GSFC SOLAR CMO
April 29, 2026
Released

SOLAR-XRIS-REQ-0029, Revision A
Space Weather Next (SW Next) Program
Space weather Observations at L1 to Advance Readiness Project
Code 493

SOLAR
X-Ray Irradiance Sensor (XRIS)
Requirements Specification (SPEC)

SWO Reviewed – Not Subject to Export Control

U.S. Department of Commerce (DOC)
National Oceanic and Atmospheric Administration (NOAA)
National Environmental Satellite, Data, and Information Service (NESDIS)
National Aeronautics and Space Administration (NASA)

SOLAR X-Ray Irradiance Sensor (XRIS) Requirements Specification (SPEC) Review/Signature/Approval Page

Prepared by:
Eward Glanfield

Reviewed by:
Martin (Marty) Fraeman
Terrence (Terry) Smith
Frederick Blanchette
Charles (Hudson) DeLee
Kevin Kane
Stephen Husty
John Volmmer
Yaireska Collado Vega
John Panek
Jeffrey (Jeff) Kronnenwetter

Approved by:
James (Jim) Morrissey

Electronic Approval available on-line at: IPD TDMS -Instruments Project Division TDMS (nasa.gov)

Preface

This document is under Space weather Observations at L1 to Advance Readiness (SOLAR) configuration management (CM)-controlled document. Changes to this document require prior approval and shall be submitted in the SOLAR project Technical Data Management System (TDMS) via a Configuration Control Request (CCR) or Signature Control Request (SCoRe), along with supportive material justifying the proposed change. Changes to this document will be made via complete revision.

All of the requirements within this document assume the use of the word “shall” unless otherwise stated.

Questions or comments concerning this document should be addressed to:

SOLAR Project Configuration Management Office
Goddard Space Flight Center
Code 493
Greenbelt, MD 20771

Change History Log

Change History Log will be

Revision	Effective Date	Description of Changes
Revision -	03/16/2026	Baseline Release following the approval of SOLAR-CCR-0182
Revision A	04/24/2026	Update following the approval of SOLAR-CCR-0206

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

Deviations/Waivers Record

Section # / Rqmt.	Deviation / Waiver #	CCR #	Date Approved	Description
None.	None.	None.	None.	None.

Table of TBCs/TBDs/TBRs/TBSs

Item No.	Identifier	Section	Summary
None.	None.	None.	None.

Table of Contents

1 INTRODUCTION 1
1.1 General Information 1
1.2 Scope 1
2 APPLICABLE DOCUMENTS 2
3 CONTRACT DESCRIPTION 4
3.1 X-ray Irradiance Sensor Instrument 4
3.2 Ground Support Equipment Description 4
4 FUNCTIONAL/PERFORMANCE REQUIREMENTS 5
4.1 XRIS Performance Requirements 5
4.1.1 Solar X-Ray Irradiance Measurement Products 5
4.1.1.1 X-Ray Irradiance Band Ranges 5
4.1.1.2 X-Ray Irradiance Flux Ranges 5
4.1.1.3 X-ray Irradiance Accuracy 5
4.1.1.4 X-ray Flare Location 5
4.1.2 Refresh Rate and Latency 6
4.1.3 Field of View (FOV) 6
4.2 Resource Allocations 7
4.2.1 Mass Allocation 7
4.2.2 Physical Envelope 7
4.2.3 Operational Power Allocation 7
4.2.4 Peak Operational Power Allocation 7
4.2.5 Survival Heater Power Allocation 7
4.2.6 Telemetry 7
4.2.7 Transient Magnetic Field 8
4.2.8 Static Magnetic Field 8
4.3 XRIS Operating Modes 8
4.3.1 In-Flight Calibration 8
4.3.2 Safe Mode 9
4.3.3 Normal Operational Mode 9
4.3.4 On-Orbit Operations 9
4.3.5 Activation 9
4.4 Instrument Data Availability 10
4.5 Power 10
4.5.1 Voltage Range 10
4.5.2 Abnormal Voltages 11
4.5.3 Turn-on Current Transients 11
4.6 Electrical Grounding 11
4.6.1 Primary Power DC Isolation 11
4.6.2 Survival Power Isolation 11
4.6.3 Mechanical Contact Resistance 11
4.6.4 Grounding 12
4.6.5 Connector DC Resistance 12
4.7 Signal And Data Interfaces 12

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

4.7.1 Passive Analog Telemetry 12
4.7.2 Data Signal Interface 13
4.7.2.1 Telemetry Source Packet Format 13
4.7.2.2 Command Source Packet Format 14
4.7.3 Clock Signal Interface 14
4.7.4 Command and Housekeeping telemetry 15
4.7.5 Commands for Autonomous Functions 15
4.8 Flight Software 15
4.8.1 Flight Load Non-volatile Memory 15
4.8.2 Software Updates 15
4.8.3 Software Table Updates 16
4.8.4 Flexibility and Ease of Software Modification 16
4.8.5 Version Identifiers 16
4.8.6 Warm Restart 16
4.8.7 Memory Tests 17
4.8.8 Memory Dump 17
4.8.9 Fault Detection and Correction Control 17

5 PHYSICAL REQUIREMENTS 18
5.1 Mass Properties Accuracy 18
5.1.1 Instrument Mass Accuracy 18
5.1.2 Center of Mass Accuracy 18
5.2 Mounting 18
5.3 Coordinate System and Alignment 18

6 ENVIRONMENTAL REQUIREMENTS 20
6.1 Mechanical Factors of Safety 20
6.2 Quasi-Static Acceleration 21
6.3 Frequency Requirement 21
6.3.1 Fundamental Launch Frequencies 21
6.4 Vibration 22
6.4.1 Sinusoidal Vibration 22
6.4.2 Random Vibration 22
6.5 Shock 24
6.6 Acoustics 25
6.7 Transportation 27
6.7.1 Transportation Cleanliness 27
6.8 Pressure 27
6.8.1 Operating Pressure Range 27
6.8.2 Maximum Depressurization Rate 28
6.8.3 Launch Vehicle (LV) Environmental Control System (ECS) Impingement 28
6.9 On-Orbit Dynamic Environment 28
6.9.1 Dynamic Linear Acceleration 28
6.9.2 Dynamic Angular Acceleration 28
6.10 Thermal Requirements 28
6.10.1 Flight Interface Design Temperature Limits 29
6.11 Charged Particle Radiation Requirements 30
6.11.1 Definitions 30

6.11.2 Radiation Hardness Assurance 32
6.11.3 Charging Environment 32
6.11.3.1 Charging Environments During Magnetic Substorm 33
6.11.3.1.1 Radiation Belt Geomagnetic Sub-Storm Surface Charging 33
6.11.3.1.2 Radiation Belt Geomagnetic Sub-Storm Bulk Charging (Energetic Particle) Environment 33
6.11.3.2 L1 Surface Charging Environment 34
6.11.3.3 Solar Photon Environment 36
6.11.4 Total Ionizing Dose 42
6.11.5 Displacement Damage Dose Levels 43
6.11.6 Single Event Effects 46
6.11.7 Solar Cycle Performance 47
6.11.8 Solar Particle Event Operations 51
6.11.9 Solar Flare Survival 54

7 CONTAMINATION CONTROL 55
7.1 Surface Cleanliness Requirements 55
7.2 Contamination Generation 55
7.2.1 Particulate Generation 55
7.2.2 Molecular Contamination Generation 56
7.2.2.1 Molecular Material Restrictions 57
7.3 Vacuum Bakeouts 57
7.4 Venting Requirements 57
7.5 Reserved 58
7.6 Cleanability and Protection 58

8 DESIGN AND CONSTRUCTION REQUIREMENTS 59
8.1 Electrical 59
8.1.1 Connector Specifications 59
8.1.2 Test Interfaces 59
8.1.2.1 Spacecraft Charging Limits 59
8.2 Safety 59
8.3 Electromagnetic Compatibility 60
8.3.1 Conducted Emissions 62
8.3.1.1 Applicability of Conducted Emissions 62
8.3.1.2 CE101 – Differential Mode Conducted Emissions Limits 62
8.3.1.3 CE03 – Differential Mode Conducted Emissions Limits 62
8.3.1.4 CMBCE – Common Mode Bulk Conducted Emissions Limits 62
8.3.2 Conducted Susceptibility 63
8.3.2.1 Applicability of Conducted Susceptibility 63
8.3.2.2 CS101 – Conducted Susceptibility Limit 64
8.3.2.3 CS114 – Conducted Susceptibility Differential Mode Limit 65
8.3.2.4 CS114 – Conducted Susceptibility, Common Mode, Interconnected Power and Signal Cables 66
8.3.2.5 CS06 – Conducted Susceptibility, Transients Limits 67
8.3.2.6 CS115 - Conducted Susceptibility, Bulk Cable Injection, Impulse Excitation 68
8.3.3 Radiated Emissions 69
8.3.3.1 RE102 - Electric Field Emissions Limits 69

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

8.3.4 Radiated Susceptibility 69
8.3.4.1 RS103 - Radiated Susceptibility, Electric Field, Launch Limit 69
8.3.4.2 RS103 - Radiated Susceptibility, Electric Field, On-Orbit Limit 70
8.4 Identification and Marking 70
8.5 Workmanship 70
8.5.1 Connectors 70
8.6 Reliability and Mission Lifetime 70
8.6.1 Mission Life 70
8.6.2 Operating Time 71
8.6.3 Trouble-Free Time 71
8.7 Ground Handling 71
8.7.1 GSE Cleanliness 71
8.7.2 GSE Bakeout 71
8.7.3 Test Harness 72
9 MECHANICAL DESIGN REQUIREMENTS 73
9.1 Structural Requirements 73
9.1.1 Component Fatigue 73
9.1.2 Fracture Control Requirements 73
9.2 Fastening Systems 74
9.2.1 Factors of Safety 74
9.3 Mechanisms 74
10 LOGISTICS 75
10.1 Ground Support Equipment 75
10.1.1 Electrical System Test Equipment 75
10.1.2 Instrument Emulator 76
10.2 Transportation Equipment 78
10.2.1 Shipping Container 78
11 VERIFICATION REQUIREMENTS 79
11.1 Test Requirements 79
11.1.1 Definitions 79
11.1.2 Test Factors 80
11.1.3 Test Tolerances 81
11.1.4 Test Restrictions 81
11.1.4.1 Failure During Tests 81
11.1.4.2 Modification of Hardware 81
11.1.4.3 External Adjustment 81
11.1.4.4 Re-Test Requirements 82
11.1.4.5 Configuration 82
11.2 Required Tests 82
11.2.1 Performance Tests 83
11.2.1.1 Comprehensive Performance Test 83
11.2.1.2 Limited Performance Test 83
11.2.1.3 Abbreviated Functional Test 83
11.2.2 Mass Properties Measurement 83
11.2.3 Static Loads/Strength Test 83

11.2.3.1 Sine Burst 84
11.2.3.2 Static Pull 84
11.2.4 Sine Sweep Survey 84
11.2.5 Sine Vibration 85
11.2.6 Random Vibration 85
11.2.7 Acoustic Test 86
11.2.8 Shock 86
11.2.9 Thermal Vacuum Bake-out 87
11.2.10 Thermal Vacuum Test 87
11.2.10.1 Thermal Vacuum Test Parameters 87
11.2.10.2 Thermal Vacuum Test Profile 87
11.2.11 Magnetics Test 88
APPENDIX A ABBREVIATIONS AND ACRONYMS 89

