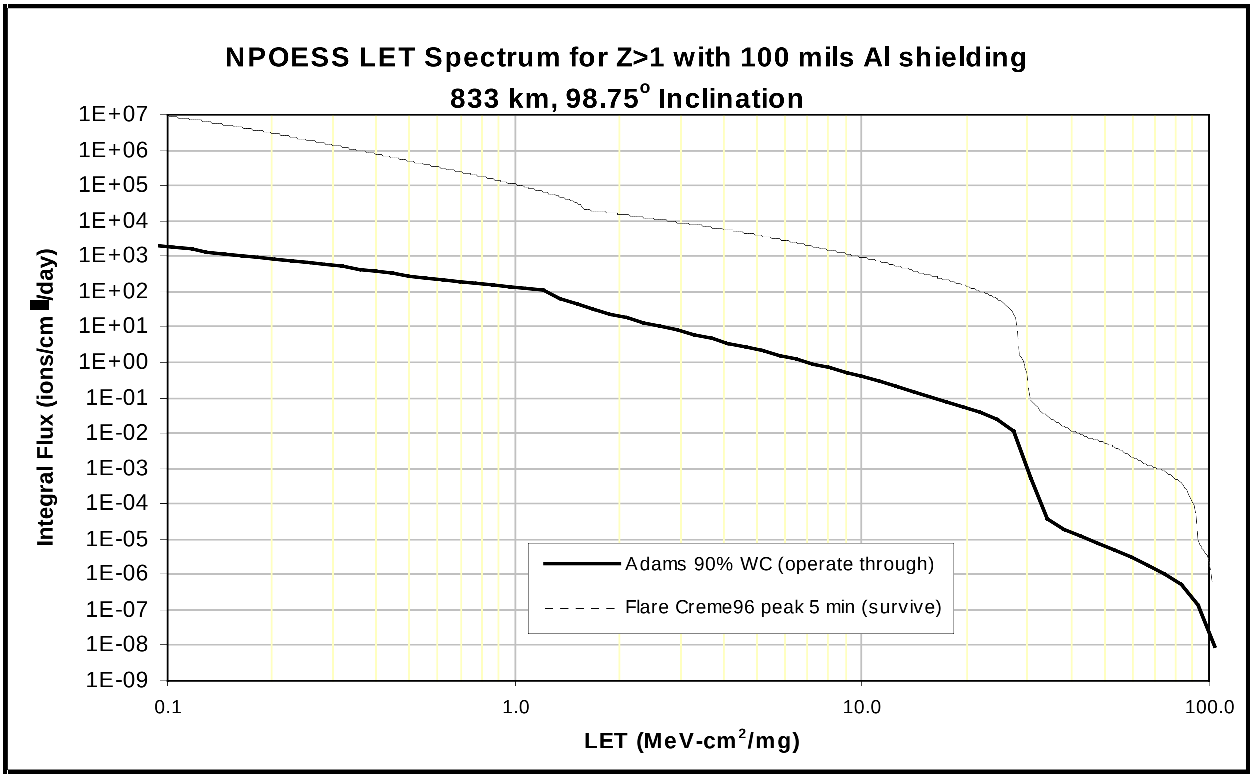

Figure 10. Integral Galactic Cosmic Ray Linear Energy Transfer Spectrum

3.7.2.1.2 High Energy Proton Fluence

IRD3.7.2.1.2-1
The integral proton flux is provided in Figure 11 for the background environment (trapped protons plus 90% worst case galactic cosmic ray protons and the peak flux of a large solar flare. These fluxes shall be used to determine the single event rates for the background environment and during a flare.

IRD3.7.2.1.2-2
The sensor and the spacecraft shall meet all performance requirements for the background environment, and shall survive the peak flux of the solar flare.

IRD3.7.2.1.2-3
The total number of single events shall be determined from the total proton fluence as provided in Figure 12, which combines the trapped, galactic cosmic ray, and flare total fluences.

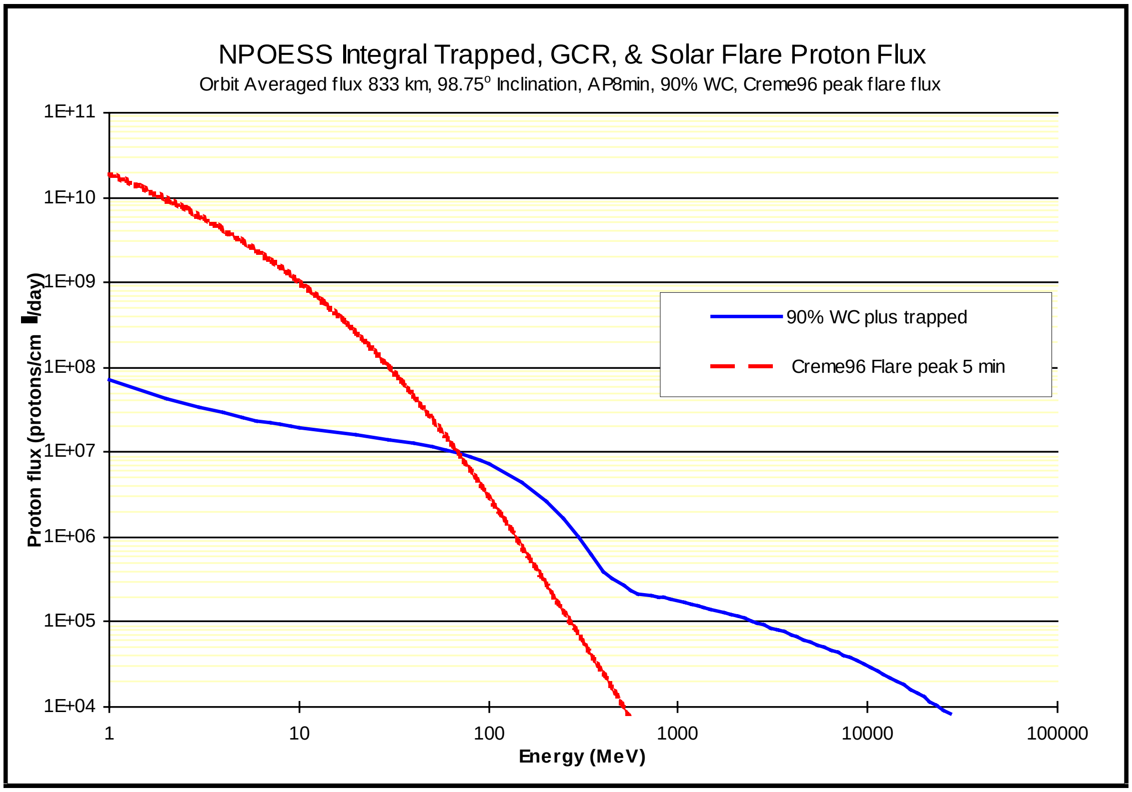

Figure 11. Integral Proton Flux

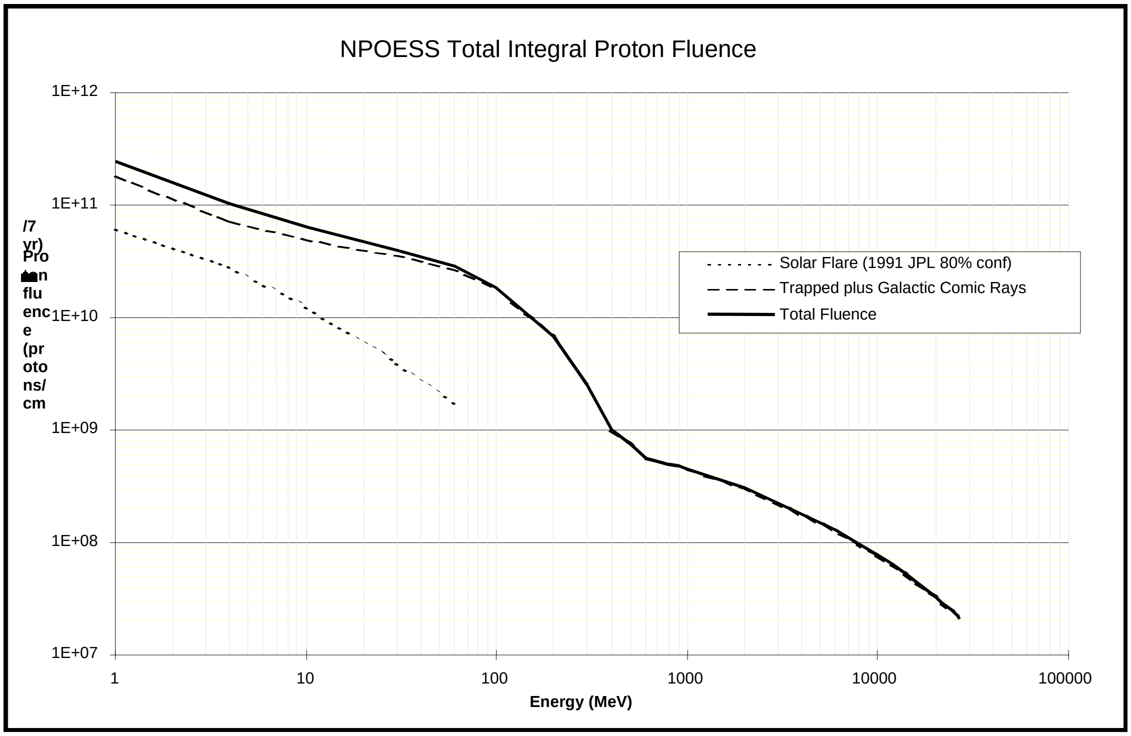

Figure 12. Total Integral Proton Fluence

3.7.2.1.3 Peak Fluxes

IRD3.7.2.1.3-1
The sensor and spacecraft shall be capable of meeting all performance requirements when exposed to trapped proton (E ≥ 5 MeV) flux of 9.98×10^3 particles/cm^{2}sec, trapped electron (E ≥ 0.5 MeV) flux of 3.31×10^4 particles/cm^{2}sec with the estimated solar flare proton peak fluxes and associated total event integral fluences, shown in Table 7, for each extremely large solar flare:

TABLE 7. PROTON PEAK FLUXES.

Energy (MeV)	Flux Total Event(Particles/cm^{2}sec) (Creme96 Peak)	Total Event Integral Fluence (Particles/cm^2) (1991 JPL 80% Confidence)
>10	1.16×10^4	6.61×10^{10}
>30	1.07×10^3	2.88×10^{10}
>60	1.68×10^2	1.27×10^{10}
>100	-	1.72×10^9

The total event integral fluence is accumulated within a time interval of a few hours to two days.

3.7.2.2 Displacement Damage

IRD3.7.2.2-1
Charge Coupled Devices (CCDs) and electronic parts shall survive the displacement damage caused by the total proton fluence shown in Figure 12, and the depth-displacement damage curve in Figure 13.

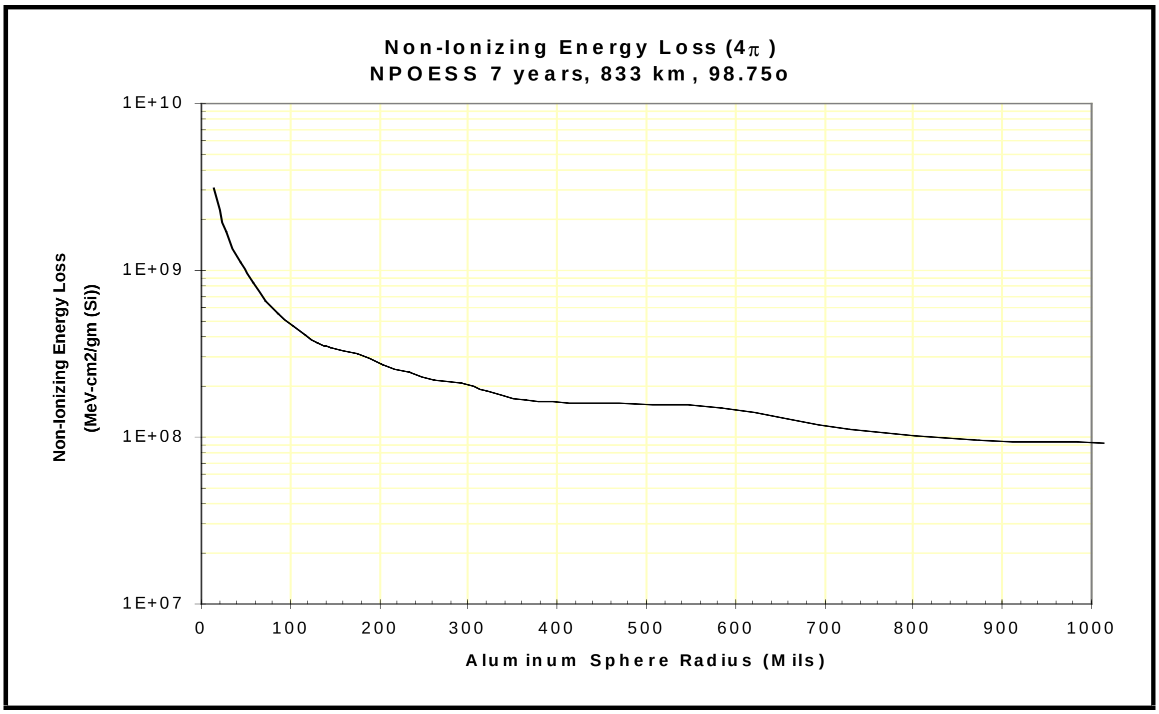

Figure 13. Depth Displacement Damage Curve

IRD3.7.2.2-2
Where CCD detectors are used, the design shall incorporate features that minimize the effects of displacement damage.

3.7.3 Atomic Oxygen

IRD3.7.3-1
The sensor shall meet performance requirements during exposure to atomic oxygen (AO) experienced during a 833 km polar orbit for seven years. Atomic oxygen fluence is shown in .Figure 14

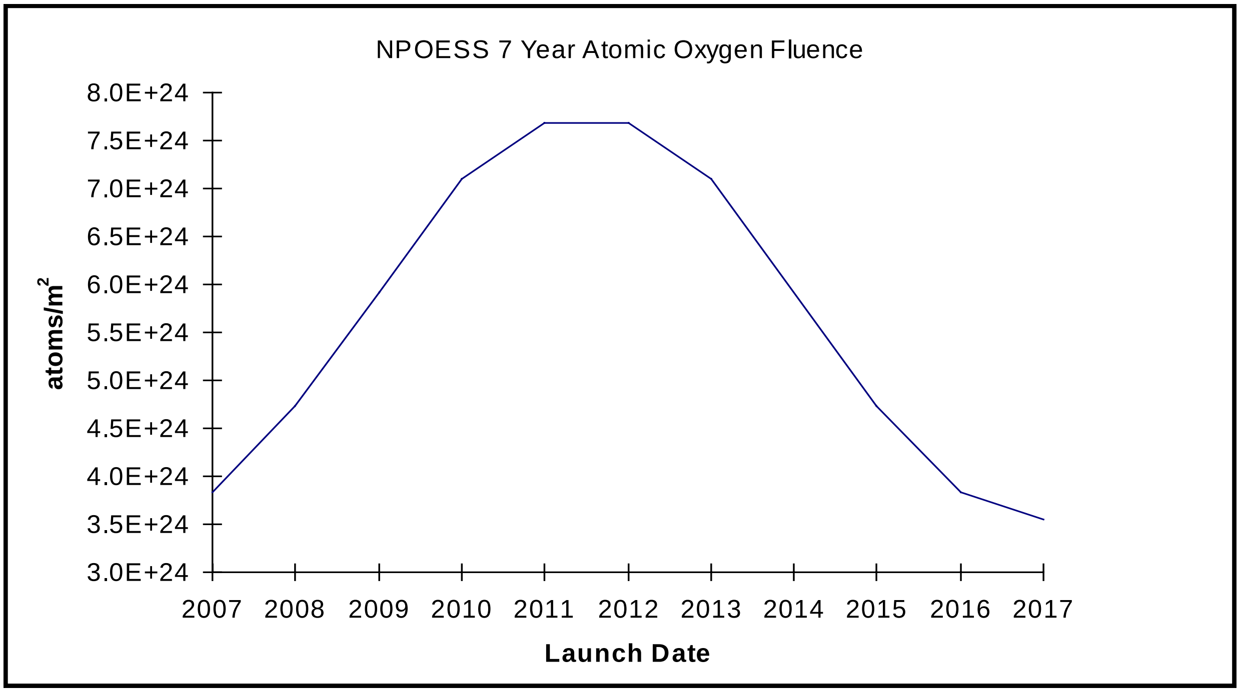

Figure 14. Atomic Oxygen Fluence

3.7.4 Electromagnetic Compatibility

3.7.4.1 General

There are seven macro-level interfaces to consider for EMC:

1) Interface between sensors and spacecraft bus
2) Interface between spacecraft and external environment
3) Interface between spacecraft and launch vehicle
4) Interface between spacecraft and ground support equipment.
5) Interface between spacecraft bus/sensors and test equipment.
6) Interface between sensors and launch vehicle
7) Interface between sensors and sensors

There are four EMC interfaces:

1) Conducted Emissions/Susceptibility
2) Radiated Emissions/Susceptibility
3) Grounding
4) Wiring

3.7.4.2 Baseline Requirements

3.7.4.2.1 System Electromagnetic Compatibility

IRD3.7.4.2.1-1
The spacecraft bus, sensors, ground support equipment, and test equipment shall operate within acceptable limits, i.e., without performance degradation with each other, and the external environment.

3.7.4.2.1.1 Search and Rescue Sensor Compatibility

IRD3.7.4.2.1.1-1
The Search and Rescue sensor shall not be impacted at any of its receiver frequencies.

3.7.4.2.2 Interface Margins

IRD3.7.4.2.2-1
Each interface margin shall have at least 12 dB for qualification testing and 6 dB for acceptance testing.

IRD3.7.4.2.2-2
Electroexplosive device circuits shall have at least a 20 dB margin to the defined susceptibility requirements. The interface requirements defined in this section have the required 12 dB margin.

3.7.4.2.3 Frequency Management

IRD3.7.4.2.3-1
All intended receivers and transmitters shall have frequency assignment and allocation in accordance with all National Telecommunications and Information Administration regulations.

3.7.4.3 External Environment

3.7.4.3.1 External RF Environment

IRD3.7.4.3.1-1
The system shall operate without degradation in the following RF environments (Table 8): ("Operate" is defined as the standard operations for that part of the mission). For example, if a payload is powered off during ascent or at the launch pad then the 'operate' requirement is that it will operate without degradation upon being powered on.

TABLE 8. EXTERNAL RF ENVIRONMENT

Frequency, Hz	Factory/Transport, V/m	Launch Pad, V/m	Ascent, V/m	On Orbit, V/m
10 k - 100 M	20	20	20	20
100 M - 1 G	20	20	100	20
1 G - 5 G	100	100	200	30
5 G – 10 G	100	100	200	90
10 G - 40 G	20	20	20	20

These values are design to environments. The ‘Factory/Transport’ and ‘Launch’ pad environments can be reduced by any combination of procedures, facility shielding, or shipping container shielding. The ‘Factory/Transport’ and ‘Launch Pad’ assume a 40 dB shielded shipping container for transport and standard RF operation reduction practices at the launch pad. The probable launch vehicle specifications are such that the launch vehicle is allowed to produce a RF environment of 100 to 200 V/m at the space vehicle interface from 1 GHz to 10 GHz.

IRD3.7.4.3.1-2
The intended receivers shall operate without performance degradation for the external environment outside their pass band and shall survive and automatically recover for the external environment inside their pass band.

3.7.4.3.2 Lightning

IRD3.7.4.3.2-1
The system shall be capable of detecting any change in the criteria for launch caused by either a direct or nearby lightning strike.

IRD3.7.4.3.2-2
This shall not preclude the use of monitoring external to the satellite.

3.7.4.3.3 Spacecraft Charging from All Sources

IRD3.7.4.3.3-1
The system shall operate without performance degradation due to surface charging, bulk charging, and deep charging in accordance with MIL-STD-1541A.

3.7.4.4 Wiring

IRD3.7.4.4-1
The power and signal wiring shall be shield twisted pairs with EMI backshells.

IRD3.7.4.4-2
The shield shall be terminated on the backshell.

IRD3.7.4.4-3
Each power and signal wire shall have a dedicated return.

3.7.4.5 Grounding

IRD3.7.4.5-1
The impedance between the sensor and spacecraft ground shall be 10 milli-ohms or less.

3.7.4.6 Conducted and Radiated Interface Requirements

IRD3.7.4.6-1
The interfaces shall meet the requirements (including CE101, CE102, CE106, CS101, CS103, CS104, CS105, CS114, CS116, RE101, RE102, RS101, and RS103) of MIL-STD-461D as tailored by MIL-STD-1541A. CS114 and CS116 only apply to power cables. CS103, CS104, CS105, and CE106 apply only to subsystems with transmit/receive antennas.

IRD3.7.4.6-2
The upper frequency range of RE102 and RS103 shall be extended to envelope all sensor, transmitter, and receiver frequencies.

3.7.4.6.1 Radiated Emission RE101

IRD3.7.4.6.1-1
The radiated AC magnetic field levels from a component/instrument shall be limited to 60 dB above 1 pT between 20 Hz and 50 kHz and measured per MIL-STD-462D, test RE101, with a field measured at a distance of one meter.

IRD3.7.4.6.1-2
The measurement bandwidth shall be 10 Hz between 20 Hz and 200 Hz, 100 Hz between 200 Hz and 20 kHz, and 1 kHz between 20 kHz and 50 kHz.

3.7.4.6.2 Radiated Emissions RE102

IRD3.7.4.6.2-1
The interface requirement shall be less than 60 dBuV/m for unintentional emissions except as tailored in the Table 9 for the search and rescue receivers, the DCS receivers, the Space Ground Link System (SGLS) receivers, sensor receivers, and launch vehicle receivers.

IRD3.7.4.6.2-2
Intentional transmitter emissions from the spacecraft bus or payload transmitters shall be below 5 V/m (134 dBuV/m) at the interface between the spacecraft bus and payload. Radiated emissions testing, RE102, is from 10 kHz to the greater of 18 GHz or twice the intentional frequency usage.