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

List of Figures

Figure 1. Telemetry Source Packet Definition 13
Figure 2. Time Code Format 14
Figure 3. Command Source Packet 14
Figure 4. Spacecraft Time Message Packet 15
Figure 5. Spacecraft Reference System 19
Figure 6. Sine Vibration Environment 22
Figure 7. Random Vibration Environment 23
Figure 8. Candidate LV Shock Environment 25
Figure 9. SOLAR Acoustic Design Envelope 26
Figure 10. Allowable Analytical, Flight Operation, Flight Acceptance Test, and Qualification (ProtoFlight or Prototype) Test Temperature Ranges 29
Figure 11. Integral Electron Fluence vs Energy 34
Figure 12. Direction Axes Used in L1 Plasma Flux Environments 36
Figure 13. Total Ionizing Dose-Depth Curve 43
Figure 14. Displacement Damage Dose in Silicon vs. Shielding Thickness Dose-Depth Curve 45
Figure 15. Displacement Damage Dose in Gallium Arsenide Dose-Depth Curve 46
Figure 16. LET Spectrum of GCR Background 48
Figure 17. Solar LET Spectrum Background 50
Figure 18. Solar Proton Background Spectra 51
Figure 19. Peak Solar Particle Event Flux 53
Figure 20. Peak Solar Proton Flux 54
Figure 21. CE101/CE03 Differential Mode Conducted Emissions Limits 62
Figure 22. Common Mode Bulk Conducted Emissions Limit 63
Figure 23. Power Lead Conducted Susceptibility (CS101) Voltage Limit 64
Figure 24. Conducted Susceptibility Power Limit (30 Hz to 150 kHz) 64
Figure 25. Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz Limit 65
Figure 26. Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz Current Limits 66
Figure 27. Common Mode Conducted Susceptibility Calibration Limit 67
Figure 28. Conducted Susceptibility Transient Waveform 68
Figure 29. CS115 Impulse Waveform 68
Figure 30. Unit Level RE102 Radiated Electric Field Emission Limits 69

List of Tables

Table 1. Applicable Documents 2
Table 2. Factors of Safety1,2 20
Table 3. Design Limit Loads 21
Table 4. Candidate LV Shock Environment 24
Table 5. SOLAR Acoustic Design Envelope 25
Table 6. Transportation Loads 27
Table 7. Orbit Thermal Parameters 30
Table 8. Geo-magnetic Sub-storm Region Surface Charging Environment Model Parameters . 33
Table 9. Geomagnetic Sub-storm Bulk Charging (Energetic Electron) Environment 33

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

Table 10. Plasma Moments for L1 Solar Wind Environment Surface Charging 35
Table 11. Directionality for L1 Surface Charging Analysis 35
Table 12. Solar Photon Flux Density 36
Table 13. Total Ionizing Dose vs. Shielding Thickness 42
Table 14. Displacement Damage Dose in Silicon vs. Shielding Thickness 44
Table 15. Displacement Damage Dose in Gallium Arsenide vs. Shielding Thickness 45
Table 16. LET Spectrum of GCR Background 47
Table 17. Solar Background LET Spectrum 48
Table 18. Solar Proton Background Flux 50
Table 19. Abbreviate Peak Solar Particle Event Flux 51
Table 20. Peak Solar Proton Flux 53
Table 21. Lifetime Contamination Requirements 55
Table 22. Emission and Susceptibility Requirements 60
Table 23. RS103 Radiated Susceptibility Levels 69
Table 24. Test Factors and Durations 80
Table 25. Thermal Vacuum Test Parameters 87

1 INTRODUCTION

1.1 General Information

1.2 Scope

This specification describes the performance and electrical, mechanical, environmental, and verification testing requirements for a space-qualified solar X-ray irradiance sensor for the Space Weather Observations at L1 to Advance Readiness (SOLAR) Mission.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

2 APPLICABLE DOCUMENTS

The following documents and drawings in effect on the day this specification was signed shall apply to the fabrication and to the electrical, mechanical, and environmental requirements of the XRIS to the extent specified herein. Where requirements that point to these standards provide alternatives, they will be accepted. In the event of conflict between this specification and any referenced document, this specification will govern, with the exception of the SOLAR XRIS Statement of Work (SOLAR-XRIS-SOW-0007), in which case the Statement of Work takes precedence.

The following is a list of the applicable specifications and publications.

Table 1. Applicable Documents

Document Number	Title
SOLAR-XRIS-SOW-0007	SOLAR X-ray Irradiance Sensor (XRIS) Instrument Statement of Work (SOW)
SOLAR-XRIS-REQ-0031	SOLAR X-ray Irradiance Sensor (XRIS) Instrument Contract Deliverables Requirements List (CDRL)
SOLAR-SYS-REQ-0016	Radiation Hardness Assurance Requirements Document (RHARD)
ANSI/TIA/EIA-422 Revision - B	Electrical Characteristics of Balanced Voltage Digital Interface Circuits
ANSI/TIA/EIA-644 Revision - A-2001	Electrical Characteristics of Low Voltage Differential Signaling (LVDS) Interface Circuits
NFPA 70	National Fire Protection Association National Electric Code
NASA-STD-5001B	Structural Design and Test Factors of Safety For Spaceflight Hardware
NASA-STD-8719.24	NASA Expendable Launch Vehicle Payload Safety Requirements
NASA-STD-6016C	Standard Materials and Processes Requirements for Spacecraft
NASA-HDBK-7005	Dynamic Environment Criteria
NASA-STD-7001	Payload Vibroacoustic Test Criteria
IEST-STD-CC-1246E	Product Cleanliness Levels and Contamination Control Program
ASTM E-595-07	Standard Test Method for Total Mass Loss and Collected Volatile Condensable Materials from Outgassing in a Vacuum Environment
MIL-DTL-5541	Chemical Conversion Coatings on Aluminum and Aluminum Alloys
MIL-A-8625F	Anodic Coatings for Aluminum and Aluminum Alloys
EEE-INST-002	Instructions for EEE Parts Selection, Screening, Qualification, and Derating
MIL-STD-461G	Requirements for the Control of Electromagnetic Interference Characteristics of Subsystems and Equipment

GSFC-STD-7000B	General Environmental Verification Standard (GEVS) for GSFC Flight Programs and Projects
NASA-STD-5019A	Fracture Control Requirements for Spaceflight Hardware
NASA-STD-5020B	Requirements for Threaded Fastening Systems in Spaceflight Hardware
NASA-STD-5017B	Design and Development Requirements for Mechanisms
NASA-HDBK-4002A	Mitigating In-Space Charging Effects—A Guideline
FAA AC 20-71	Federal Aviation Administration Advisory Circular (AC) 20-71, “Dual Locking Devices on Fasteners”
NASM 33540	Safety Wiring, Safety Cabling, Cotter Pinning, General Practices for
SAE AS567	Safety Cable, Safety Wire, Key Washers, and Cotter Pins for Propulsion Systems, General Practices for Use of
MSFC-STD-3029A	Guidelines for the Selection of Metallic Materials for Stress Corrosion Cracking Resistance in Sodium Chloride Environments
MIL-STD-462, Notice 4	Electromagnetic Interference Characteristics, Measurement of, 1 May 1970
ISO 14644-1:2015	Part 1: Classification of air cleanliness by particle concentration
ISO 14644-3: 2005	Cleanrooms and associated controlled environments -- Part 3: Test methods
CCSDS 133.0-B-2	Space Packet Protocol, Blue Book, Issue 2, June 2020
CCSDS 301.0-B-4	Time Code Formats. Blue Book. Issue 3, January 2002
NA-GSFC-2005-05	Latent Damage Reliability Issues Due to Single-Event Latchup and High-Current Error Modes

3 CONTRACT DESCRIPTION

3.1 X-ray Irradiance Sensor Instrument

The X-ray Irradiance Sensor (XRIS) instrument is used to measure the energy power from solar radiation per unit area. This can be used to detect changes in solar activity like solar flares and help predict solar energetic particle events that can penetrate the Earth's magnetic field. It is also important to understand the X-ray input into the Earth's upper atmosphere. The instrument is essential in monitoring X-ray flares that can cause changes in the Earth's ionosphere like strong radio bursts that may interfere with radio frequencies and cause problems for satellite communication and navigation systems. Coronal Mass Ejections (CMEs) are sometimes associated with solar flares which can lead to geomagnetic storms.

The XRIS contract includes the delivery of one Flight Models (FM) instrument, one spare Flight Model instrument and one (1) Engineering Development Unit (EDU), flight harnesses between instrument boxes, if applicable, and spares, an electronics unit, and harnesses.

3.2 Ground Support Equipment Description

The XRIS contract includes the delivery of two (2) sets of the Electrical System Test Equipment (ESTE), four (4) X-ray Irradiance Sensor Emulators (XRISE), and one (1) Ground Processing Development System (GPDS). Additionally, Mechanical Ground Support Equipment (MGSE) such as lifting fixtures/handles, shipping containers, purge carts, drill templates, test fixtures, non-flight protective covers, etc. are included in the delivery. Electrical Ground Support Equipment (EGSE) includes, but is not limited to, test cables and break-out boxes as required.

Ground Support Equipment is necessary to operate the instrument during spacecraft testing. Supplying this equipment is considered part of the delivery for the instrument. More information is provided in the Contract Deliverable Requirement List (CDRL).

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

4 FUNCTIONAL/PERFORMANCE REQUIREMENTS

This section defines the functional and performance requirements for the SOLAR XRIS.

The requirements in this Specification pertain to the XRIS "system", which may include all instrument hardware, software, and ground processing algorithms. The XRIS contractor is not responsible for the operational ground system, but certain specifications will require ground processing after collection but before data distribution for which the contractor is responsible for defining and developing.

4.1 XRIS Performance Requirements

4.1.1 Solar X-Ray Irradiance Measurement Products

4.1.1.1 X-Ray Irradiance Band Ranges

XRIS-116: The XRIS **shall** measure solar X-ray irradiance through the band ranges of 0.05 – 0.4 nm and 0.1 – 0.8 nm.

4.1.1.2 X-Ray Irradiance Flux Ranges

XRIS-118: The XRIS **shall** measure solar X-ray irradiance with a flux range of $1x10^{-9}$–$1x10^{-3}$ Wm^{-2} in the 0.05 – 0.4 nm band range and $1x10^{-8}$–$4x10^{-3}$ Wm^{-2} in the 0.1 – 0.8 nm band range.

4.1.1.3 X-ray Irradiance Accuracy

XRIS-2642: The XRIS **shall** measure solar X-ray irradiance with a measurement accuracy of +/- 20%.

4.1.1.4 X-ray Flare Location

XRIS-2662: The XRIS **shall** provide the flare location for flares classification M1 and above (above 10^{-5} W/m^2) within a five (5) arcmin accuracy.

> ***Rationale:*** *The flare location measurement is essential to increase the forecasting capability of flare and solar energetic particle occurrence. Many tools and models need the flare location to provide significant warning time of a strong space weather event and to access the location and probability of a possible solar energetic particle event arrival. The location frequency also serves as a factor for the probability occurrence of the active region flaring activity. Currently, this support is needed specifically for human space exploration. It will increase the awareness and warning time needed to make critical decisions to protect the astronauts. The measurement requirements are consistent with those currently used in the GOES program.*

4.1.2 Refresh Rate and Latency

The SOLAR mission will make solar X-ray irradiance data available to SWPC forecasters within 30 seconds of the completion of each data acquisition. The SOLAR mission will provide flare location measurements with a latency of two (2) minutes or less.

XRIS-134: The XRIS **shall** have a refresh rate of the solar X-ray irradiance data acquisition and output of ≤ three (3) seconds.

XRIS-2663: The XRIS **shall** provide flare location measurement and output with a refresh rate of 60 seconds or less.

XRIS-135: The processing of a completed on-orbit data collection by the instrument and the transmission to the spacecraft **shall** contribute ≤1 second to the overall latency of the X-ray irradiance data products.

XRIS-2664: The processing of a completed on-orbit data collection by the instrument and the transmission to the spacecraft **shall** contribute ≤10 seconds to the overall latency of the X-ray flare location data products.

XRIS-136: The XRIS ground processing algorithms contribution to data latency of the XRIS algorithm outputs **shall** be ≤ 3 seconds for irradiance data products.

XRIS-4498: The XRIS ground processing algorithms contribution to data latency of the XRIS algorithm outputs **shall** be ≤ 84 seconds for flare location data products.

4.1.3 Field of View (FOV)

XRIS-138: The XRIS FOV **shall** not exceed a 15-degree half-width cone centered on the +X SRF axis (XRIS- Coordinate System is defined in Section 5.3).

> ***Rationale:*** *Allows for glint or other interference of spacecraft structures at angles larger than this.*

XRIS-139: The width of the FOV **shall** be sufficient to ensure measurement requirements can be met 99.9% of the time under normal operating conditions. See Section 5.3 for spacecraft pointing performance under normal operating conditions.

XRIS-140: The XRIS solar X-ray irradiance observation extent **shall** be from 0 – 1. 3 R_{sun}.

The spacecraft will provide the XRIS +Z URF face a 2pi steradian FOV to space, excepting a FOV blockage with less than 80 W/m^2 back loading.

The XRIS Unit Reference Frame (URF) is defined as the following:

The URF is orthogonal and right-handed

The URF is fixed to the body of the XRIS instrument

The location of the XRIS origin will be specified by the XRIS provider

The $+Y_{URF}$ axis is approximately aligned with the instrument line of sight

The $+Z_{URF}$ axis is approximately perpendicular with the instrument mounting plane

The $+X_{URF}$ axis completes the triad

4.2 Resource Allocations

4.2.1 Mass Allocation

XRIS-144: The XRIS (including mounting hardware, thermal blankets, any separate electronics units and cabling) **shall** have a mass of less than or equal to 6.0 kg. Note: This mass includes mass margin.

XRIS-145: The XRIS instrument mass margin **shall** comply with GSFC-STD-1000I Rule 1.06.

4.2.2 Physical Envelope

XRIS-147: The XRIS volume (length*width*height), including mounts, thermal blankets and connectors, for both stowed and operational configurations **shall** have dimensions that do not exceed a total volume of 11424 cm^3. These dimensions pertain to both static and dynamic envelopes of the instrument.

XRIS-148: The XRIS maximum dimension in the sun-pointing vector **shall** be less than 32 cm.

XRIS-2666: The XRIS maximum height from spacecraft interface plate **shall** be less than 21 cm.

4.2.3 Operational Power Allocation

XRIS-150: The XRIS operational power, including operational heater power, **shall** be ≤ 12. 1W.

4.2.4 Peak Operational Power Allocation

XRIS-152: The XRIS **shall** have a peak (instantaneous) operational power of ≤ 16 W for a duration of <= 20 msec at 34 V.

4.2.5 Survival Heater Power Allocation

XRIS-154: The XRIS **shall** have a survival heater power consumption of ≤ 5.5 W.

4.2.6 Telemetry

XRIS-156: The XRIS instrument science and engineering data rate, including all overhead associated with Consultative Committee for Space Data Systems (CCSDS) packetization by the

instrument at the spacecraft interface, **shall** not exceed 30.0 kilobits per second (kbps) over any one second period.

The Spacecraft-to-instrument Interface Control Document (ICD) will document peak data rate and average data rate for each instrument mode.

4.2.7 Transient Magnetic Field

XRIS-158: The XRIS instrument **shall** limit varying magnetic field less than 5 Hz to less than 4 nT zero-to-peak in any axis 1 meter from any face of the unit for any operating mode and changing configurations.

> ***Rationale:*** *Derived from a magnetic cleanliness budget and based on magnetic characteristics of similar instruments.*

4.2.8 Static Magnetic Field

XRIS-160: The XRIS units **shall** limit the static magnetic field to less than 40 nT in any axis 1 meter from any face of the unit for any operating mode.

> ***Rationale:*** *Derived from a magnetic cleanliness budget and based on magnetic characteristics of similar instruments.*

4.3 XRIS Operating Modes

XRIS-162: The XRIS current operating mode **shall** be identified in its telemetry stream.

> ***Rationale:*** *Common best practices deem that being able to quickly identify an instrument's operating mode is critical for safe on-orbit operations.*

4.3.1 In-Flight Calibration

XRIS-164: The XRIS **shall** provide an in-flight calibration mode that provides a test input to support calibration and testing both on the ground and in space.

> ***Rationale:*** *Common best practices for trending/identifying any functional/performance issues with the electronics of the instrument.*

XRIS-166: The XRIS in-flight calibration **shall** be both self-terminating and able to be terminated by a ground command.

> ***Rationale:*** *To ensure health and safety of the instrument; if one cannot terminate in-flight calibration sequence, the instrument cannot return to the normal operations mode.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

4.3.2 Safe Mode

XRIS-174: The instrument **shall** provide a Safe Mode. Safe mode is an operating mode of the instrument during which all non-essential subsystems are shut down and only essential functions are active. Safe mode represents a configuration in which the instrument is thermally, mechanically, and optically safe without receiving commands from the Spacecraft.

> ***Rationale:*** *To provide a mode where the instrument is in thermally, mechanically, and optically "safe" configuration without receiving commands from the Spacecraft. This mode supports the Observatory Safe Mode strategy.*

XRIS-177: The instrument **shall** be capable of remaining in a safe configuration for at least 168 hours without ground intervention.

> ***Rationale:*** *To ensure survivability of the instrument should an extended safe mode configuration be required or due to any ground outages.*

4.3.3 Normal Operational Mode

XRIS-185: In normal operational mode, the XRIS **shall** be in a fully functional configuration in which designed measurements are made in accordance with the performance requirements listed in section 4.1 of this document and are available to be sent to the spacecraft for downlink.

> ***Rationale:*** *This is the mode in which the instrument will be operated most of the time.*

4.3.4 On-Orbit Operations

XRIS-189: The XRIS **shall** continuously operate during all spacecraft maneuvers.

> ***Rationale:*** *Simplifies on-orbit operations if no special configurations are required for spacecraft maneuvers. Minimizes any impact to instrument performance due to thermal variations if changing configurations.*

XRIS-2667: XRIS **shall** survive any spacecraft orientation.

XRIS-190: XRIS **shall** meet all performance requirements within 300 seconds after the spacecraft interface has returned to being within specification following spacecraft maneuvers.

4.3.5 Activation

XRIS-192: The instrument **shall** require no active commanding prior to fourteen (14) days after launch.

> ***Rationale:*** *The 14 days allows time for the Spacecraft to be fully checked out.*
>
> *To support ground on-orbit anomaly troubleshooting.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

4.4 Instrument Data Availability

XRIS-197: The XRIS **shall** limit on-orbit calibrations to no more than four (4) calibrations per year with each calibration lasting no more than 30 minutes.

XRIS-198: The instrument **shall** be capable of maintaining nominal observation mode without ground interaction for a minimum of 4 days.

XRIS-199: The instrument **shall** be capable of re-scheduling planned instrument events that occur during NWS-declared Critical Space Weather Days.

> ***Rationale:*** *Once a Critical Space Weather Day is declared by NWS it is expected that any pre-planned instrument activities (e.g., calibration maneuvers) will have the flexibility to be re-scheduled without impacting instrument performance. L2RD-239 requires that the sum of daily data gaps for in situ Solar Wind HAP be less than 5 min and any instrument's pre-planned event has the potential to impact Solar Wind HAP acquisition. The expected cadence and duration of NWS-declared Critical Space Weather Days could last between 3 and 21 consecutive days.*

XRIS-200: The XRIS **shall** provide a 99% or greater probability of providing all of the solar X-ray Irradiance data during S4 class solar radiation storms or solar flare conditions including class X50, verifying through analysis using the "Worst Week" October 1989 model in CREME96 or equivalent.

> ***Rationale:*** *The CREME96 model includes the October 1989 solar particle event worst week model that conservatively bounds the single event effects environment in S4 and X50 solar events. It is expected that the probability of occurrence of outages as well as Bit Error Rates (BER) due to SEE will be used in the analysis that verifies this probability. Further definition of the worst week model can be found here:* *https://creme.isde.vanderbilt.edu/CREME-MC/help/solar-energetic-particle-environment*

4.5 Power

The spacecraft will supply one operational power bus for normal instrument operation.

The spacecraft will supply one survival power bus for the instrument survival heaters.

4.5.1 Voltage Range

XRIS-205: The instrument **shall** operate over the operational bus voltage range of +28 to +34 VDC at the primary power inputs for all expected load conditions (except when turned off).

XRIS-206: The instrument survival heaters **shall** operate over the voltage range of +26 to +34 VDC at the power inputs for all expected load conditions.

> ***Rationale:*** *The spacecraft power services required to be between +28 to +34 Volts DC. This is based on an unregulated lithium-ion battery.*

4.5.2 Abnormal Voltages

XRIS-208: The instrument **shall** survive without performance degradation after indefinite exposure to an anomalous voltage range of 0 to +40 VDC.

> ***Rationale:*** *During a spacecraft power circuit failure, the instrument operational power bus could experience an overvoltage transient until the spacecraft turns off the faulty circuit. The overvoltage transient to +40 V peak assumes the high end of the dc voltage range on the spacecraft bus.*

4.5.3 Turn-on Current Transients

XRIS-216: The instrument **shall** meet its performance requirements without degradation after exposure to a turn-on input voltage (0 – 34 VDC) with a rise time of 500µs.

> ***Rationale:*** *This is required to be compatible with the power service requirements of the spacecraft.*

Note: The spacecraft bus will not drop below 26 V during instrument turn on.

4.6 Electrical Grounding

4.6.1 Primary Power DC Isolation

XRIS-220: The instrument primary power interfaces, primary power and primary power returns, **shall** be isolated from the unit chassis by a DC resistance of greater than or equal to 1 Megaohms.

> ***Rationale:*** *Separating the instrument power ground from the instrument chassis will prevent the instrument power supply noise current from flowing into the instrument secondary grounds.*

4.6.2 Survival Power Isolation

XRIS-222: The instrument survival heater power interfaces **shall** be isolated from the unit chassis by a DC resistance of greater than or equal to 1 Megaohms.

> ***Rationale:*** *Returning the instrument survival heater power returns to the spacecraft single point ground and isolating the survival heater power returns from the instrument chassis helps control common mode noise current and minimize magnetic field effects due to ground loops.*

4.6.3 Mechanical Contact Resistance

XRIS-228: The DC resistance of the mechanical contact between two conductive mating surfaces (internal to the unit) **shall** be less than or equal to 2. 5 milliohm DC resistance.

Rationale: *Low resistance contacts are essential for maintaining a zero potential ground.*

4.6.4 Grounding

XRIS-230: The DC resistance between the XRIS chassis and the observatory chassis **shall** be ≤ 2. 5 milliohms.

Rationale: *Low resistance contact between instrument chassis to spacecraft structure is essential for maintaining a zero potential ground.*

XRIS-231: The XRIS **shall** provide two ground lugs for two grounding straps to be attached from the XRIS chassis for connection to the spacecraft conductive structure.

Rationale: *This provides a method/location for tying the instrument ground to the spacecraft conductive structure. Per XRIS-2668, the instrument needs to be thermally independent from the spacecraft.*

XRIS-232: The grounding lug location on the XRIS instrument chassis or the tie points in contact with the ground strap will be as defined in the Mechanical Interface Control Drawing (MICD). The ground lug contact area must remain free of any material finish that may affect the reliability of the ground connection and will be shown in the MICD.

Rationale: *Minimum contact area is a best practice for maintaining integrity of the ground.*

4.6.5 Connector DC Resistance

XRIS-234: XRIS connector backshells **shall** be electrically connected to Electronic Unit chassis with a DC resistance ≤ 2.5 milliohms.

Rationale: *Low resistance contacts are essential for maintaining a zero potential ground.*

4.7 Signal And Data Interfaces

4.7.1 Passive Analog Telemetry

XRIS-237: The XRIS **shall** require no more than two analog signals to monitor critical temperature points when the instrument is powered off.

Rationale: *The Spacecraft will be able to monitor health and safety of the instrument if the instrument is in any configuration other than the normal operational configuration, e.g. safe or survival mode. Two signals is also deemed reasonable given the complexity of the instrument.*

4.7.2 Data Signal Interface

The spacecraft will provide an interface for sending commands to the instrument and will provide a receiving interface for instrument telemetry. The spacecraft will also provide an output line for a Pulse Per Second (PPS) time signal.

XRIS-240: The instrument data transfer interface to or from the spacecraft **shall** either use a serial Universal Asynchronous Receiver Transmitter (UART) using ANSI/TIA/EIA-422-B Electrical Characteristics of Balanced Voltage Differential Interface Circuits, or ANSI/TIA/EIA-644-A-2001 Electrical Characteristics of Low Voltage Differential Signaling (LVDS) Interface Circuits.

4.7.2.1 Telemetry Source Packet Format

XRIS-242: The instrument **shall** transfer all data to the spacecraft using the CCSDS 133. 0-B-2 Section 4.1 Protocol Data Unit definition shown in the Telemetry Source Packet Definition (Figure 1).

> ***Rationale:*** *CCSDS is a data format interface standard developed by NASA to create commonality to simplify data processing and routing.*

PRIMARY HEADER							SECONDARY HEADER		DATA
PACKET IDENTIFICATION				PACKET SEQUENCE CONTROL					
VERSION NUMBER	TYPE	SEC. HDR FLAG	APPLICATION PROCESS ID	SEQUENCE FLAGS	PACKET SEQUENCE COUNT	PACKET LENGTH	TIME CODE	SPARE	VARIABLE
3 bits	1 bit	1 bit	11 bit	2 bits	14 bits	16 bits	72 bits	8 bits	
							10-8192 octets		

Figure 1. Telemetry Source Packet Definition

XRIS-245: The instrument telemetry Source packets **shall** be variable length with a maximum data zone of 8192 octets including Secondary Header.