TABLE 9. TRANSMITTER/RECEIVER RADIATED EMISSIONS TAILORING (TBR)

Frequency (MHz)	Type	Gain dBi	Transmit Power W	Received Power dBm	Bandwidth MHz	Radiated Emissions dBμV/m
137.5	LRPT-TX	-0.71	3.5-72.3	N/A	0.049-0.150	134
137.62	LRPT-TX	-0.71	3.5-72.3	N/A	0.049-0.150	134
400.15	LRD-TX	-0.71	48-726	N/A	0.1 - 0.25	134
401.0	LRD-TX	-0.71	48-726	N/A	0.1 - 0.25	134
1698	TT&C.a-TX	-0.97-11.14	0.002-0.3	N/A	4	134
1698	TT&C.a-RX	-0.97-11.14	N/A	-100	4	37
1702.5	TT&C.a-TX	-0.97-11.14	0.002-0.17	N/A	4	134
1702.5	TT&C.a-RX	-0.97-11.14	N/A	-100	4	37
1707	TT&C.a-TX	-0.97-11.14	0.002-0.17	N/A	4	134
1707	TT&C.a-RX	-0.97-11.14	N/A	- 100	4	37
2202.5	TT&C.b-TX	-0.97-13.35	0.002-0.17	-145	5	134
2028.135	TT&C.b-RX	-0.97	N/A	TBD	5	37
2247.5	TT&C.c-TX	-0.97-13.35	TBD	N/A	5	134
2026.0	TT&C.c-RX	-0.97	N/A	TBD	5	37
2237.5	TT&C.d-TX	-0.97-13.35	TBD	N/A	5	134
1791.748	TT&C.d-RX	-0.97	N/A	TBD	5	37
x-band	TT&C.e-TX	TBD	TBD	N/A	100	134
x-band	TT&C.e-RX	TBD	N/A	TBD	100	37
2202.5	TT&C.e-TX	-0.97	TBD	TBD	5	134
2028.135	TT&C.e-RX	-0.97	N/A	TBD	5	37
7750-7850	HRD.a-TX	7	13.5-452.7	N/A	8.10-9.26	134
7450-7550	HRD.b-TX	7	13.5-452.7	N/A	8.10-9.26	134
8025-8400	SMD.a-TX	7-25	0.3-175.1	N/A	375	134
25500-27000	SMD.b-TX	34	4.8-46.5	N/A	1500	134
7450-7550	SMD.c-TX	24.39	0.5-1.5	N/A	100	134
7750-7850	SMD.c-TX	24.39	0.5-1.5	N/A	100	134
2020.4-2123.3	SPACE.a-TX	TBD	TBD	TBD	100	134
2200-2300	SPACE.a-RX	TBD	N/A	N/A	100	37
13747-13802	SPACE.b-TX	TBD	TBD	N/A	55	134
25250-27250	SPACE.b-TX	TBD	TBD	N/A	2000	134
14807-15110	SPACE.b-RX	TBD	N/A	TBD	303	37
1100-1700	CMIS-B1-RX	TBD	N/A	TBD	TBD	10
4900-5900	CMIS-B2-RX	TBD	N/A	TBD	TBD	17
6200-7000	CMIS-B3-RX	TBD	N/A	TBD	TBD	17
10400-11000	CMIS-B4-RX	TBD	N/A	TBD	TBD	13
18200-23700	CMIS-B5-RX	TBD	N/A	TBD	TBD	13
35100-38900	CMIS-B6-RX	TBD	N/A	TBD	TBD	59
50100-58500	CMIS-B7-RX	TBD	N/A	TBD	TBD	56
59100-59700	CMIS-B8-RX	TBD	N/A	TBD	TBD	56
60250-61350	CMIS-B9-RX	TBD	N/A	TBD	TBD	56
62900-63600	CMIS-B10-RX	TBD	N/A	TBD	TBD	56
83700-93300	CMIS-B11-RX	TBD	N/A	TBD	TBD	56
112600-120000	CMIS-B12-RX	TBD	N/A	TBD	TBD	56
148700-153300	CMIS-B13-RX	TBD	N/A	TBD	TBD	56
163700-168300	CMIS-B14-RX	TBD	N/A	TBD	TBD	56
174310-192310	CMIS-B15-RX	TBD	N/A	TBD	TBD	56
217700-222300	CMIS-B16-RX	TBD	N/A	TBD	TBD	56

TABLE 9. TRANSMITTER/RECEIVER RADIATED EMISSIONS TAILORING (TBR) - (CONT)

Frequency (MHz)	Type	Gain dBi	Transmit Power	Received Power dBm	Bandwidth	Radiated Emissions dBμV/m
339000-341000	CMIS-B17-RX	TBD	N/A	TBD	TBD	56
372000-389000	CMIS-B18-RX	TBD	N/A	TBD	TBD	56
408000-412000	CMIS-B19-RX	TBD	N/A	TBD	TBD	56
547000-567000	CMIS-B20-RX	TBD	N/A	TBD	TBD	56
23000	AMSU-A-CH1-RX	TBD	N/A	TBD	TBD	59
31400	AMSU-A-CH2-RX	TBD	N/A	TBD	TBD	59
50300	AMSU-A-CH3-RX	TBD	N/A	TBD	TBD	56
52800	AMSU-A-CH4-RX	TBD	N/A	TBD	TBD	56
53600	AMSU-A-CH5-RX	TBD	N/A	TBD	TBD	56
54400	AMSU-A-CH6-RX	TBD	N/A	TBD	TBD	56
54940	AMSU-A-CH7-RX	TBD	N/A	TBD	TBD	56
55500	AMSU-A-CH8-RX	TBD	N/A	TBD	TBD	56
57300	AMSU-A-CH9-RX	TBD	N/A	TBD	TBD	56
57300+/- 217 MHz	AMSU-A-CH10-RX	TBD	N/A	TBD	TBD	56
L+/-48 MHz	AMSU-A-CH11-RX	TBD	N/A	TBD	TBD	56
L+/-22 MHz	AMSU-A-CH12-RX	TBD	N/A	TBD	TBD	56
L+/-10 MHz	AMSU-A-CH13-RX	TBD	N/A	TBD	TBD	56
F+/-4.5 MHz	AMSU-A-CH14-RX	TBD	N/A	TBD	TBD	56
89000	AMSU-A-CH15-RX	TBD	N/A	TBD	TBD	56
89000	AMSU-B-CH16-RX	TBD	N/A	TBD	TBD	56
150000	AMSU-B-CH17-RX	TBD	N/A	TBD	TBD	56
183000	AMSU-B-CH18-RX	TBD	N/A	TBD	TBD	56
183000	AMSU-B-CH19-RX	TBD	N/A	TBD	TBD	56
183000	AMSU-B-CH20-RX	TBD	N/A	TBD	TBD	56
503000	CrMS-CH1-RX	TBD	N/A	TBD	180	56
517600	CrMS-CH2-RX	TBD	N/A	TBD	400	56
528000	CrMS-CH3-RX	TBD	N/A	TBD	400	56
53481,53711	CrMS-CH4-RX	TBD	N/A	TBD	2x170each	56
54400	CrMS-CH5-RX	TBD	N/A	TBD	400	56
54940	CrMS-CH6-RX	TBD	N/A	TBD	500	56
55500	CrMS-CH7-RX	TBD	N/A	TBD	330	56
56020	CrMS-CH8-RX	TBD	N/A	TBD	270	56
57950	CrMS-CH9-RX	TBD	N/A	TBD	400	56
58220,58540	CrMS-CH10-RX	TBD	N/A	TBD	2x100	56
58390	CrMS-CH11-RX	TBD	N/A	TBD	68	56
58323.88+/-24	CrMS-CH12-RX	TBD	N/A	TBD	2X16	56
58323.88+/-12	CrMS-CH13-RX	TBD	N/A	TBD	2x8	56
58323.88+/-6	CrMS-CH14-RX	TBD	N/A	TBD	2x4	56
58323.88-1.35	CrMS-CH15-RX	TBD	N/A	TBD	1.3	56
58323.88-1.35	CrMS-CH16-RX	TBD	N/A	TBD	1.3	56
58323.88-1.4	CrMS-CH17-RX	TBD	N/A	TBD	1.4	56
58323.88+3	CrMS-CH18-RX	TBD	N/A	TBD	1.3	56
58323.88+3	CrMS-CH19-RX	TBD	N/A	TBD	2.0	56
58446.59-12	CrMS-CH20-RX	TBD	N/A	TBD	2x8	56
58446.59-6	CrMS-CH21-RX	TBD	N/A	TBD	2x4	56
58446.59-2.8	CrMS-CH22-RX	TBD	N/A	TBD	2.4	56
58446.59	CrMS-CH23-RX	TBD	N/A	TBD	1.0	56
58323.88+2.8	CrMS-CH24-RX	TBD	N/A	TBD	2.4	56

TABLE 9. TRANSMITTER/RECEIVER RADIATED EMISSIONS TAILORING (TBR) - (CONT)

Frequency (MHz)	Type	Gain dBi	Transmit Power W	Received Power dBm	Bandwidth MHz	Radiated Emissions dBμV/m
113250	CrMS-CH25-RX	TBD	N/A	TBD	1000	56
115250	CrMS-CH26-RX	TBD	N/A	TBD	1000	56
116200	CrMS-CH27-RX	TBD	N/A	TBD	500	56
117700	CrMS-CH28-RX	TBD	N/A	TBD	500	56
117150	CrMS-CH29-RX	TBD	N/A	TBD	400	56
117550	CrMS-CH30-RX	TBD	N/A	TBD	400	56
118750+/-800	CrMS-CH31-RX	TBD	N/A	TBD	2x400	56
118750+/-450	CrMS-CH32-RX	TBD	N/A	TBD	2x300	56
118750+/-225	CrMS-CH33-RX	TBD	N/A	TBD	2x150	56
183310-17300	CrMS-CH34-RX	TBD	N/A	TBD	4000	56
183310+/-7000	CrMS-CH35-RX	TBD	N/A	TBD	2x2000	56
183310+/-4500	CrMS-CH36-RX	TBD	N/A	TBD	2x2000	56
183310+/-3000	CrMS-CH37-RX	TBD	N/A	TBD	2x1000	56
183310+/-1800	CrMS-CH38-RX	TBD	N/A	TBD	2x1000	56
183310+/-1000	CrMS-CH39-RX	TBD	N/A	TBD	2x500	56
121.5	SARR-RX	TBD	N/A	-145	TBD	-20 to -5
243	SARR-RX	TBD	N/A	-145	TBD	-20 to -5
406.05	SARR-RX	TBD	N/A	-145	TBD	-20 to -5
401.65	ARGOS-RX	TBD	N/A	TBD	TBD	20
1575.42	GPS-L1-RX	TBD	N/A	-135	TBD	20
1227.2	GPS-L2-RX	TBD	N/A	-135	TBD	20
1602	GLONASS-L1-RX	TBD	N/A	-135	TBD	20
1246	GLONASS-L2-RX	TBD	N/A	-135	TBD	20
18700	JMR-RX	TBD	N/A	TBD	TBD	13
23000	JMR-RX	TBD	N/A	TBD	TBD	59
34000	JMR-RX	TBD	N/A	TBD	TBD	59
5300	JASON-3-TX	TBD	TBD	N/A	320	134
13575	JASON-3-TX	TBD	TBD	N/A	320	134
1544.5	SARR-TX	TBD	10	N/A	TBD	134
2205.5 (TBR)	LV-TX	3.4	12	N/A	256 kHz	146.5
2206.5 (TBR)	LV-TX	3.4	12	N/A	200 kHz	146.1
5765 (TBR)	LV-TX	3.4	700	N/A	6	163.1
416.5 (TBR)	LV-RX	1.5	N/A	-103	190 kHz	14
5690	LV-RX	3.4	N/A	-70	11	26
60.1	Receiver	TBD	N/A	TBD	+/- 0.5 kHz	56 (TBR)
142.9	Receiver	TBD	N/A	TBD	+/- 0.5 kHz	66 (TBR)
285.813	Receiver	TBD	N/A	TBD	+/- 0.5 kHz	61 (TBR)
375.972	Receiver	TBD	N/A	TBD	+/- 0.5 kHz	62 (TBR)
631.73	Receiver	TBD	N/A	TBD	+/- 0.5 kHz	67 (TBR)
751.944	Receiver	TBD	N/A	TBD	+/- 0.5 kHz	80 (TBR)

For Information Only - Light Frequency Usage
Infrared Sounder: 3.7 to 4.7u, 5.7 to 8.3u, 8.7 to 15.5u
Ozone Mapper: 0.25 to 0.38u
Cloud Lidar: 1.064u
Space Environmental Sensor Suite: 80-630 nm
1u=10(exp)-6m

3.7.4.6.3 Radiated Susceptibility RS101

IRD3.7.4.6.3-1
The component / instrument shall not exhibit any undesired response, malfunction or performance degradation beyond the tolerances allowed by this specification when it is subjected to an AC magnetic field of 124 dBpT over the frequency range of 30 Hz to 200 kHz applied at a distance of 7 cm. from the component.

IRD3.7.4.6.3-2
The RS101 test method of MIL-STD-462D shall be used for this measurement.

3.7.4.6.4 Radiated Susceptibility RS103

IRD3.7.4.6.4-1
The satellite shall meet the RS103 requirements of MIL-STD-461D as tailored by the external RF environment, launch vehicle transmitters, and satellite transmitters.

3.7.5 Electrostatic Discharge

3.7.5.1 Provisions

IRD3.7.5.1-1
Appropriate provisions shall be made to avoid and to protect against the effects of static electricity generation and discharge in areas containing electrostatic sensitive devices such as microcircuits, initiators, explosive bolts, or any loaded explosive device. DOD-HDBK-263 provides examples of appropriate provisions.

3.7.5.2 Design and Test

IRD3.7.5.2-1
If voltages over 100 V are present, the design shall be protected (e.g. potting, pressure vessel) and tested for arcing.

3.7.5.3 Grounding

IRD3.7.5.3-1
There shall be a capability to ground both equipment and personnel working on and around the sensor, subsystems, and components.

3.8 General Considerations

3.8.1 Spacecraft Reference Coordinate Frame

IRD3.8.1-1
A right-hand, orthogonal, body-fixed XYZ coordinate system shall be used. The +Z-axis is downward towards nadir, the Y-axis is along the orbit normal (+Y is opposite the orbital angular

momentum) and the X-axis is along the spacecraft velocity vector (+X toward the direction of spacecraft travel).

3.8.2 Dimension Unit Standard

IRD3.8.2-1
All documents shall provide units in metric as the minimum, with English units as an option.

IRD3.8.2-2
All interfaces shall be specified in the international system of units, System Internationale (SI), unless design heritage precludes this.

IRD3.8.2-3
Dimensioning shall be in the as-designed units and identified when other than SI.

3.8.3 Nominal Orbit Parameters

IRD3.8.3-1
The NPOESS satellite shall operate in a near circular, sun-synchronous orbit. The nominal orbit for the satellite is 833 km altitude, 98.7 degree inclination. The orbit will be a "precise" orbit (i.e., altitude maintained to ± 17 (TBR) km, ± 0.05 (TBR) degrees inclination, nodal crossing times maintained to ± 10 minutes throughout the mission lifetime) to minimize orbital drift (precession).

IRD3.8.3-2
NPOESS shall be capable of flying at any equatorial node crossing time except for the restriction in 3.8.3-3. However, the nominal configuration is with the satellite orbits equally spaced, with 0530 and 1330 nodal crossing times for the U.S. Government satellites and 0930 for the METOP satellite.

IRD3.8.3-3
The satellite shall only be flown in orbits that keep sunlight off the cold side of the spacecraft. Because of natural variations in the orbit, the 10 minute nodal crossing time constraint, and variations in the solar illumination of the satellite, this will restrict the spacecraft from flying in orbits within about 30 (TBR on satellite contractor) minutes of noon.

3.8.4 Sensor/Spacecraft Integration Responsibility

IRD3.8.4-1
The Government will be the system integrator until a Total System Performance Responsibility (TSPR) contractor is selected in 4Q2000. Until that time the government will be responsible for accommodation trades, resource allocation (weight, power, space, bandwidth), and resolving interface issues.

3.9 Launch Vehicle Environments

The baseline NPOESS launch vehicle for the baseline NPOESS satellite constellation is planned for a medium launch vehicle. The levels specified in this section reflect the Evolved Expendable Launch Vehicle (EELV) Medium Launch Vehicle (MLV) proposed environments.

3.9.1 Sensor Fairing Dynamic Envelopes

IRD3.9.1-1
There are standard minimum sizes for payload fairing envelopes (actual fairing envelopes may be larger). These envelopes define the useable volume inside the fairing and forward of the Standard Interface Plane (SIP). However, there will be some stay-out zones in the fairing envelope, which will be negotiated as part of the Satellite to Launch Vehicle ICD. It shall be the responsibility of the integrating contractor to ensure that none of the sensors protrude into the stay-out zones as defined in the Satellite to Launch Vehicle ICD.

Nominal fairing envelope size is as shown in Figure 16.

Figure 15. Deleted

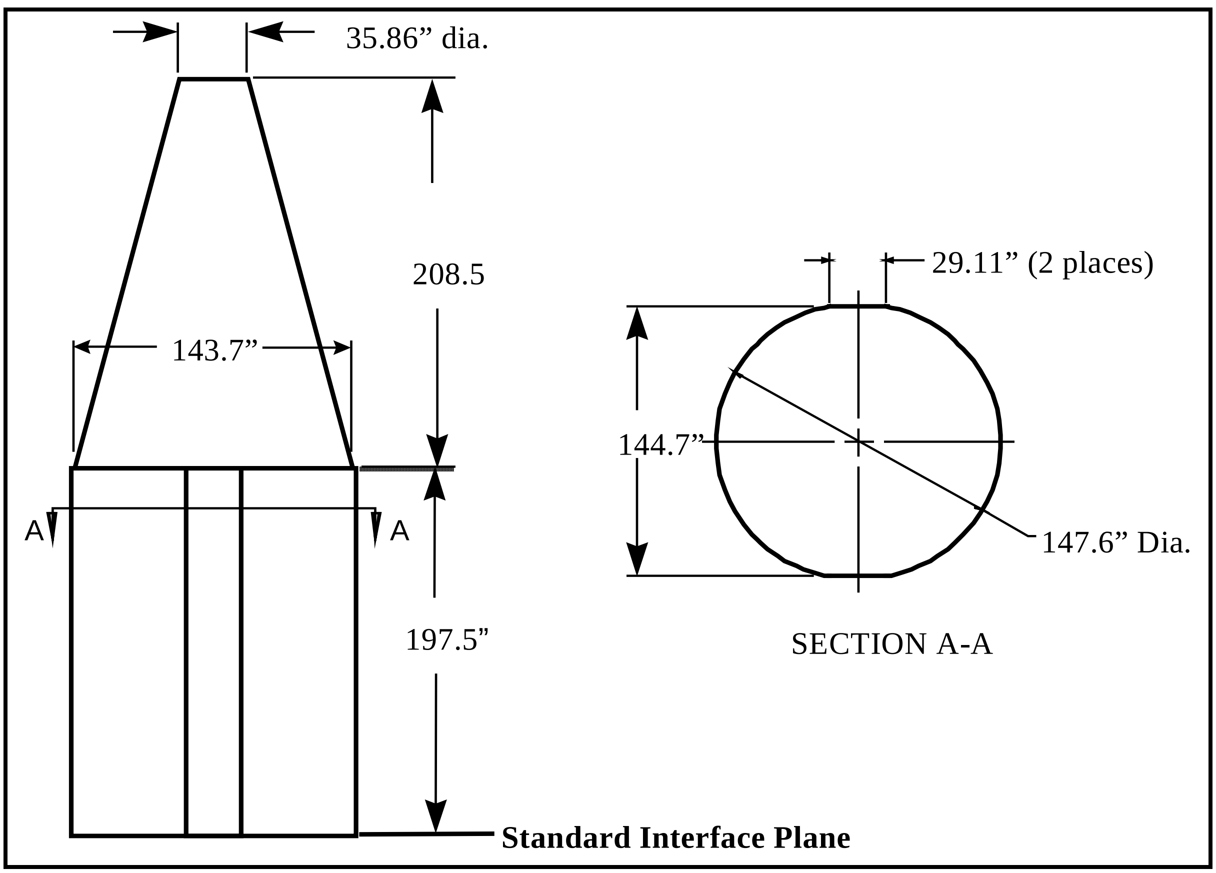

Figure 16. MLV Spacecraft Dynamic Envelope

3.9.2 Thermal

Reserved

3.9.3 Payload Fairing Thermal Environment

IRD3.9.3-1
The worst case effective internal environment in the satellite compartment within the fairing during ascent is defined in Figure 17. The surfaces seen by the satellite will generally fall into one of two categories: surfaces with low emissivity (e ≤ 0.3) and those of higher emissivity (e ≤ 0.9). Maximum temperatures as a function of the time from launch, 149°C for a surface emissivity of 0.3 and 93°C for a surface emissivity of 0.9, are shown in the plot. The exact configuration and percentages of each type of surface is both mission specific and launch vehicle (LV) specific. Temperatures may exceed those shown but in no case shall the total integrated thermal energy imparted to the spacecraft exceed the maximum total integrated energy that result in the temperature profile shown in Figure 17.

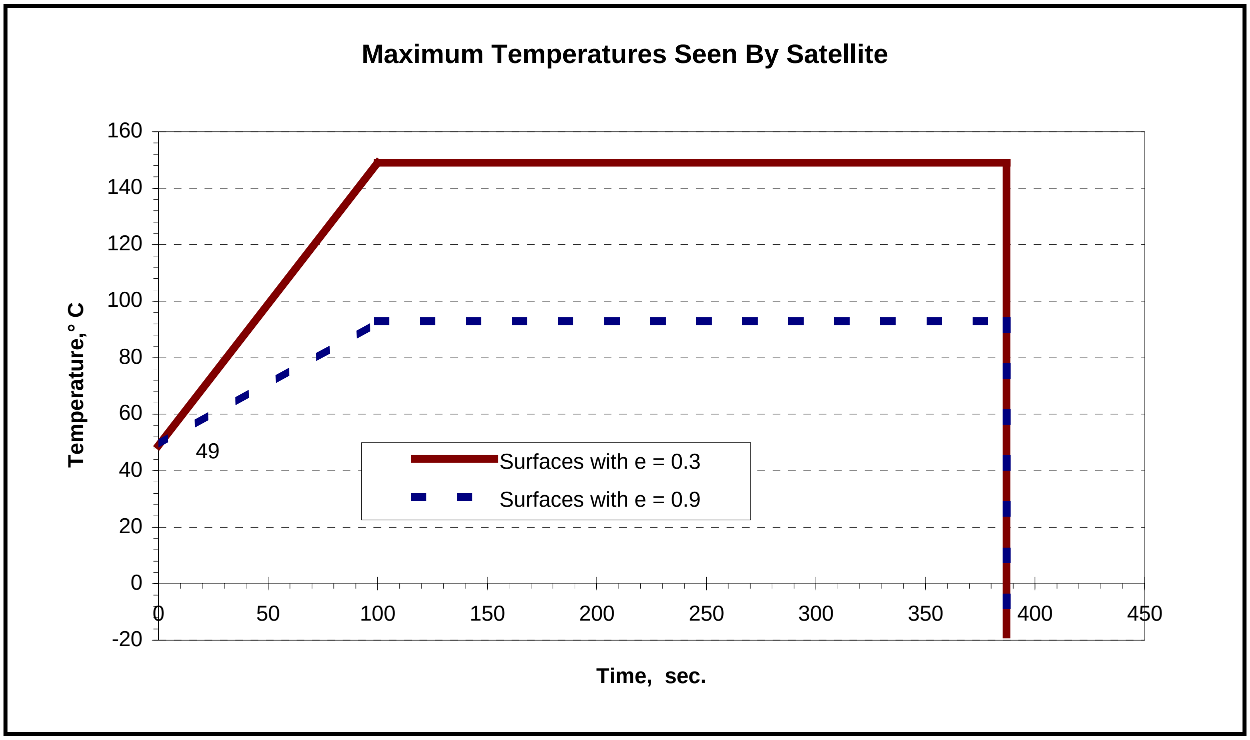

Figure 17. Maximum PLF Inner Temperatures

3.9.4 Heat Flux

Reserved

3.9.5 Free Molecular Heating

IRD3.9.5-1
The maximum instantaneous 3-sigma Free Molecular Heating on satellite surfaces perpendicular to the velocity vector at the time of fairing separation shall not exceed 0.1814 watts/ cm^2 °C (320 Btu/hr-ft^2).