> ***Rationale:*** *CCSDS command pack definition and SWSRD 636*

XRIS-246: The instrument **shall** set the telemetry source packet Secondary Header Flag to the value 1.

> ***Rationale:*** *CCSDS 133.0-B-1 Space Packet Protocol, Blue Book, Issue 1, September 2003*

XRIS-249: The instrument **shall** set the Telemetry Source Packet Time Code per CCSDS 301.0-B-4 Time Code Formats, Day Segmented format in the Time Code Format Figure 2.

> ***Rationale:*** *CCSDS is a data format interface standard developed by the Consultative Committee for Space Data Systems (CCSDS) with input from NASA to create commonality to simplify data processing and routing.*

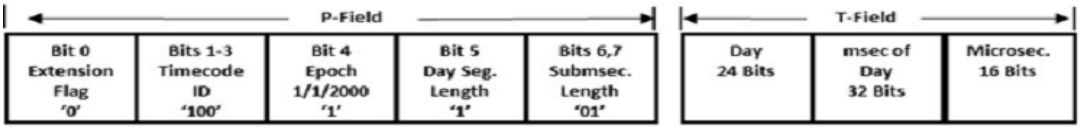

Figure 2. Time Code Format

Note: The P-Field is implied and not included in the actual time message.

The spacecraft on-board reference time will be Coordinated Universal Time (UTC).

4.7.2.2 Command Source Packet Format

XRIS-255: The instrument **shall** receive all data from the spacecraft formatted per CCSDS 133.0-B-2 Section 4.1 Protocol Data Unit definition shown in the Command Source Packet Definition Figure 3.

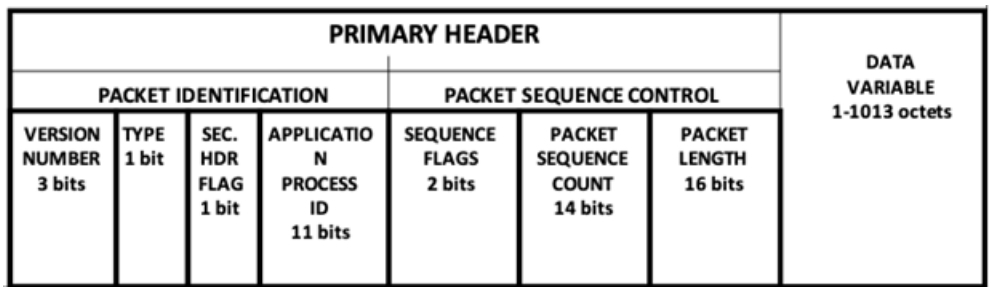

Figure 3. Command Source Packet

XRIS-258: The instrument Command Source Packets **shall** be variable length with a maximum data zone of 1013 octets.

> ***Rationale:*** *CCSDS 133.0-B-1 Space Packet Protocol, Blue Book, Issue 1, September 2003*

4.7.3 Clock Signal Interface

XRIS-263: The instrument will receive a Pulse Per Second (PPS) time pulse from spacecraft clock line by the interface defined in section 4.7.2 of this document.

> ***Rationale:*** *RS-422 and LVDS are two of the common physical interfaces used on spacecraft. The NOAA RFI responses indicate most spacecraft vendors can support one or the other or both.*

The Spacecraft will maintain Spacecraft time correlation to within +/- 0.95 sec of UTC and will provide the instrument a 1 PPS with a time code message.

XRIS-265: The instrument will receive from the spacecraft a time code message on the data line as defined in Spacecraft Time Message Packet Figure 4 Spacecraft Time Message Packet. The time code message is the time applicable to receipt of the 1 PPS.

Rationale: *CCSDS 133.0-B-1 Space Packet Protocol, Blue Book, Issue 1, September 2003*

PRIMARY HEADER							SECONDARY HEADER	
PACKET IDENTIFICATION				PACKET SEQUENCE CONTROL				
VERSION NUMBER	TYPE	SEC. HDR FLAG	APPLICATION PROCESS ID	SEQUENCE FLAGS	PACKET SEQUENCE COUNT	PACKET LENGTH	TIME CODE	SPARE
"0"d	"1"b	"1"b	"1"d	"11"b		"9" d		
3 bits	1 bit	1 bit	11 bits	2 bits	14 bits	16 bits	72 bits	8 bits

Figure 4. Spacecraft Time Message Packet

4.7.4 Command and Housekeeping telemetry

XRIS-269: The instrument **shall** provide command and housekeeping telemetry functions in all powered modes.

Rationale: *Housekeeping telemetry are data required to monitor instrument operation, health, and safety. Command functions may be required to maintain the health and safety of the instrument.*

4.7.5 Commands for Autonomous Functions

XRIS-271: The instrument **shall** execute commands to individually enable and disable each autonomous function.

Rationale: *Allows flexibility to maintain the health and safety of the instrument during on-orbit operations.*

4.8 Flight Software

4.8.1 Flight Load Non-volatile Memory

XRIS-282: The instrument flight software image **shall** be contained in its entirety in non-volatile memory at launch.

Rationale: *Allows for power-cycling without loss of the stored flight software image.*

4.8.2 Software Updates

XRIS-284: The flight software modules **shall** be reprogrammable.

Rationale: *Provides flexibility should operational changes be required following an anomaly, as the instrument ages, etc.*

4.8.3 Software Table Updates

XRIS-286: Instrument configuration data (e. g. , Table Loads or Configuration Parameters) **shall** be reprogrammable during integration and test phases and on-orbit without computer restart.

Rationale: *Common best practices dictate this capability to avoid unnecessary power cycles of the instrument.*

XRIS-287: Modified instrument configuration data (e. g. , Table Loads or Configuration Parameters) **shall** be committed to operational use by ground command.

Rationale: *Common best practice to ensure "a person in the loop" before making a configuration change.*

4.8.4 Flexibility and Ease of Software Modification

XRIS-290: Instrument configuration data (e.g., Table Loads or Configuration Parameters) **shall** be referenced such that data can be loaded and dumped by the ground without reference to memory address.

Rationale: *Common best practices dictate this capability for ease of operation and to reduce risk of loading to the wrong memory address.*

XRIS-291: The definition of instrument commands within the ground database (excluding memory diagnostic/load commands) **shall** not be dependent on physical memory addresses within the flight software.

Rationale: *To avoid the need to rebuild the command and telemetry database every time the flight software is recompiled; modifying the code results in address changes leading to command changes and then re-testing commands.*

4.8.5 Version Identifiers

XRIS-293: The instrument software and firmware versions **shall** each have an internal identifier (embedded in the executive program) that can be included in the instrument engineering data.

Rationale: *Common best practice to ensure that knowledge of version is always accessible.*

4.8.6 Warm Restart

XRIS-296: The instrument flight software **shall** provide a restart by command with preservation of instrument configuration data and memory tables.

4.8.7 Memory Tests

XRIS-298: The instrument flight software **shall** provide a mechanism to verify the contents of all memory areas.

4.8.8 Memory Dump

XRIS-300: The instrument flight software, and associated on-board computer hardware, **shall** provide the capability to dump any memory location.

XRIS-301: The flight software memory dump capability **shall** not disturb normal operations and instrument data processing.

4.8.9 Fault Detection and Correction Control

XRIS-303: If applicable, the instrument **shall** provide the capability to enable and disable any internal Fault Detection and Correction (FDC) features.

Rationale: *Allows flexibility in the on-orbit operations.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

5 PHYSICAL REQUIREMENTS

5.1 Mass Properties Accuracy

5.1.1 Instrument Mass Accuracy

XRIS-314: The mass of the instrument **shall** be measured to within ±0.1 kg.

Rationale: *Defines an allowable tolerance on instrument unit weight measurements.*

5.1.2 Center of Mass Accuracy

XRIS-318: The instrument center of mass **shall** be determined to within ±5 mm relative to a reference that will be defined in the MICD.

Rationale: *Defines an allowable tolerance on instrument center of mass measurements.*

5.2 Mounting

XRIS-2058: Instrument mounting **shall** not require access from inside the spacecraft.

Rationale: *To simplify installation and reduce risk for damage.*

5.3 Coordinate System and Alignment

XRIS-327: The instrument **shall** use the spacecraft on-orbit coordinate system which uses a right-hand orthogonal, body-fixed XYZ coordinate system, Spacecraft Reference Frame (SRF), as follows: the +X-axis is aligned with the nominal direction of the spacecraft to Sun vector pointing towards the Sun, the +Y-axis nominal to normal of the ecliptic plane and pointing to celestial North, and the Z-axis is obtained from the right-hand rule. The roll, pitch and yaw rotations are defined about the X, Z, and Y axes, respectively. The origin of the coordinate system is at the center of the Launch Vehicle separation ring attachment plane. See Figure 5 Spacecraft Reference System.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

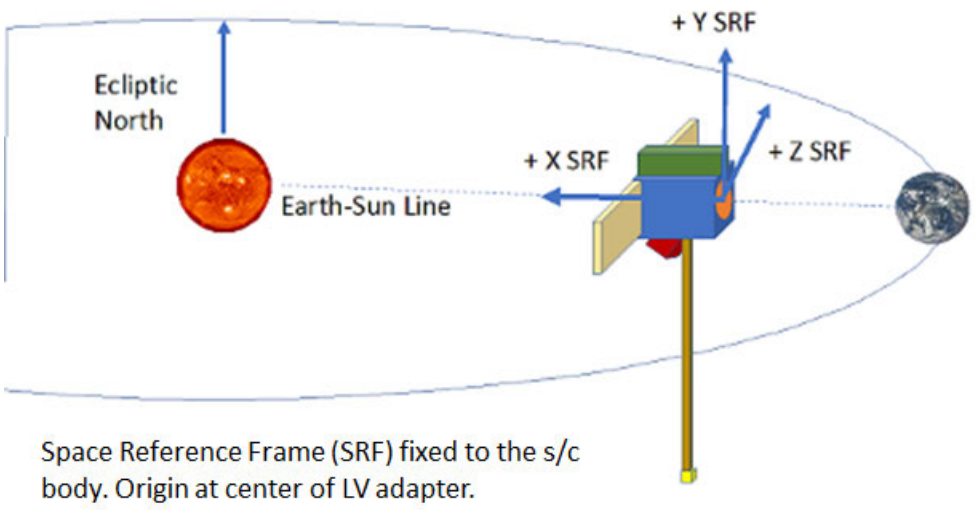

Figure 5. Spacecraft Reference System

The spacecraft will physically align the XRIS axes to within ±0.25 degrees of the direction specified in section 4.1.3 and provide alignment knowledge of the sensor axes to within ±0.1 degrees of the direction specified in section 4.1.3.

XRIS-331: The instrument **shall** have clear fiducial marks to enable alignment during integration with spacecraft.

> ***Rationale:*** *Required for accurate mechanical alignment during integration onto the spacecraft.*

The instrument unit mounting frame is an orthogonal reference frame defined by the locations of the spacecraft side of the instrument unit mounting points. A rigorous definition of this frame will be documented in the ICD.

6 ENVIRONMENTAL REQUIREMENTS

Environmental design requirements for the instrument are specified in this section.

XRIS-335: The instrument **shall** meet its performance requirements in section 4 after exposure to the environments specified in this section.

> ***Rationale:*** *Required to demonstrate that the instrument will survive launch, transportation, storage, and on-orbit environments.*

6.1 Mechanical Factors of Safety

XRIS-337: The instrument flight hardware **shall** demonstrate positive Margins of Safety under limit loads for all yield and ultimate failures using the Factors of Safety (FS) defined in Table 2. Margin of Safety (MS) is defined as follows:

> ***Rationale:*** *The yield factor of safety ensures that a prototype or protoflight test can be conducted with low risk of the hardware experiencing detrimental yielding. The ultimate factor of safety provides adequate separation between yield and ultimate failure modes and ensures that the hardware will not experience an ultimate failure under expected loading conditions.*

MS = (Allowable Stress (or Load) / (Applied Limit Stress (or Load) x FS)) -1

Table 2. Factors of Safety[1,2]

Type	Static	Sine	Random/Acoustic[4,5]
Metallic Yield	1.25[3]	1.25	1.6
Metallic Ultimate	1.4[3]	1.4	1.8
Stability Ultimate	1.4	1.4	1.8
Beryllium Yield	1.4	1.4	1.8
Beryllium Ultimate	1.6	1.6	2.0
Composite Ultimate	1.5	1.5	1.9
Bonded Inserts/Joints Ultimate	1.5	1.5	1.9

1 Factors of safety for pressurized systems to be compliant with AFSPCMAN 91-710 (Range Safety).
2 Factors of safety for glass and structural glass bonds specified in NASA-STD-5001
3 If qualified by analysis only, positive margin must be shown for factors of safety of 2.0 on yield and 2.6 on ultimate. See section 2.4.1.1.1 GSFC-STD-7000B
4 Factors shown should be applied to statistically derived peak response based on RMS level. As a minimum, the peak response **shall** be calculated as a 3-sigma value.
5 Factors shown assume that qualification/protoflight testing is performed at acceptance level plus 3dB. If difference between acceptance and qualification levels is less than 3dB, then above factors may be applied to qualification level minus 3dB instead of analyzing to acceptance level.

XRIS-383: Primary and secondary structure comprised of composite materials, Beryllium, bonded joints and/or bonded inserts **shall** be proof tested to 1.25 x Limit Load; qualification by

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

analysis only is not acceptable. Actual flight hardware testing is preferred, but testing of representative sets of hardware with a similar qualification argument can be used if approved by the NASA/GSFC COR.

> ***Rationale:*** *In order to account for uncertainties in workmanship, material properties and local stress levels special screening/criteria is required per NASA-STD-5001.*

6.2 Quasi-Static Acceleration

Quasi-static acceleration represents the combination of steady-state accelerations and the low frequency mechanically transmitted dynamic accelerations that occur during launch.

XRIS-386: The instrument **shall** be designed to withstand the quasi-static design limit loads defined in the mass-acceleration curve (MAC) shown in Table 3 without damage or degradation of performance and are to be applied one axis at a time. The design loads shown below will be updated based on the results of coupled loads analysis.

Linear interpolation should be used between breakpoints to determine the appropriate limit load as a function of instrument weight. Note that these design limit loads are intended to cover only the low frequency launch environment and must be used in conjunction with the random vibration environments to assess structural margins.

Table 3. Design Limit Loads

Mass (kg)	Accel (g)
1 or less	51.3
5	42.4
10	36.6
20	30.3
40	24.2
60	20.9
80	18.7
100 or greater	18.1

6.3 Frequency Requirement

6.3.1 Fundamental Launch Frequencies

XRIS-419: The instrument **shall** have a fundamental frequency greater than 100 Hz when hard mounted at its spacecraft interface. Any unit which fails to meet the specified fundamental frequency must supply a finite element model, correlated to modal survey test results or measured responses from vibration testing, so that it accurately represents the frequency and response amplitudes from the test. Requirements for the submitted finite element model are shown in the CDRL.

6.4 Vibration

6.4.1 Sinusoidal Vibration

XRIS-422: The instrument **shall** undergo qualification, protoflight or acceptance (level depends on qualification status or qualification approach for the unit) sine vibration testing in all three axes at the levels shown in Figure 6. Instruments with a first mode greater than 150 Hz can be exempted from sine vibration testing. upon approval by the NASA/GSFC COR. See Section 11.1.2 for definitions of Protoflight, Qualification, and Acceptance

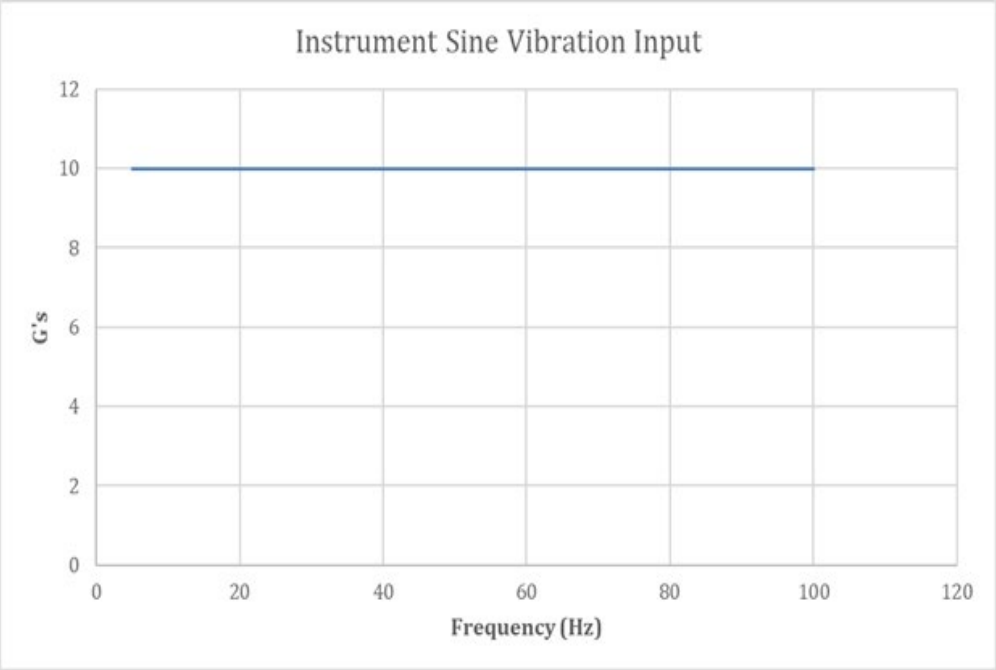

Figure 6. Sine Vibration Environment

Levels may be notched to not exceed 1.25 times the design limit load outlined in section 6.2. Peak levels at the low end of the frequency range (5 – 20 Hz typically) may be ramped up as needed to accommodate shaker table displacement limitations.

6.4.2 Random Vibration

XRIS-427: The instrument **shall** demonstrate its ability to meet its performance requirements after being subjected to the random vibration environment in Figure 7 for units weighing 22.7 kg (50 lb.) or less, applied at the Spacecraft to instrument interface.

> ***Rationale:*** *Test as you fly. GEVS standard to verify performance after (or during) environmental tests if component/ subassembly or assembly is on during launch.*

Frequency (Hz)	ASD Level (g^2/Hz)	
	Qualification	Acceptance
20 20-50 50-800 800-2000 2000	0.026 +6 dB/oct 0.16 -6 dB/oct 0.026	0.013 +6 dB/oct 0.08 -6 dB/oct 0.013
Overall	14.1 G_{rms}	10.0 G_{rms}

The acceleration spectral density level may be reduced for components weighing more than 22.7-kg (50 lb) according to:

	Weight in kg	Weight in lb	
dB reduction	= 10 log(W/22.7)	10 log(W/50)	
$ASD_{(50\text{-}800\ Hz)}$	= 0.16•(22.7/W)	0.16•(50/W)	for protoflight
$ASD_{(50\text{-}800\ Hz)}$	= 0.08•(22.7/W)	0.08•(50/W)	for acceptance

Where W = component weight.

The slopes shall be maintained at + and - 6dB/oct for components weighing up to 59-kg (130-lb). Above that weight, the slopes shall be adjusted to maintain an ASD level of 0.01 g^2/Hz at 20 and 2000 Hz.

For components weighing over 182-kg (400-lb), the test specification will be maintained at the level for 182-kg (400 pounds).

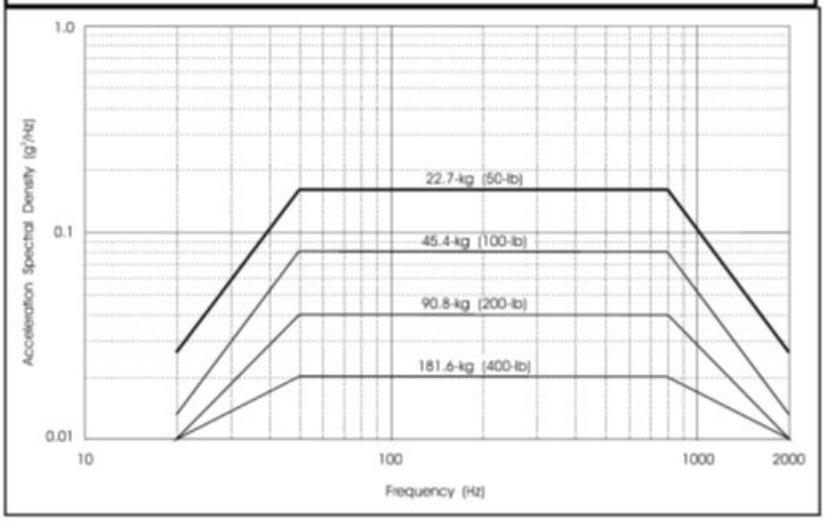

Figure 7. Random Vibration Environment

The random environment will be updated by NASA once more information is available at the mission level. Note for lightweight XRIS, the highest design loads may be from this random vibration environment.

XRIS-431: The contractor **shall** provide random vibration analysis along with static loads analysis. Please see NASA-HDBK-7005 and NASA-STD-7001 for more information.

Rationale: *Common best practices and multiple subsections in GEVS section 2.4.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

6.5 Shock

XRIS-434: The instrument **shall** be designed to meet its performance requirements after being subjected to the shock environment shown in Table 4 and Figure 8, which is currently defined at the LV-spacecraft interface. The shock environment envelopes all the Observatory shock events including separation from the launch vehicle and other actuations and will not exceed those shown in Table 4 and Figure 8.

> ***Rationale:*** *Testing in-flight configuration ensures that hardware which is difficult to analyze (i.e. blankets, harnesses, mechanisms) will be adequately screened by environmental testing for design or workmanship flaws. The presence of the optical system in this testing enables verification that the performance stability of the as-built opto-mechanical configuration is compliant to requirements (e.g., wave-front error, alignment, etc.) before and after testing.*

Note: Figure 8 is the acceptance shock input at the LV-spacecraft interface, not at the instrument-spacecraft interface. Further reduction between that interface and the instrument interfaces is expected due to joint and distance reductions. Information on the number of spacecraft structural joints between the LV-spacecraft interface and the instrument-spacecraft interface will be provided to the instrument to aid in their shock susceptibility and attenuation assessment once the spacecraft vendor has been selected.

XRIS-437: A shock susceptibility and attenuation assessment (analysis) **shall** be performed for the instrument. If the flight shock environment as shown on a Shock Response Spectra (SRS) plot (Q=10) is enveloped by the curve shown in Figure 8, then the shock environment can be considered benign and there is low risk in deferring the shock test to the Observatory level.

> ***Rationale:*** *The decision to perform component shock testing is typically based on an assessment of the shock susceptibility of the component and the expected shock levels. If the component is not considered shock sensitive and if there is low potential for damage due to the shock environment, then the project may choose to defer shock testing to the payload level of assembly.*

XRIS-438: Analysis supporting this conclusion (i.e., deferral of shock testing) **shall** be provided to the NASA/GSFC COR for review and deferral approval.

Any component within the instrument considered susceptible to the shock environment should have shock testing performed at the component level (preferably on a qualification unit).

XRIS-440: Instrument self-induced shock testing **shall** be accomplished by two actuations at the instrument level for each self-induced shock source (in order to account for the scatter associated with the actuation of the device) for the first flight unit, and a single actuation on subsequent units.