3.9.6 Shock

IRD3.9.6-1
The maximum shock spectrum at the SIP (value at 95% probability with 50% confidence; resonant amplification factor, Q=10) shall not exceed the levels shown in Table 10. These levels are shown graphically in Figure 18.

TABLE 10. EELV SHOCK SPECIFICATION

Shock Spectrum from EELV to Satellite (g's)		**Shock Spectrum from Satellite to EELV (g's) (due to satellite separation)**	
Freq-Hz	MLV	Freq-Hz	MLV
100	40	100	150
125	-	125	175
160	-	160	220
200	-	200	260
250	-	250	320
315	-	315	400
400	-	400	500
500	-	500	725
630	-	630	1100
800	-	800	2000
1600	-	1600	5000
2000	-	2000	5000
5000	3000	5000	5000
10000	3000	10000	5000

Refer to graph in Figure 18 for intermediate frequencies

3.9.7 Launch Pressure Profile

IRD3.9.7-1
The satellite shall be designed to withstand a payload fairing internal pressure decay rate of 20 mb/sec.

3.9.8 Acceleration Load Factors

Figure 19 defines satellite center-of-gravity acceleration values that when used to calculate launch vehicle/satellite interface bending moments, axial loads and shear loads will yield values that are guaranteed not to be exceeded. Satellite weights include any payload adapter which may be required.

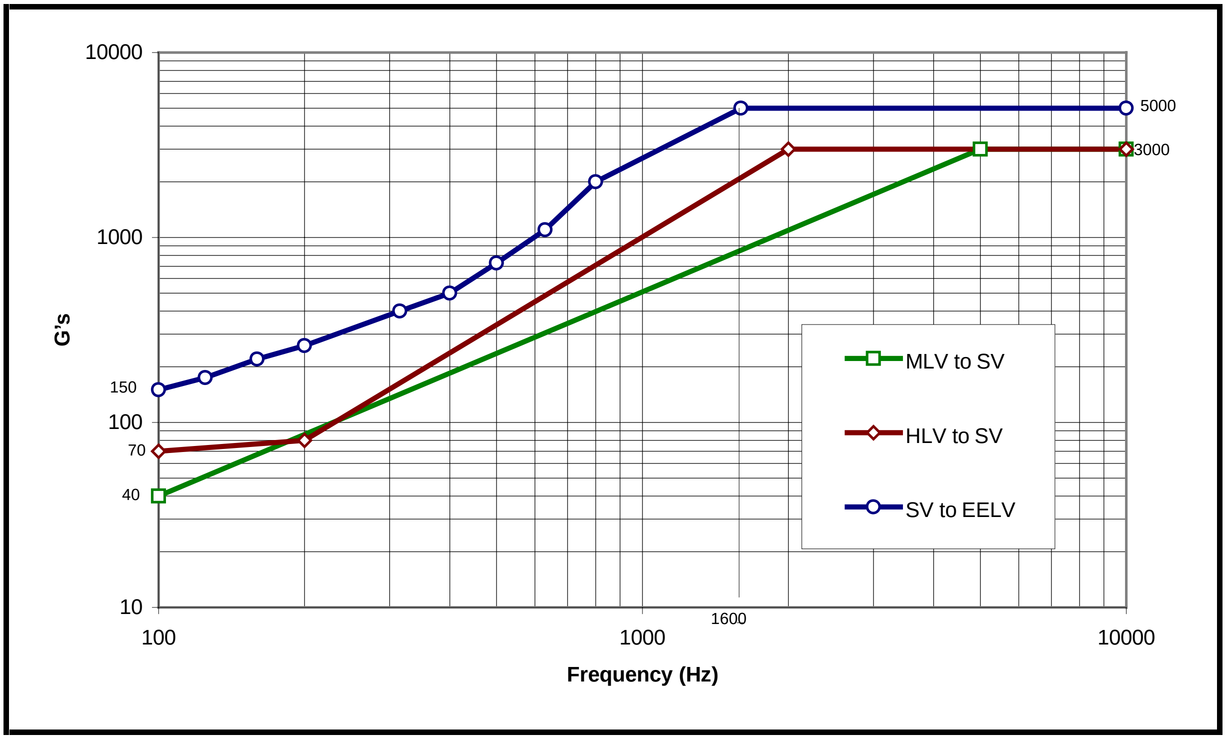

Figure 18. EELV Shock Specification

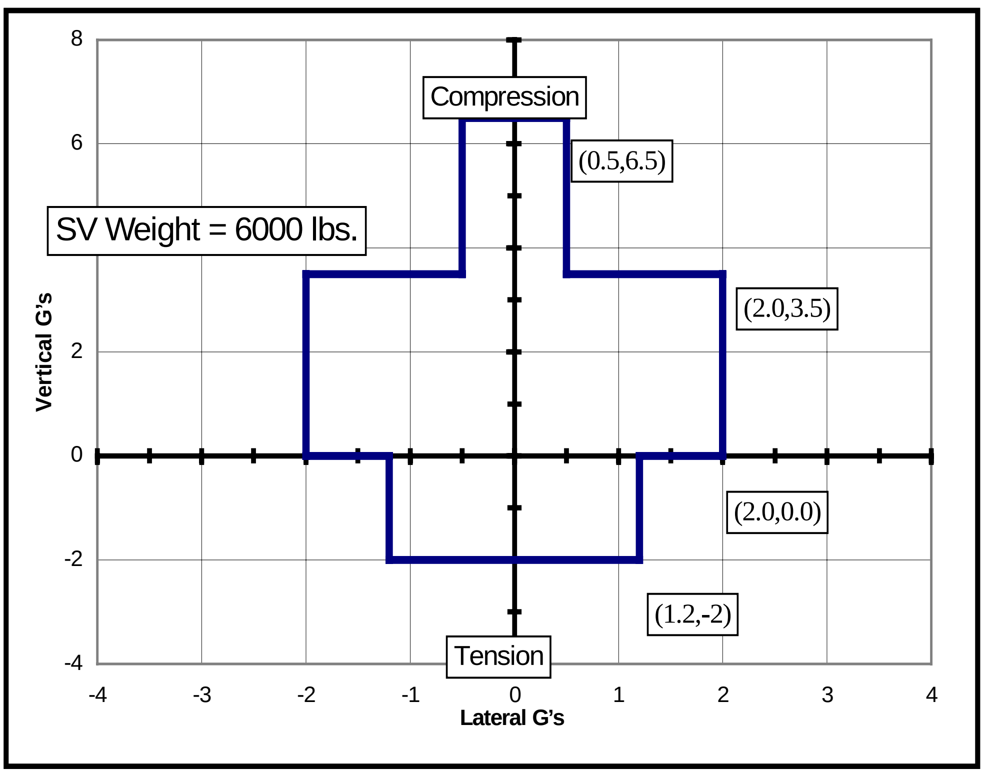

Figure 19. MLV Quasi-Static Load Factors

3.9.9 Vibration

The maximum in-flight vibration levels will be provided in the LV to spacecraft ICD, but are not defined in this document. Satellite design should be performed using the EELV acoustic data (provided in the next section).

3.9.10 Acoustics

IRD3.9.10-1
The free-field maximum predicted sound pressure levels (value at 95th percentile with a 50% confidence), from liftoff through payload deployment shall not exceed those shown in Table 11. These levels are shown graphically in Figure 20 as one-third octave band sound pressure levels versus one-third octave band center frequency for the MLV. The values shown are for a typical satellite with an equivalent cross-section area fill factor of 60%. Higher fill factors may produce higher acoustic levels.

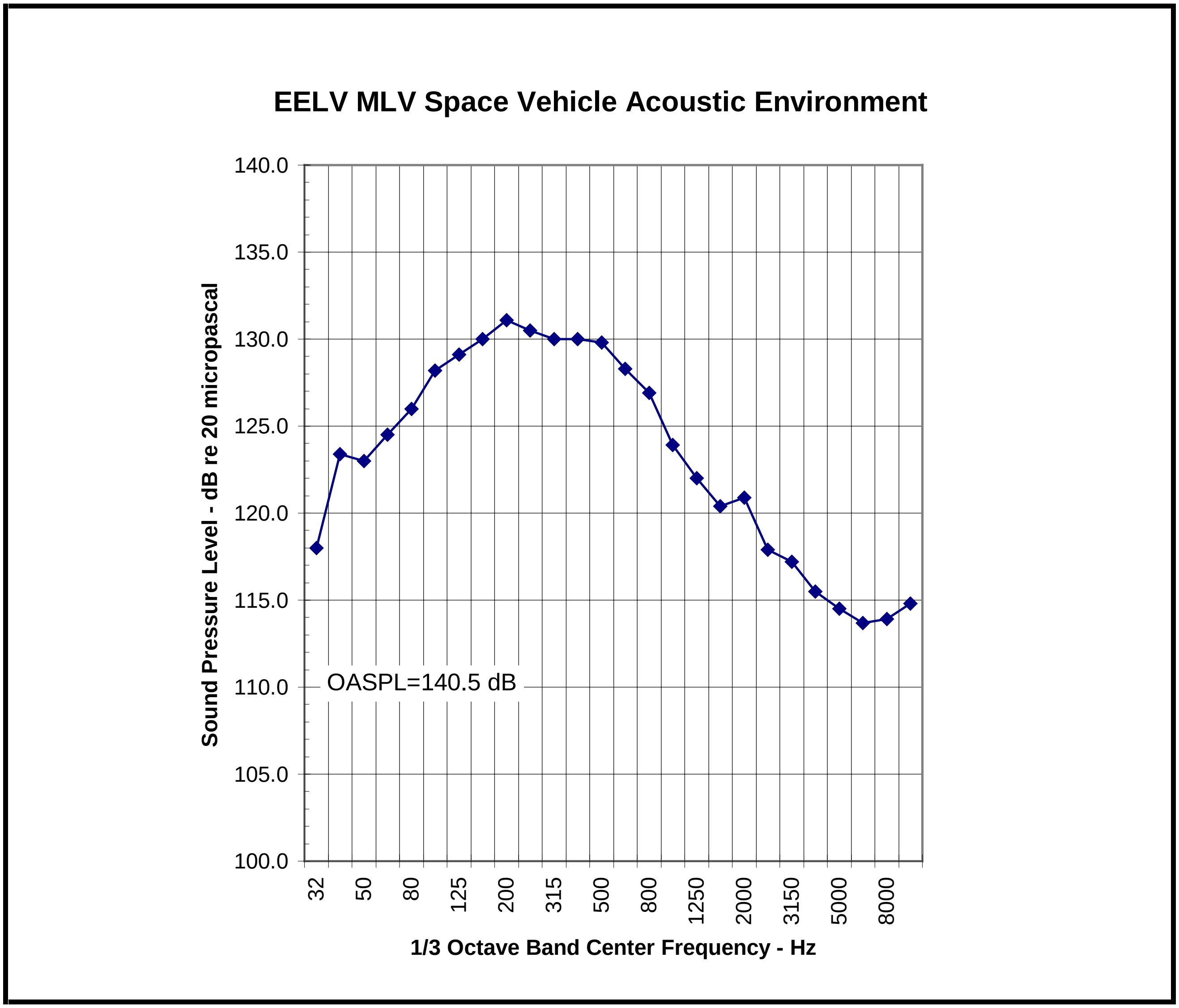

Figure 20. MLV Acoustic Levels

TABLE 11. MAXIMUM ACOUSTIC LEVELS

1/3 Octave Band Center Frequency (Hz)	MLV Satellite Sound Pressure Level (dB re 20 micropascal)
32	118.0
40	123.4
50	123.0
63	124.5
80	126.0
100	128.2
125	129.1
160	130.0
200	131.1
250	130.5
315	130.0
400	130.0
500	129.8
630	128.3
800	126.9
1000	123.9
1250	122.0
1600	120.4
2000	120.9
2500	117.9
3150	117.2
4000	115.5
5000	114.5
6300	113.7
8000	113.9
10000	114.8
OASPL	140.5

3.10 Math Model Requirements (Mechanical Section)

Thermal and structural models are required for the spacecraft and sensors to predict the thermal, structural and dynamic loads.