Table 4. Candidate LV Shock Environment

Candidate LV Separation Shock Design Envelope (P95/50 MPE)	
Frequency [Hz]	**SRS [G-peak]**

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

100	70
500	431.9
1000	1554
1550	2000
10000	2000

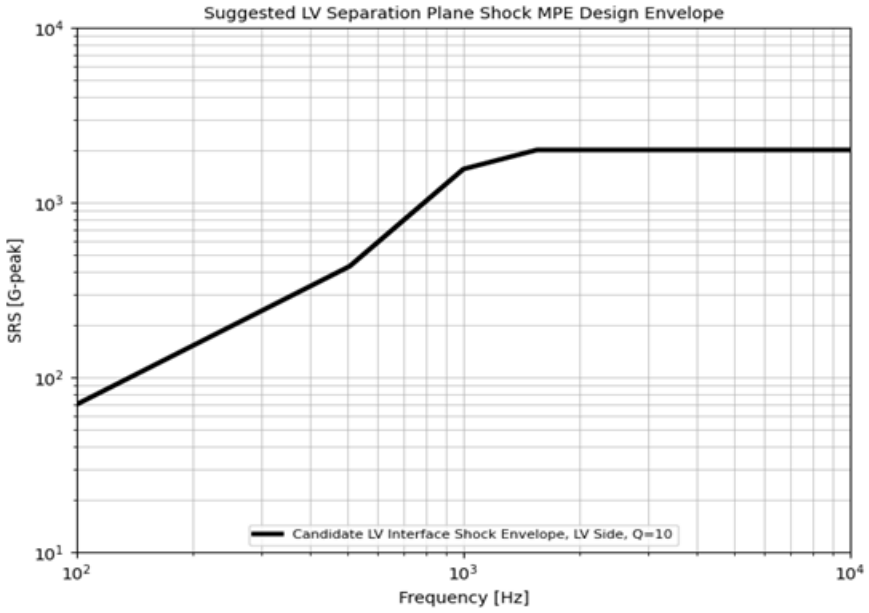

Figure 8. Candidate LV Shock Environment

6.6 Acoustics

XRIS-466: The instrument **shall** be designed to withstand, without any damage or degradation of performance, the protoflight acoustic noise environment equivalent to +3 dB over that shown in Table 5 and Figure 9.

Table 5. SOLAR Acoustic Design Envelope

Frequency [Hz]	SPL [dB]
25	120
31.5	125
40	126.5
50	127
63	127
80	128
100	130
125	131
160	131

200	131
250	130
315	128
400	126
500	124
630	122
800	119.5
1000	117.8
1250	116.4
1600	114.5
2000	112.5
2500	111
3150	109
4000	108
5000	107
6300	106
8000	104
10000	102
OASPL	**140**

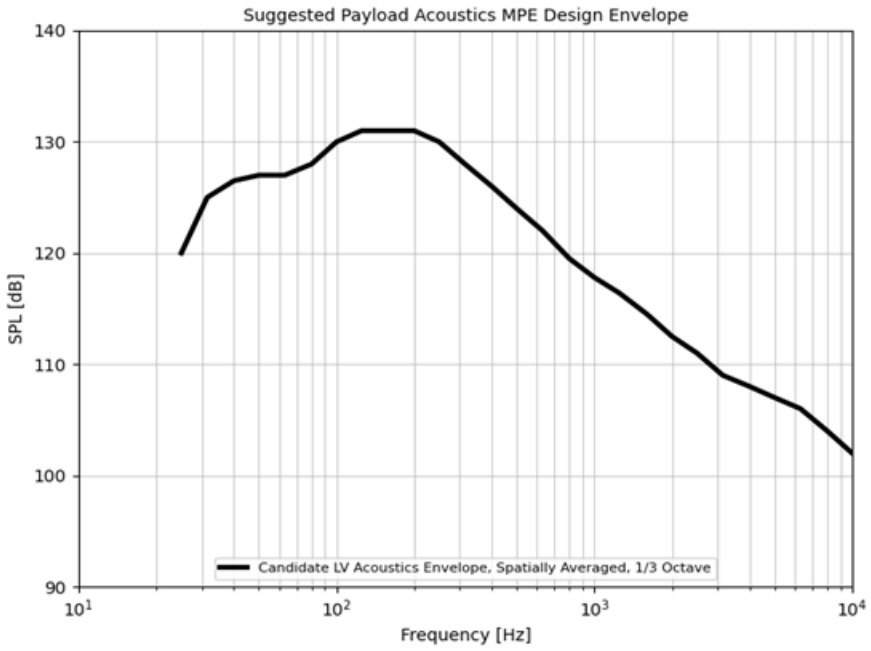

Figure 9. SOLAR Acoustic Design Envelope

XRIS-558: An acoustic test **shall** be performed unless an assessment of the unit indicates that it is not susceptible to the expected acoustic environment, or responses are enveloped by random

vibration testing, or that testing at higher levels of assembly provides sufficient exposure at an acceptable level of risk to the program.

This assessment and associated rationale will be provided to NASA/GSFC to support the decision to defer acoustic testing to the observatory level.

XRIS-560: Unit acoustic tests **shall** be to Protoflight levels (Acceptance + 3 dB) with a duration of 1 minute or equivalent Contractor Standard Practice.

Rationale: *Levels as specified in the SSRD, SIS and in this ERD.*

6.7 Transportation

XRIS-562: In addition to the launch loads shown above, the instrument **shall** also be designed to withstand the maximum transportation loads shown in Table 6 without damage or degradation of performance. Contractor Standard Practice may be used in lieu of these loads.

Table 6. Transportation Loads

Method	Fore(+)/Aft(-) (G)		Lateral (G)		Vertical (G)	
Air (C5)	+2.00	-1.00	-1.00	1.00	-3.60	1.60
Road	+1.00	-1.00	-0.75	0.75	-2.50	0.50
Notes: - Minus sign denotes downward for vertical load. Vertical loads include gravity (-1G) - All loads to be applied independently. For Road loads, in-plane loads to be combined with 1-G vertical loading						

6.7.1 Transportation Cleanliness

XRIS-587: Materials and enclosure used for transportation and storage **shall** not generate molecular or particle contaminants or degrade the surface cleanliness of the item or adjacent items to level below those required in Section 7.

6.8 Pressure

6.8.1 Operating Pressure Range

XRIS-590: The instrument **shall** be designed to be capable of aliveness testing at ambient pressure and at vacuum.

XRIS-591: The instrument **shall** meet all performance requirements while under vacuum (≤ 1 x 10^-5 Torr). The operating pressure range for the L1 Series Project Observatory will be 1.08 x 10^5 Pa (813 Torr) to 1.3 x 10^{-12} Pa (1 x 10^{-14} Torr).

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

6.8.2 Maximum Depressurization Rate

XRIS-593: The instrument **shall** be designed to meet all performance requirements after exposure to a maximum depressurization rate of -50 mbar/sec (-0.72 psi/sec) experienced during launch and ascent.

6.8.3 Launch Vehicle (LV) Environmental Control System (ECS) Impingement

XRIS-595: The instrument exterior surfaces **shall** not suffer damage or degradation when exposed to the LV ECS airflow velocity of 5 m/sec.

6.9 On-Orbit Dynamic Environment

XRIS-597: The instrument **shall** be designed to handle all permutations of the following linear and angular acceleration requirements simultaneously.

6.9.1 Dynamic Linear Acceleration

XRIS-599: The instrument **shall** survive on-orbit when subjected to the maximum linear acceleration due to nominal thruster firing of ±0.060 m/sec^2 in all axes.

6.9.2 Dynamic Angular Acceleration

XRIS-601: The instrument **shall** survive the maximum angular acceleration of ±2.5E-5 rad/sec^2 about all axes.

6.10 Thermal Requirements

XRIS-2668: The instrument **shall** be thermally independent from the spacecraft. Thermally independent units are conductively and radiatively decoupled from the spacecraft and reject their heat directly to space.

XRIS-2669: The instrument net heat transfer to/from the spacecraft **shall** be limited to ≤ 0.01 W/°C or as agreed in the ICD.

XRIS-605: The XRIS **shall** meet all of its performance requirements after exposure to air temperature between +5 and +30 degrees C and relative humidity between 30% and 70%.

> ***Rationale:*** *Common best practices and typical S/C I&T and LV supplied temperature controls.*

XRIS-607: The XRIS flight operation interface **shall** meet a temperature range of -10 to +40 degrees C.

XRIS-608: The XRIS survival interface **shall** meet a temperature range of -25 to +50 degrees C.

6.10.1 Flight Interface Design Temperature Limits

XRIS-610: When powered "OFF", the instrument **shall** be capable of surviving indefinitely when the spacecraft interface temperatures are within the survival limits without damage or permanent performance degradation.

If required, the spacecraft will provide survival heater power as described in section 4.2.5.

The spacecraft interface temperature ranges (Allowable Analytical or Flight Operation and Survival) and inclusion/exclusion of analytical uncertainty margin will be defined in the spacecraft to instrument ICD.

XRIS-613: Flight Operation, Acceptance and Qualification (Protoflight) temperature ranges **shall** be calculated as defined in Figure 10.

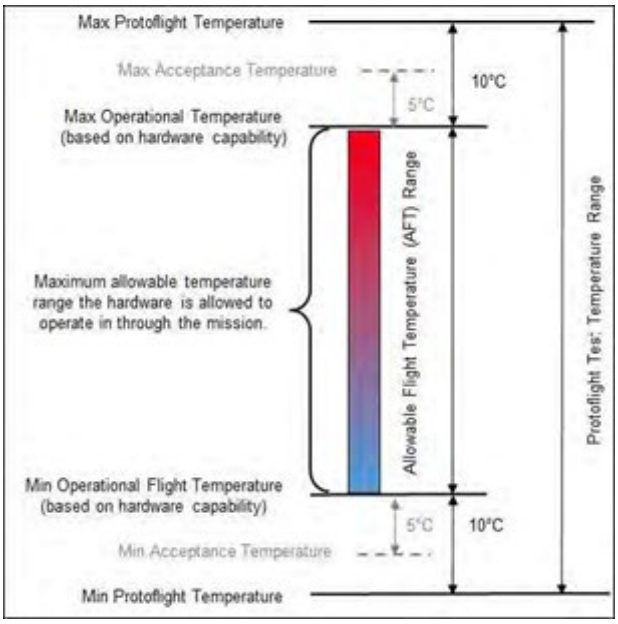

Figure 10. Allowable Analytical, Flight Operation, Flight Acceptance Test, and Qualification (ProtoFlight or Prototype) Test Temperature Ranges

XRIS-616: The instrument **shall** meet all performance requirements when operated within the Qualification limits.

Rationale: *Test as you fly.*

XRIS-617: The instrument **shall** remain within the Flight Operation temperature range when the spacecraft is in the nominal flight operation configuration.

XRIS-618: The instrument **shall** demonstrate turn on at the Minimum Survival and Maximum Protoflight temperature limits.

Rationale: *Test as you fly.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

XRIS-619: Thermal analysis **shall** be based on a Solar Flux range between 1312.1 W/m² and 1448.8 W/m².

XRIS-620: The instrument **shall** be designed to operate in and survive exposure to an orbital thermal environment consisting of solar radiation, given in Table 7.

Table 7. Orbit Thermal Parameters

SOLAR Solar Flux	L1 Solar Distance (AU)	Solar Flux (W/m^2)
Average	0.99	1380.5
Minimum	1.0067	1312.1
Maximum	0.9733	1448.8

XRIS-635: Instrument heaters **shall** be sized to have a duty cycle less than 70% in the worst-case cold conditions at the minimum supplied voltage or contractor standard practice.

6.11 Charged Particle Radiation Requirements

The SOLAR mission radiation environment will consist of galactic cosmic ray ions; protons and electrons trapped in the Van Allen belts; protons and heavier ions from solar events; and, solar wind plasma consisting of low energy electrons, protons and heavier ions as described in the SOLAR Radiation Hardness Assurance Requirements Document (RHARD) (SOLAR-SYS-REQ-0016). EEEE parts as well as some materials are subject to degradation or anomalous behavior in this environment due to multiple phenomena: Total Ionizing Dose (TID), Displacement Damage Dose (DDD), and both destructive and non-destructive single event effects (SEE).

6.11.1 Definitions

Displacement Damage Dose (DDD) - the mean energy deposited in a device region by radiation that goes into atomic displacements divided by the mass of the region. One common unit is MeV/g.

Electro-static Discharge (ESD) - sudden discharge of high intensity electric field that potentially degrades or damages insulating materials.

Enhanced Low Dose Rate Sensitivity (ELDRS) - used to refer to a part that shows enhanced radiation-induced damage at dose rates below 50 rad(Si)/s. The enhancement is the result of true dose rate effects.

Linear Energy Transfer (LET) - a measure of the ionizing energy deposited per unit length as an energetic particle travels through a material. The common LET unit is MeV·cm^2/mg of material.

Non-Ionizing Energy Loss (NIEL) - a measure of the energy loss per unit path length due to atomic displacements as a particle traverses a material. The common NIEL unit is MeV·cm^2/g of material.