IRD3.10-1
All models shall use software and formats which are compatible with, or convertible to those used by the integrating systems contractor.

IRD3.10-2
Sufficient model documentation and descriptions shall be provided to permit someone familiar with similar models to make and run nominal/parametric changes to the model.

3.10.1 NASTRAN Finite Element Model

NASTRAN models are to be used to determine the structural adequacy to withstand transportation, launch and on-orbit loads.

IRD3.10.1-1
The model shall adequately represent all dynamic modes up to 100 Hz when rigidly supported at the interface.

IRD3.10.1-2
If analysis shows modes below 50 Hz, these modes shall be verified by a modal survey test.

IRD3.10.1-3
Before the model delivery, various validity checks of the NASTRAN model shall be performed: a rigid body or stiffness equilibrium check should be performed to verify that the model is not grounded; a unit gravity loading case for all three axis direction needs to be evaluated; an Eigenvalue analysis for modes up to 100 Hz. needs to be performed; the grid-point force balance and grid-point weight generator should be checked.

IRD3.10.1-4
NASTRAN bulk data deck shall be transmitted electronically or by other agreed upon means according to specified formats.

3.10.2 Thermal Math Model (Thermal Section)

For thermal control of integration of the sensor to the spacecraft, two types of thermal models a Geometric Math Model and a Thermal Math Model, are required.

IRD3.10.2-1
The sensor vendor shall develop a set of Detailed Geometric Math Model (DGMM) and a thermal balance test correlated (to within 3°C) to the Detailed Thermal Math Model (DTMM). The sensor vendor will then condense the two models respectively into a Reduced Geometric Math Model (RGMM) and a Reduced Thermal Math Model (RTMM).

The RGMM should include 50 external surfaces or less, in TRASYS or NEVADA compatible format. The RTMM should, if possible, have 50 nodes or less in a SINDA compatible format. Note that the RTMM (that is condensed from the DTMM) must still be correlated to within 3°C of the thermal balance test. The DTMM and RTMM must also have one to one nodal correspondence to the flight temperature sensors that will either report through the sensor data stream or the equipment status telemetry (EST).

IRD3.10.2-2
The sensor vendor shall provide RTMM and RGMM model predictions for the worst case hot and cold orbital/seasonal/environmental conditions for all possible sensor configurations (stowed, deployed, spinning, despun, operational, survival).

IRD3.10.2-3
The sensor vendor shall also provide pretest predictions for the spacecraft/system level thermal vacuum test. Prediction results should include all critical temperatures, heater power and margins for the same.

IRD3.10.2-4
Conservative values for contact conductance, conductivity, absorptivity, emissivity, and MLI effective emittance shall be used.

IRD3.10.2-5
All models shall be fully documented to permit ease of use by other contractors in the system.

3.11 Safety Requirements

3.11.1 Design Criteria

IRD3.11.1-1
All subsystems and interfaces shall be designed to comply with the safety requirements of EWR 127-1.

IRD3.11.1-2
The use of electro explosive devices (EED's) shall be avoided. Electro explosive devices may be used where use of such devices can be shown to reduce risk.

IRD3.11.1-3
Paraffin and other non-explosive actuators (NEA) shall be activated through the standard command and data interface, and within the sensor envelope.

IRD3.11.1-4
Dedicated EED circuits shall not be included in the baseline standard interface.

IRD3.11.1-5
Space debris shall not be generated.

4.0 TESTING PROVISIONS

IRD4.0-1
A comprehensive sensor test program as defined in MIL-STD-1540C, as tailored for the NPOESS program, shall be conducted in conjunction with the spacecraft test program to demonstrate that the sensor can meet its performance requirements and ensure that all interface requirements are satisfied.

IRD4.0-2
These interface requirements shall include interface structural and thermal loads, electrical power, electrical signals and other interface performance characteristics for ground handling, launch, deployment (where applicable), and on-orbit operations as well as for worst case systems tests conducted after delivery.

IRD4.0-3
Many of the tests will be conducted by the sensor developer before delivery of the sensors to the integrating contractor. Additional tests will be conducted at the satellite level after integration of the sensor onto the spacecraft. The allocation of tests between the sensor developer and the integrating contractor will be coordinated by the integration contractor as part of the interface control function. The coordination of testing shall include such items as sequence of tests, primary test responsibility, test levels, repetition of tests, duration of tests and test location.

IRD4.0-4
Acceptance level testing (for workmanship) shall be required on all flight articles except for the protoqual unit. The integrating contractor will verify that all required tests are completed successfully. The types of testing to be performed include:

- Thermal vacuum and Thermal cycling
- EMI/EMC characterization to understand and measure radiative and conductive emissions and susceptibility
- Static and Dynamic structural testing (including pressure vessel and ordnance testing)
- Electrical and Mechanical functional testing to demonstrate performance

4.1 Random Vibration

The random vibration test levels are dependent on the payload fairing internal acoustic environment and design of the spacecraft bus. The test levels found in Table 12 and Table 13, and Figure 21and Figure 22, are considered a conservative estimate of the random vibration environment on a representative spacecraft bus and are the minimum test levels recommended to detect workmanship defects.

IRD4.1-1
The acceptance test duration shall be 1 minute per axes.

IRD4.1-2
In no case shall the test levels for the sensor or its components be less than those shown in Table 12 and Table 13.

IRD4.1-3
The protoqualification test duration shall be 2 minutes per axis.

TABLE 12. RANDOM VIBRATION - ACCEPTANCE TEST LEVELS

Frequency	Acceleration Spectral Density (g^2/Hz)
20	0.01
20 to 160	+3 dB/oct
160 to 250	0.08
250 to 2000	-3 dB/oct
2000	0.01
Overall	7.4 g_{rms}

The plateau acceleration spectral density (ASD) level may be reduced for components between 25 Kg and 200 Kg according to the component mass (W) up to a maximum of 9 dB as follows:
dB reduction = 10 LOG(W/25)
$ASD_{(plateau)}$level = 0.08 x (25/W)
where W = component mass in Kg

IRD4.1-4
The sloped portions of the spectrum shall be maintained at ± 3 dB/oct. Therefore, the lower and upper break points, or frequencies at the ends of the plateau become:

F_L = 160 (25/W) F_L = frequency break point low end of plateau

F_H = 250 (W/25) F_H = frequency break point high end of plateau

IRD4.1-5
The test spectrum shall not go below 0.01 g^2/Hz. For components whose mass is greater than 200 Kg, the workmanship test spectrum is 0.01 g^2/Hz from 20 to 2000 Hz with an overall level of 4.4 g_{rms}.

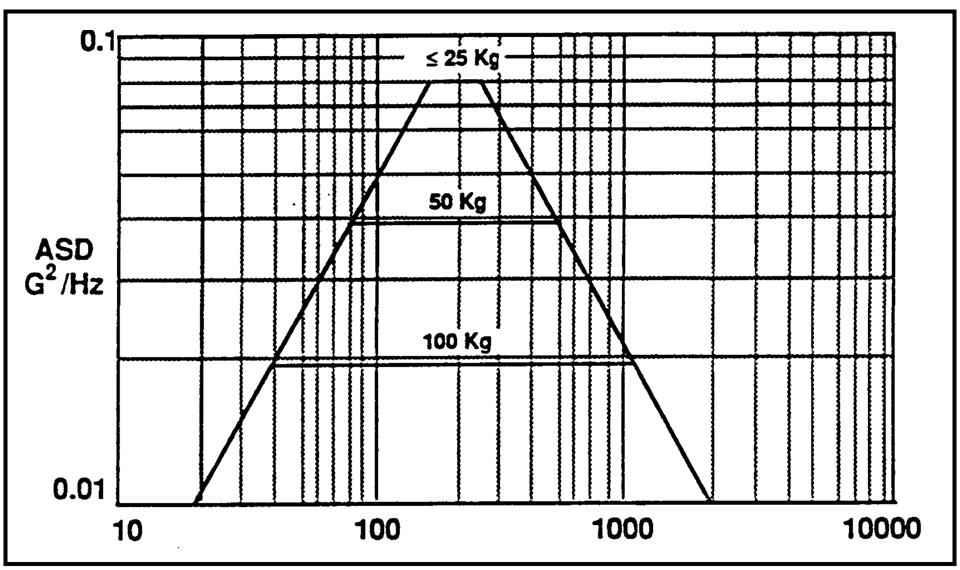

Figure 21. Random Vibration - Acceptance Levels

TABLE 13. RANDOM VIBRATION - PROTOQUALIFICATION LEVELS

Frequency	Acceleration Spectral Density (g^2/Hz)
20	0.026
20 to 50	+6 dB/oct
50 to 800	0.16
800 to 2000	-6 dB/oct
2000	0.026
Overall	14.1 g_{rms}

The acceleration spectral density (ASD) level may be reduced for components more than 25 Kg according to:
dB reduction = 10 LOG(W/25)
ASD(25 to 400) = 0.16 x (25/W)
where W = component mass in Kg

IRD4.1-6
The slope shall be maintained at ± 6 dB/Oct for components up to 65 Kg.

IRD4.1-7
Above that mass, the slopes shall be adjusted to maintain an ASD level of 0.01 g^2/Hz at 20 and 200 Hz.