Multiple Bit Upset (MBU) - an event induced by a single energetic particle such as a cosmic ray or proton that causes multiple upsets or transients during its path through a device or system in a single logical structure (ex., 2 bits affected in a single 16-bit word)

Non-recoverable SEE - single event effects without mitigation or protection schemes (generally applies to unit/box-level and above)

Recoverable SEE - non-destructive single event effects (generally applies to unit/box-level assessments and above)

Single Event Burnout (SEB) - An event in which a single energetic-particle strike through a high electric field induces a localized high-current state in the device, resulting in catastrophic device failure or in permanent degradation that is usually characterized by a significant increase in leakage current that exceeds the manufacturer's maximum specification.

Single Event Effect (SEE) - any measurable effect to a circuit due to a single particle strike, commonly an ion or a neutron-induced secondary ion. This includes, but is not limited to, single event upsets (SEUs), single event transients (SETs), single hard errors (SHEs), single event latchups (SELs), single event functional interrupts (SEFIs), single event burnouts (SEBs), single event gate ruptures (SEGRs), and single event dielectric ruptures (SEDRs).

Single-Event Functional Interrupt (SEFI) - a condition that causes loss of device functionality due to a change induced in a critical portion of a device, commonly a control structure, configuration file, or mode register. It generally requires a device reset or a reinitialization to resume normal device operations, but for many devices, a power cycle is necessary to initiate a full device reset to resume normal operations. A device undergoing a SEFI may simply be non-responsive or may have a sustained high-current state as it is no longer operating as designed.

Single-Event Gate Rupture (SEGR) - an event in which a single energetic-particle strike results in a breakdown and subsequent conducting path through the gate oxide of a MOSFET, MOS capacitor, or floating-gate memory. An SEGR is manifested by an increase in gate leakage current and can result in either the permanent degradation or the complete failure of the device.

Single Event Latchup (SEL) - a condition that may cause device failure due to a single event induced high current state associated with the turn-on of a real or parasitic thyristor that creates a short circuit between two power supply rails. A SEL may or may not cause permanent device damage but requires power cycling of the device to resume normal device operations. In addition, SEL that appear recoverable may suffer hidden degradation and must be evaluated for latent damage (device does not fail from the immediate single particle event, but reliability is degraded, and premature failure may occur). Guidance on radiation test and failure analysis of SEL-induced latent damage is defined in NASA Alert NA-GSFC-2005-05.

Single Event Transient (SET) – a temporary glitch or deviation from expected operation caused by one particle, with a subsequent return to normal operating behavior.

Single Event Upset (SEU) - a change of state induced by an energetic particle such as a cosmic ray or proton in a device, such as a bit flip in memory. These are "soft" errors in that a reset or rewriting of the device will usually return the device to normal behavior thereafter.

Single Hard Error (SHE) - a SEU that causes a permanent change to the operation of a device. An example is a stuck bit in a memory device.

Single-Bit Upset (SBU) and Multiple Bit Upset (MBU) – a distinction between events that upset a single circuit node (like a memory cell) and those that upset multiple nodes (or memory cells) at once.

Threshold LET (LETth) - the maximum LET at which no SEE is observed.

Total Ionizing Dose (TID) - the mean energy deposited by ionizing radiation in a device region divided by the mass of the region. This is often given in units of rad(Si), where 1 rad(Si) = 100 erg deposited per gram of silicon.

6.11.2 Radiation Hardness Assurance

XRIS-659: The instrument **shall** comply to the Radiation Hardness Assurance Requirement Document (SOLAR-SYS-REQ-0016).

XRIS-660: The instrument **shall** implement full triple modular redundancy on all Field Programable Gate Arrays (FPGAs) to mitigate the effects of non-destructive Single Event Upsets (SEUs).

XRIS-661: The instrument **shall** not be permanently damaged or placed into an unrecoverable state due to any single-event phenomena.

6.11.3 Charging Environment

XRIS-666: The instrument **shall** be designed to withstand the degradation of surface materials and associated surface charging as well as internal charging effects due to the radiation environment defined in this section for a mission with a transfer trajectory through the radiation belts and an orbit about the Sun-Earth first Lagrange point, L1.

Charging guidelines are described in NASA-HDBK-4002B.

This section presents the “worst case” charged particle (electrons and protons) environments likely to be encountered by the L1 Series Observatory from launch through operation at L1. This section provides the environments only. These environments will be used to establish design and verification of space charging mitigation requirements for the subsystems, elements, and Observatory in the various requirements documents and control plans of the program.

The two main categories of space charging environments: “surface charging” and “bulk charging” (also known as “internal” or “deep dielectric” charging).

The term “integral electron fluence” represents the total number of electrons impinging on a flat plate per unit area that have energy greater than or equal to the energy value shown. The integral fluence density can be used to conservatively model the number of absorbed electrons per unit area during the radiation belt transfer trajectory.

A solar photon radiation model is also included in section 6.11.3.3 to predict "photo emission" charging effects applied simultaneously with the particle flux environments for surface charging.

6.11.3.1 Charging Environments During Magnetic Substorm

6.11.3.1.1 Radiation Belt Geomagnetic Sub-Storm Surface Charging

The parameters for the geomagnetic sub-storm surface charging models are given in Table 8 in terms of electron density, electron temperature, ion density and ion (proton) temperature.

Table 8. Geo-magnetic Sub-storm Region Surface Charging Environment Model Parameters

Input Parameter	Particle Type	
	Electrons	Protons
N1 (density)	0.2 #/cm^3	1.6 #/cm^3
T1 (temperature)	0.4 keV	0.3 keV
N2 (density)	2.3 #/cm^3	1.3 #/cm^3
T2 (temperature)	24.8 keV	28.2 keV

ISO 19923 SCATHA Mullen1 Round Robin Worst Case

The values in Table 8 are input parameters to the "Maxwell-Boltzmann" isotropic speed distribution function below:

$$F(v) = \left(\frac{m}{2\pi k}\right)^{3/2}\left[\left(\frac{N}{T^{3/2}}\right)\exp\left(-\frac{mv^2}{2kT}\right)\right]$$

k is Boltzmann's constant

m is the mass of the particle

v is the speed of the particle

N is the particle density

T is the mean temperature of the particles

6.11.3.1.2 Radiation Belt Geomagnetic Sub-Storm Bulk Charging (Energetic Particle) Environment

The energetic particle environments have been derived for use in evaluating the effects of burying charge within dielectrics. The model, shown in Figure 11 and Table 9, is given in terms the integral fluence density.

Table 9. Geomagnetic Sub-storm Bulk Charging (Energetic Electron) Environment

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

Energy, MeV	Integral Electron Fluence, #/cm^2
0.04	3.94E+12
0.07	1.67E+12
0.1	8.99E+11
0.25	3.75E+11
0.5	1.34E+11
0.75	6.18E+10
1	3.09E+10
1.5	1.07E+10
2	3.44E+09
2.5	1.21E+09
3	5.46E+08
3.5	2.79E+08
4	1.50E+08
4.5	8.23E+07
5	4.72E+07
5.5	2.98E+07
6	2.05E+07
6.5	1.48E+07
7	1.10E+07
8.5	4.27E+06
10	0.00E+00

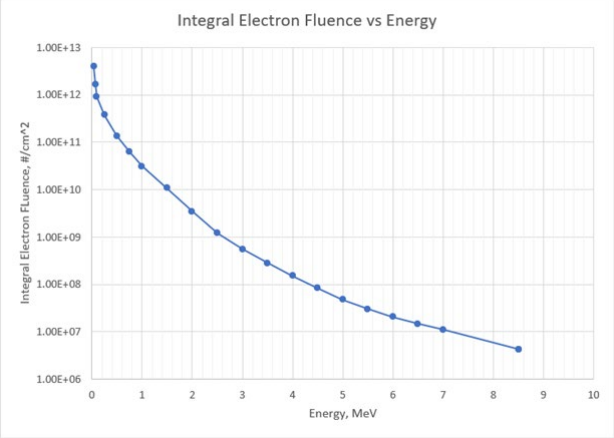

Figure 11. Integral Electron Fluence vs Energy

6.11.3.2 L1 Surface Charging Environment

The surface charging environment parameters at L1 are specified in Table 10.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

The environments are applied one-at-a-time during analysis of the effects. The environments in the table for each particle are applied one-at-a-time. The “low” and “high” cases are applied one at a time.

Directions (X, Y, and Z) used in the table are defined in Table 10 and Figure 12 The values in Table 13 are input parameters to the “Maxwell-Boltzmann” speed distribution function below:

$$F(v) = \left(\frac{m}{2\pi k}\right)^{3/2}\left[\left(\frac{N}{T^{3/2}}\right)\exp\left(-\frac{mv^2}{2kT}\right)\right]$$

where:

k is Boltzmann’s constant

m is the mass of the particle

v is the speed of the particle

N is the particle density

T is the mean temperature of the particles

The bulk flow velocity parameter, Vx, indicates a value added to each particle’s individual velocity calculated from the F(v) distribution formula in the X direction as defined in Table 10.

The direction of each environment specified in this section is defined in Table 11 and Figure 12.

Table 10. Plasma Moments for L1 Solar Wind Environment Surface Charging

Level	Input Parameter	Particle Type	
		Electrons	Protons
Low	N (density)	1.0 #/cm³	1.0 #/cm³
	T (temperature)	12 eV	0.8 eV
	V_x (bulk flow velocity)	+300 km/sec	+300 km/sec
High	N (density)	100 #/cm³	100 #/cm³
	T (temperature)	12 eV	50 eV
	V_x (bulk flow velocity)	+900 km/sec	+900 km/sec

Table 11. Directionality for L1 Surface Charging Analysis

L1 Environment	Particle Type	
	Electron	Protons
Solar Wind	isotropic	Beam in +Vx direction

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

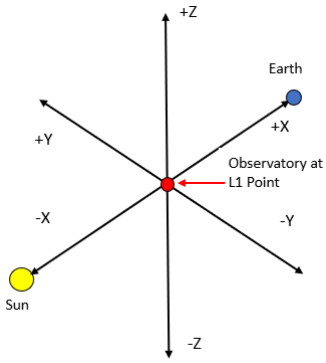

Figure 12. Direction Axes Used in L1 Plasma Flux Environments

6.11.3.3 Solar Photon Environment

The Solar Photon environment is described in Table 12. The table is used for calculating the photo emission charging effects on the surfaces of the Observatory. The solar photon environment is applied simultaneous with each particle charging environment.

Lyman-alpha (Ly-α) intensity is listed first in the Table 12 since this single emission line frequently represents >90% of the UV photon flux at wavelengths shorter than 150 nm with the full spectra following the Lyman-alpha values.

Table 12. Solar Photon Flux Density

Wavelength (nm)	Photon Flux Density (photons/cm^2-sec)		Wavelength (nm)	Photon Flux Density (photons/cm^2-sec)
Ly-α (121.567)	7.12E+11		250.5	7.54E+12
0.5	0.00E+00		251.5	5.79E+12
1.5	1.49E+08		252.5	5.25E+12
2.5	5.98E+08		253.5	6.54E+12
3.5	3.49E+07		254.5	7.59E+12
4.5	3.38E+08		255.5	1.01E+13
5.5	3.40E+08		256.5	1.31E+13
6.5	3.76E+08		257.5	1.62E+13
7.5	3.99E+08		258.5	1.62E+13
8.5	2.59E+08		259.5	1.35E+13
9.5	5.39E+08		260.5	1.11E+13
10.5	4.25E+08		261.5	1.18E+13
11.5	7.98E+08		262.5	1.36E+13
12.5	7.15E+08		263.5	2.19E+13
13.5	4.65E+08		264.5	3.31E+13
14.5	1.42E+09		265.5	3.36E+13
15.5	1.97E+08		266.5	3.33E+13
16.5	1.29E+08		267.5	3.36E+13