IRD4.1-8
For components over 200 Kg, the test specification shall be maintained at the level for 200 Kg

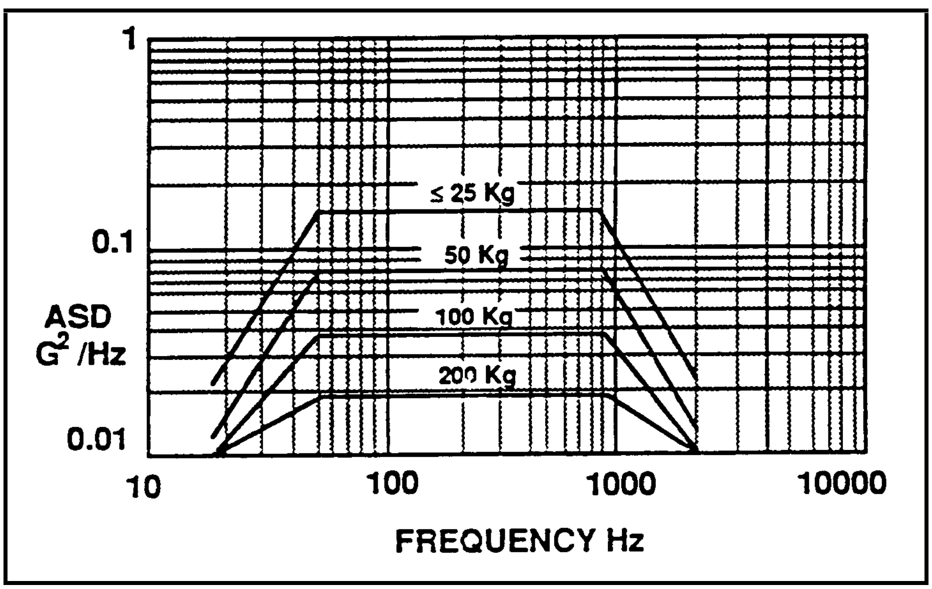

Figure 22. Random Vibration - Protoqualification Levels

4.2 Sine Vibrations

IRD4.2-1
The sensor shall be acceptance tested to the sine vibration test levels specified in Table 14 and in Figure 23 in each of three orthogonal axes.

IRD4.2-2
During this test the sensor shall be in the launch configuration.

IRD4.2-3
There shall be one sweep from 5 Hz to 50 Hz for each axis.

IRD4.2-4
The acceptance test sweep rate shall be 4 oct/min except in the frequency range of 25-35 Hz, where the sweep rate shall be 1.5 oct/min.

IRD4.2-5
For protoqual testing, the sine vibration levels shall be the same as the acceptance test levels specified in Table 14 however, the sweep rates shall be reduced by a factor of two to 2 oct/min and 0.75 oct/min respectively.

TABLE 14. SINUSOIDAL TEST LEVELS

Frequency	Amplitude/Acceleration
5 to 18 Hz	Displacement = 12 mm (double amplitude)
18 to 50 Hz	8 g_{peak}

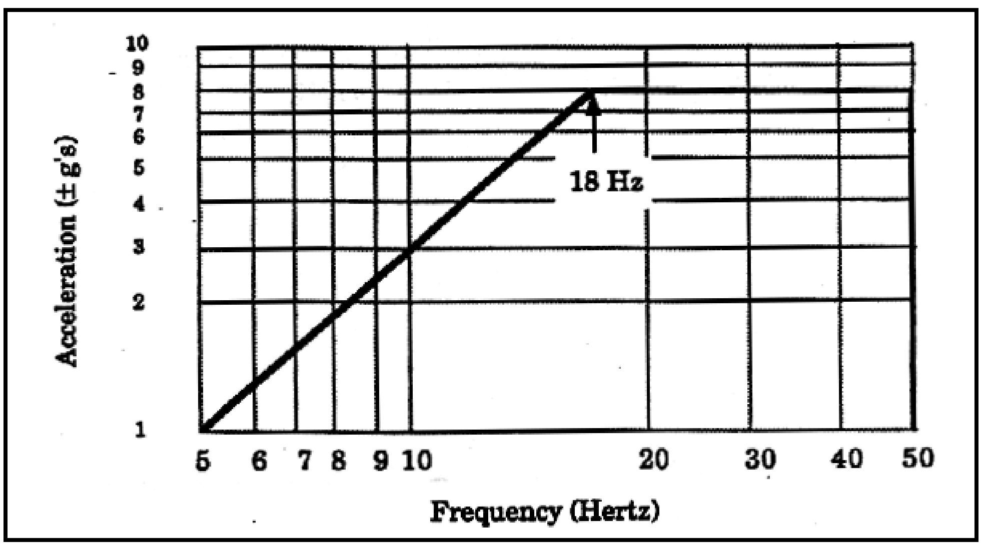

Figure 23. Sinusoidal Protoqualification Test Levels

4.2.1 Design Strength Qualification

IRD4.2.1-1
Sensors whose fundamental structural modes of vibration are demonstrated by test to be above 50 Hz shall be tested in each of three axes to loads generated by applying an ultimate factor of safety of 1.25 times the acceleration load factor obtained from the mass-acceleration curve in Figure 24.

IRD4.2.1-2
Sensors with fundamental modes of vibration below 50 Hz shall apply an additional factor of 1.5 to portions of the structure whose motion in the mode of vibration exceeds the motion of the centroid. The factor of 1.5 may be linearly varied (reduced) along the structure to 1.0 at the centroid. The loads may be applied by acceleration testing, static load testing, or vibration testing.

IRD4.2.1-3
When a coupled loads analysis of the flight configuration is available, it shall be verified that the predicted loads times 1.25 are within the tested loads.

IRD4.2.1-4
The analysis shall be based upon test verified dynamic models of the sensor for any sensors whose fundamental modes of vibration fall below 50 Hz.

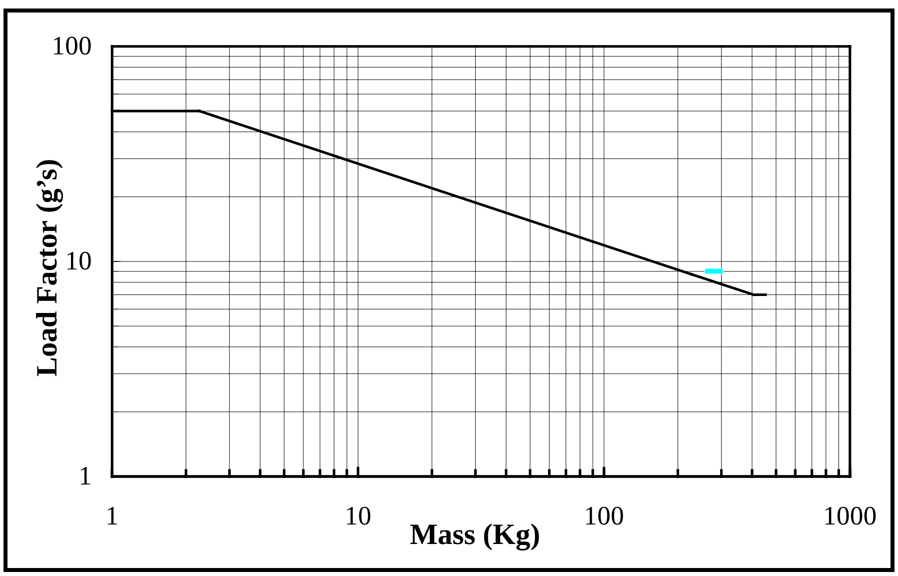

Figure 24. Acceleration Load Factors (Limit)

4.3 Acceleration

IRD4.3-1
Sensor flight hardware shall be designed to withstand a maximum acceleration of 0.015g on orbit without permanent degradation of performance.

4.4 Shock

Shock testing is required at the sensor level if there are any self induced shocks (i.e., launch lock releases, pin pullers, etc.). The testing is accomplished by actuating the device two times for each self induced shock source to account for the scatter associated with the actuation of the same device. Testing for externally induced shocks (spacecraft separation, solar array deployment, etc.) is typically accomplished at the spacecraft level.

IRD4.4-1
Sensors shall be designed and tested to survive, without permanent performance degradation, the environment shown in Figure 25 to account for externally induced shocks.

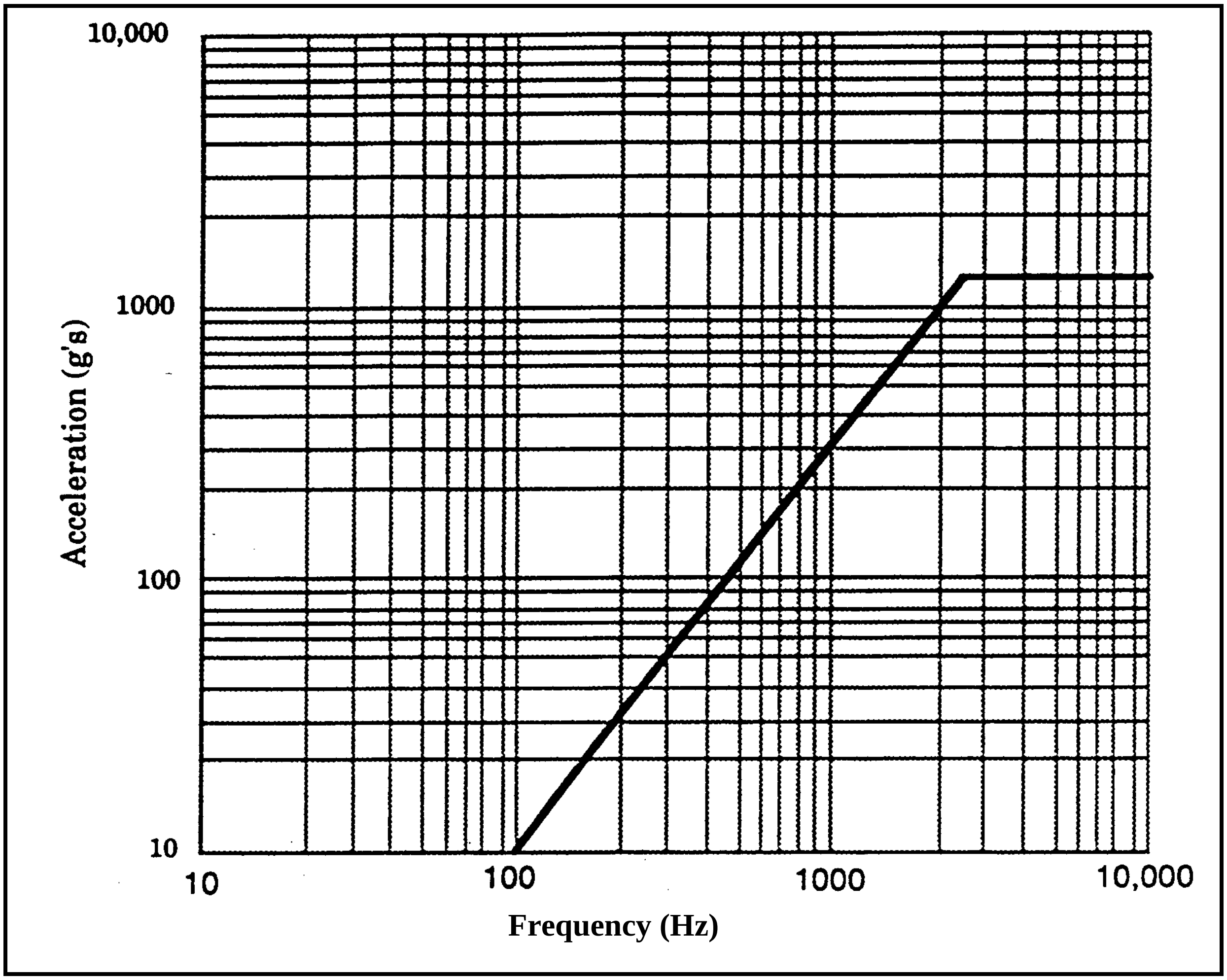

Figure 25. Shock Spectrum (Q=10)

4.5 Protoqualification Level Acoustics

IRD4.5-1
Acoustic testing shall be required for sensors with large surfaces (units with surface to mass ratio greater than 150 cm^2/Kg), which could be excited by the acoustic field directly, and for sensors greater than 180 Kg.

IRD4.5-2
The acceptance acoustics levels shall be defined in Table 15. The protoqualification levels are increased by 3 dB.

IRD4.5-3
The protoqualification test duration shall be two minutes

TABLE 15. ACCEPTANCE ACOUSTICS LEVELS

One-Third Octave Center Frequency (Hz)	Noise Level (dB) ref: 0 dB = 20μPa
25	119
31.5	119
40	123.4
50	125
63	125
80	126
100	130
125	130
160	130
200	134
250	134
315	134
400	130
500	130
630	129
800	126.9
1000	123.9
1250	123
1600	120.4
2000	120.9
2500	119
3150	117.2
4000	115.5
5000	115
6300	113.7
8000	113.9
10000	114.8
Overall	145dB
Acceptance Duration: One Minute	
Protoqualification Duration: Two Minutes	

4.6 Integrated Satellite Level Testing

IRD4.6-1
The satellite level testing shall be performed in accordance with the system level test plan. (*TBR).*

4.7 EMC/EMI

4.7.1 EMI Testing

IRD4.7.1-1
Electromagnetic testing at the system level shall be performed to verify that the interface will operate properly if subjected to conducted or radiated emissions from maximum expected external sources, and to verify that the design of the interface does not result in deleterious conducted or radiated signals that might affect other mission elements.

4.7.2 EMC Verification Requirements

The requirements in section 3.7.4 are to be verified by test in accordance with MIL-STD-462D with the exception of paragraph 3.7.4.2.3, 3.7.4.3.2 and 3.7.4.4.

IRD4.7.2-1
Paragraph 3.7.4.2.3. shall be verified by having an approved DD1494 form from the Air Force Frequency Manager.

IRD4.7.2-2
Paragraph 3.7.4.3.2 shall be verified by a combination of analysis, inspection, and demonstration.

IRD4.7.2-3
Paragraph 3.7.4.4 shall be verified by inspection.

IRD4.7.2-4
The line impedance stabilization network in MIL-STD-462D shall be tailored as necessary to be more representative of the satellite power bus impedance.

IRD4.7.2-5
The radiated emission measurement bandwidths and frequency steps in MIL-STD-462D shall be reduced as necessary to meet the requirements in paragraph 3.7.4.6.2. The existing bandwidths in MIL-STD-462D need to be reduced to show compliance with the search and rescue receiver notches. *Note*: While these requirements do not preclude the use of a shield room for testing, shield rooms do resonate causing a 15 to 25 dB amplitude error at several of the receiver frequencies in 3.7.4.6.2. This error may cause a failure where none would have been measured in an anechoic chamber.

4.7.2.1 System Verification

IRD4.7.2.1-1
The satellite electromagnetic compatibility margins defined in 3.7.4 shall be verified by a system level test in an anechoic chamber.

IRD4.7.2.1-2
Testing shall be conducted in accordance with MIL-STD-462D.

4.7.2.2 Spacecraft Charging Verification

IRD4.7.2.2-1
Testing shall be conducted in accordance with MIL-STD-1541A as tailored.

4.8 Thermal Testing

4.8.1 Thermal Vacuum Test

IRD4.8.1-1
Thermal vacuum testing of the satellite shall be performed in accordance with MIL-STD-1540C using the environmental temperature ranges specified in IRD Paragraph 3.2.4.

4.8.2 Thermal Cycle Test

IRD4.8.2-1
Thermal cycle testing of the satellite shall be performed in accordance with MIL-STD-1540C using the environmental temperature ranges specified in IRD Paragraph 3.2.4.

4.8.3 Thermal Balance Test

IRD4.8.3-1
Thermal balance testing shall be performed on the first production satellite in accordance with MIL-STD-1540C using the environmental temperature ranges specified in IRD Paragraph 3.2.4.

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5.0 NOTES

5.1 Intended Use

This document is used to establish standard NPOESS spacecraft-to-sensor interfaces and to provide guidance to sensor developers during the risk reduction and design development phases.

5.2 Definition/Glossary

Alignment Knowledge. Alignment knowledge is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor equipment and the desired orientation.

Alignment Accuracy. Alignment accuracy is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor equipment and the estimated orientation.

Alignment Stability. Alignment stability is the variation (arc-sec/time interval, peak-to-peak) in the actual orientation of the sensor equipment over specified periods of time.

High Rate Data. Refers to the real time data link to field terminals which contains all channels at the smallest scale horizontal spatial resolution (or cell size) required in Appendix D. Note that the smallest scale horizontal spatial resolution (or cell size) is the same resolution as the “regional resolution” required by the Centrals.

Key Attribute. An Environmental Data Record (EDR) attribute that is a key parameter of the system.

Key EDR. An EDR which has a key attribute.

Key Parameter. A parameter so significant that failure to meet the threshold requirement(s) pertaining to its measurement is cause for the System to be reevaluated or the program to be reassessed or terminated. Key parameters include key attributes of key EDRs and the data access requirement. Key parameter requirements are to be included in the Acquisition Program Baseline. (Equivalent to the term “Key Performance Parameter” used in the IORD)

Low Rate Data. Refers to real time data link to field terminals containing fewer channels and/or coarser resolution than the high data rate real time link.

Mission Data. The combination of data provided by any of the mission sensors (i.e. environmental data) plus satellite orbit, attitude, and time tags. It does not include other sensors (i.e. S&R, SDC) or telemetry.

Mission Sensor. Any sensor on the spacecraft directly used to satisfy any of the EDR requirements of the TRD Appendix D.

Orbital Average Power. The value of the power that occurs during normal operation, averaged over one orbit. A calculation to determine this value shall utilize at least 5-minute increments for the duration of the orbit.

Payload. Used to refer to the combination of the mission sensors and the SDC and S&R sensors carried by the spacecraft. Also used to refer to the satellite when it is still mated to the launch vehicle.

Peak Power. The value of the maximum power that occurs during normal operations.

Pointing Accuracy. Pointing accuracy is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor and the desired orientation of the sensor.

Pointing Knowledge (Real Time or Post-processed). Pointing knowledge is the angle (arc-sec, zero-to-peak) between the actual orientation of the sensor and the estimated orientation of the sensor.

Pointing Stability. Pointing stability is the variation (arc-sec/time interval, peak-to-peak) in the actual orientation of an sensor over specified periods of time.

Protoflight. A protoflight unit is one that was tested to protoqualification levels. The unit is usually the first unit fabricated.

Protoqualification. A test strategy in which qualification and acceptance tests are combined. The protoflight unit is tested to levels beyond what is expected in flight and minimum workmanship levels, with test levels and duration less than qualification levels and duration.

Satellite. The spacecraft and its sensor payloads.

Sensor. The mission-peculiar equipment or instrument to be manifested on a given space mission. The requirements specified apply to individual sensor interfaces, not the total sensor complement.

Sensor Suite. One or more sensors needed to satisfy the EDR requirements allocated to a given Sensor Requirements Document (SRD). It does not include sensors from other SRD suites which provide secondary data contributions to those EDRs.

Spacecraft. The components and subsystems which support the sensor(s) and provide housekeeping functions such as orbit and attitude maintenance, navigation, power, command, telemetry and data handling, structure, rigidity, alignment, heater power, temperature measurements, etc.

Standard Interface Plane (SIP). The SIP is the plane which defines the interface between the LV-provided and the spacecraft-provided equipment.

Tailoring. Modification of this guideline to maximize utility, considering design complexity, mission criticality, cost, and acceptable risk.

TBD. Applied to a missing requirement means that the integrating and/or sensor contractor should determine the missing requirement in coordination with the government.

TBR. The requirement will be resolved (*TBR*) between the contractors and government.

TBS. The government will clarify or supply the missing information in the course of the contract.

Transients. Short-duration changes in the power drawn by a component. Transients are a periodic and include non-recurring current surges and voltage spikes.

5.3 Acronyms

µPa	micro-Pascal
µV	micro-volt
Al	Aluminum
amp	ampere
ANSI	American National Standards Institute
arcsec	arc-second
ASD	Acceleration Spectral Density
ASTM	American Society for Testing and Materials
BC	bus controller
BTU	British thermal unit
C	Degrees Centigrade
C&DH	Command & Data Handling
C&T	Command & Telemetry
c.g.	center of gravity
C3	Command, Control, and Communications
Cal/Val	Calibration/Validation
CCD	Charge Coupled Device
CCSDS	Consultative Committee for Space Data Systems
cm	centimeter
CVCM	collected volatile condensable material
dB	decibel
dc	direct current
deg	degree
DMSP	Defense Meteorological Satellite Program
DOC	Department of Commerce
DoD	Department of Defense
DGMM	Detailed Geometric Math Model
DTMM	Detailed Geometric Model
EDR	Environmental Data Record
EED	electro explosive device
EELV	Evolved Expendable Launch Vehicle
EMC	electromagnetic compatibility
EMI	electromagnetic interference
EOS	Earth Observation System
ESD	electrostatic discharge
EST	equipment status telemetry
EUMETSAT	European Organization for the Exploitation of Meteorological Satellites
eV	electron volt
EWR	Eastern and Western Range
FED	Federal
ft	foot
g	gravity
G	Giga (1 billion)
GPS	Global Positioning System

GSE	Ground Support Equipment
hr	hour
Hz	hertz
I&T	Integration and Test
ICD	interface control document
IDP	Interface Data Processor
IOC	Initial Operational Capability
IPO	Integrated Program Office
IR	infrared
IRD	Interface Requirements Document
ISO	International Organization for Standardization
JPS	Joint Polar System
k	kilo (1000)
kΩ	kilo ohms
kbps	kilobit per second
Kg	kilogram
kHz	kilohertz
km	kilometer
kohms	kilo-ohms
LET	Linear Energy Transfer
LV	launch vehicle
M	mega (1 million)
m	meter
mA	mili ampere
MΩ	mega ohms
max	maximum
mb	millibar
Meg	Mega (1 million)
METOP	Meteorological Operational Program
MIL	Military
min	minute
MLI	multilayer insulations
MLV	medium launch vehicle
mm	millimeter
MMA	moving mechanical assembly
MOA	Memorandum of Agreement
N-m	newton-meter
N-m-sec	newton-meter-second
NASA	National Aeronautics and Space Administration
NASTRAN	NASA structural analysis
NEA	nonexplosive actuator
NOAA	National Oceanographic and Atmospheric Administration
NPOESS	National Polar-orbiting Operational Environmental Satellite System
NRZ	Non-return to zero
NVR	Non-volatile residue
OASPL	overall sound pressure level

oct	octave
OSTP	Office of Science and Technology Policy
PDD	Presidential Decision Directive
PDR	Preliminary Design Review
PLF	Payload Fairing
POES	Polar-orbiting Operational Environmental Satellite
rad	radians
RGMM	Reduced Geometric Math Model
RTMM	Reduced Thermal Math Model
rms	root-mean-square
pT	
RT	remote terminal
S&R	Search and Rescue
SDC	Surface Data Collection
sec	second
SGLS	Space Ground Link System
SI	System Internationale
SINDA	systems improved numerical differencing analyzer
SIP	standard interface plane
SOH	state of health
SRD	Sensor Requirements Document
STD	Standard
TBD	to be determined
TBR	to be resolved
TBS	To be supplied
TIROS	Television Infrared Orbital Satellite
TML	total mass loss
TRASYS	thermal radiation analysis system
TSPR	Total System Performance Responsibility
USAF	United States Air Force
USG	United States Government
UTC	universal time coordinated
UV	ultraviolet
v	volts
Vdc	volts dc
w	watts
yr	year