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

17.5	3.05E+09		268.5	3.28E+13
18.5	4.74E+09		269.5	3.22E+13
19.5	3.52E+09		270.5	3.62E+13
20.5	1.82E+09		271.5	3.05E+13
21.5	2.32E+09		272.5	2.69E+13
22.5	1.73E+09		273.5	2.69E+13
23.5	6.09E+08		274.5	1.82E+13
24.5	7.48E+08		275.5	2.41E+13
25.5	2.14E+09		276.5	3.36E+13
26.5	3.72E+08		277.5	3.25E+13
27.5	3.03E+09		278.5	2.22E+13
28.5	3.35E+09		279.5	1.20E+13
29.5	5.40E+09		280.5	1.33E+13
30.5	3.19E+09		281.5	2.93E+13
31.5	2.44E+09		282.5	4.16E+13
32.5	2.80E+07		283.5	4.45E+13
33.5	5.21E+09		284.5	3.35E+13
34.5	2.10E+09		285.5	2.33E+13
35.5	1.16E+09		286.5	4.63E+13
36.5	3.77E+09		287.5	4.84E+13
37.5	6.51E+08		288.5	4.65E+13
38.5	6.51E+08		289.5	6.85E+13
39.5	2.24E+07		290.5	8.74E+13
40.5	2.57E+08		291.5	8.47E+13
41.5	5.48E+08		292.5	7.64E+13
42.5	6.09E+08		293.5	7.87E+13
43.5	6.51E+08		294.5	7.50E+13
44.5	5.48E+08		295.5	8.18E+13
45.5	1.88E+07		296.5	7.56E+13
46.5	5.27E+08		297.5	7.55E+13
47.5	5.78E+07		298.5	6.95E+13
48.5	1.71E+08		299.5	7.26E+13
49.5	6.79E+08		300.5	6.32E+13
50.5	1.03E+09		301.5	6.87E+13
51.5	1.98E+08		302.5	7.40E+13
52.5	7.59E+08		303.5	9.43E+13
53.5	2.91E+08		304.5	9.18E+13
54.5	4.31E+07		305.5	9.09E+13
55.5	5.38E+08		306.5	8.52E+13
56.5	5.27E+08		307.5	9.47E+13
57.5	1.65E+08		308.5	9.44E+13
58.5	2.87E+08		309.5	7.69E+13
59.5	2.38E+09		310.5	9.67E+13

60.5	1.27E+09		311.5	1.14E+14
61.5	9.44E+07		312.5	1.03E+14
62.5	2.14E+09		313.5	1.10E+14
63.5	4.39E+08		314.5	1.04E+14
64.5	5.82E+08		315.5	1.00E+14
65.5	3.07E+08		316.5	1.00E+14
66.5	9.89E+07		317.5	1.23E+14
67.5	2.18E+08		318.5	1.06E+14
68.5	4.12E+08		319.5	1.14E+14
69.5	9.35E+07		320.5	1.29E+14
70.5	3.01E+08		321.5	1.13E+14
71.5	4.14E+08		322.5	1.11E+14
72.5	4.44E+07		323.5	1.07E+14
73.5	5.14E+07		324.5	1.24E+14
74.5	5.54E+07		325.5	1.43E+14
75.5	2.49E+08		326.5	1.60E+14
76.5	4.68E+08		327.5	1.56E+14
77.5	4.85E+08		328.5	1.51E+14
78.5	9.17E+08		329.5	1.75E+14
79.5	5.49E+08		330.5	1.68E+14
80.5	2.21E+08		331.5	1.60E+14
81.5	3.08E+08		332.5	1.59E+14
82.5	3.90E+08		333.5	1.54E+14
83.5	1.07E+09		334.5	1.61E+14
84.5	5.88E+08		335.5	1.58E+14
85.5	8.53E+08		336.5	1.36E+14
86.5	1.10E+09		337.5	1.42E+14
87.5	1.41E+09		338.5	1.56E+14
88.5	1.85E+09		339.5	1.63E+14
89.5	2.36E+09		340.5	1.72E+14
90.5	3.52E+09		341.5	1.58E+14
91.5	3.44E+09		342.5	1.70E+14
92.5	1.90E+09		343.5	1.63E+14
93.5	1.54E+09		344.5	1.37E+14
94.5	1.17E+09		345.5	1.62E+14
95.5	9.61E+07		346.5	1.61E+14
96.5	1.08E+08		347.5	1.57E+14
97.5	1.09E+10		348.5	1.57E+14
98.5	4.15E+08		349.5	1.56E+14
99.5	7.20E+08		350.5	1.84E+14
100.5	2.20E+08		351.5	1.72E+14
101.5	6.37E+08		352.5	1.60E+14
102.5	6.20E+09		353.5	1.83E+14

103.5	1.24E+10		354.5	1.97E+14
104.5	1.56E+10		355.5	1.86E+14
105.5	1.67E+11		356.5	1.63E+14
106.5	2.32E+11		357.5	1.42E+14
107.5	3.02E+11		358.5	1.24E+14
108.5	5.82E+11		359.5	1.78E+14
109.5	3.42E+11		360.5	1.75E+14
110.5	2.23E+09		361.5	1.59E+14
111.5	2.36E+09		362.5	1.81E+14
112.5	3.69E+11		363.5	1.81E+14
113.5	2.36E+09		364.5	1.85E+14
114.5	2.58E+09		365.5	2.11E+14
115.5	2.59E+09		366.5	2.26E+14
116.5	2.59E+09		367.5	2.17E+14
117.5	1.47E+12		368.5	2.03E+14
118.5	2.62E+09		369.5	2.27E+14
119.5	2.74E+09		370.5	2.11E+14
120.5	2.25E+12		371.5	2.18E+14
121.5	7.12E+11		372.5	1.97E+14
122.5	7.26E+10		373.5	1.72E+14
123.5	2.95E+09		374.5	1.69E+14
124.5	2.14E+09		375.5	2.00E+14
125.5	1.81E+09		376.5	2.04E+14
126.5	2.23E+09		377.5	2.42E+14
127.5	1.36E+09		378.5	2.51E+14
128.5	1.11E+09		379.5	2.02E+14
129.5	2.59E+09		380.5	2.29E+14
130.5	7.88E+09		381.5	2.07E+14
131.5	2.62E+09		382.5	1.51E+14
132.5	2.74E+09		383.5	1.31E+14
133.5	1.12E+10		384.5	1.85E+14
134.5	3.08E+09		385.5	1.94E+14
135.5	2.58E+09		386.5	1.95E+14
136.5	2.11E+09		387.5	1.95E+14
137.5	2.01E+09		388.5	1.91E+14
138.5	2.75E+09		389.5	2.29E+14
139.5	5.28E+09		390.5	2.44E+14
140.5	4.27E+09		391.5	2.63E+14
141.5	2.98E+09		392.5	2.00E+14
142.5	3.34E+09		393.5	1.15E+14
143.5	3.67E+09		394.5	2.03E+14
144.5	3.68E+09		395.5	2.60E+14
145.5	4.03E+09		396.5	1.70E+14

146.5	5.20E+09		397.5	1.85E+14
147.5	6.26E+09		398.5	3.04E+14
148.5	6.09E+09		399.5	3.33E+14
149.5	5.95E+09		400.5	3.37E+14
150.5	6.55E+09		401.5	3.51E+14
151.5	7.03E+09		402.5	3.54E+14
152.5	8.88E+09		403.5	3.40E+14
153.5	9.98E+09		404.5	3.39E+14
154.5	1.59E+10		405.5	3.38E+14
155.5	1.67E+10		406.5	3.23E+14
156.5	1.45E+10		407.5	3.32E+14
157.5	1.35E+10		408.5	3.55E+14
158.5	1.33E+10		409.5	3.52E+14
159.5	1.40E+10		410.5	3.15E+14
160.5	1.55E+10		411.5	3.74E+14
161.5	1.80E+10		412.5	3.69E+14
162.5	2.05E+10		413.5	3.63E+14
163.5	2.32E+10		414.5	3.60E+14
164.5	2.45E+10		415.5	3.61E+14
165.5	3.56E+10		416.5	3.84E+14
166.5	3.32E+10		417.5	3.48E+14
167.5	3.25E+10		418.5	3.53E+14
168.5	3.84E+10		419.5	3.57E+14
169.5	4.99E+10		420.5	3.70E+14
170.5	5.65E+10		421.5	3.79E+14
171.5	5.91E+10		422.5	3.35E+14
172.5	6.26E+10		423.5	3.63E+14
173.5	6.64E+10		424.5	3.75E+14
174.5	7.92E+10		425.5	3.61E+14
175.5	9.48E+10		426.5	3.62E+14
176.5	1.08E+11		427.5	3.36E+14
177.5	1.25E+11		428.5	3.40E+14
178.5	1.37E+11		429.5	3.17E+14
179.5	1.42E+11		430.5	2.45E+14
180.5	1.65E+11		431.5	3.64E+14
181.5	2.03E+11		432.5	3.56E+14
182.5	2.05E+11		433.5	3.76E+14
183.5	2.06E+11		434.5	3.63E+14
184.5	1.91E+11		435.5	3.76E+14
185.5	2.15E+11		436.5	4.21E+14
186.5	2.52E+11		437.5	3.95E+14
187.5	2.82E+11		438.5	3.44E+14
188.5	3.10E+11		439.5	4.01E+14

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

189.5	3.39E+11		440.5	3.78E+14
190.5	3.59E+11		441.5	4.27E+14
191.5	3.99E+11		442.5	4.39E+14
192.5	3.96E+11		443.5	4.24E+14
193.5	3.69E+11		444.5	4.39E+14
194.5	5.07E+11		445.5	4.06E+14
195.5	5.31E+11		446.5	4.23E+14
196.5	5.91E+11		447.5	4.65E+14
197.5	6.12E+11		448.5	4.43E+14
198.5	6.15E+11		449.5	4.56E+14
199.5	6.65E+11		450.5	4.83E+14
200.5	7.35E+11		451.5	4.77E+14
201.5	8.09E+11		452.5	4.40E+14
202.5	8.37E+11		453.5	4.47E+14
203.5	9.39E+11		454.5	4.50E+14
204.5	1.05E+12		455.5	4.64E+14
205.5	1.08E+12		456.5	4.75E+14
206.5	1.15E+12		457.5	4.81E+14
207.5	1.31E+12		458.5	4.52E+14
208.5	1.57E+12		459.5	4.62E+14
209.5	2.24E+12		460.5	4.70E+14
210.5	2.94E+12		461.5	4.75E+14
211.5	3.55E+12		462.5	4.87E+14
212.5	3.33E+12		463.5	4.73E+14
213.5	3.54E+12		464.5	4.59E+14
214.5	4.30E+12		465.5	4.76E+14
215.5	3.90E+12		466.5	4.49E+14
216.5	3.50E+12		467.5	4.71E+14
217.5	3.84E+12		468.5	4.68E+14
218.5	4.85E+12		469.5	4.68E+14
219.5	5.15E+12		470.5	4.42E+14
220.5	5.22E+12		471.5	4.76E+14
221.5	4.39E+12		472.5	4.83E+14
222.5	5.53E+12		473.5	4.72E+14
223.5	7.05E+12		474.5	4.87E+14
224.5	6.63E+12		475.5	4.80E+14
225.5	5.90E+12		476.5	4.66E+14
226.5	4.47E+12		477.5	4.96E+14
227.5	4.55E+12		478.5	4.81E+14
228.5	5.94E+12		479.5	4.98E+14
229.5	5.48E+12		480.5	4.89E+14
230.5	6.01E+12		481.5	5.04E+14
231.5	5.91E+12		482.5	4.89E+14

232.5	6.20E+12		483.5	4.89E+14
233.5	5.23E+12		484.5	4.77E+14
234.5	4.57E+12		485.5	4.45E+14
235.5	6.06E+12		486.5	3.96E+14
236.5	5.75E+12		487.5	4.47E+14
237.5	5.76E+12		488.5	4.68E+14
238.5	5.01E+12		489.5	4.80E+14
239.5	5.29E+12		490.5	4.93E+14
240.5	4.76E+12		491.5	4.66E+14
241.5	6.22E+12		492.5	4.67E+14
242.5	8.57E+12		493.5	4.66E+14
243.5	8.11E+12		494.5	5.09E+14
244.5	7.46E+12		495.5	4.78E+14
245.5	6.07E+12		496.5	5.01E+14
246.5	6.22E+12		497.5	5.02E+14
247.5	6.88E+12		498.5	4.66E+14
248.5	5.68E+12		499.5	4.93E+14
249.5	7.04E+12		500.5	4.65E+14

6.11.4 Total Ionizing Dose

XRIS-668: The instrument external surfaces, if required, **shall** have special coatings for electrostatic discharge (ESD) suppression in the plasma environment (except where such coatings degrade performance).

XRIS-670: The instrument **shall** meet all performance requirements after exposure to the bounding case of 95th percentile mission total ionizing dose as specified in Table 13 and Figure 13.

> ***Rationale:*** *To ensure survival of the radiation environment over during the life of the mission.*

Table 13. Total Ionizing Dose vs. Shielding Thickness

Al absorber thickness			**5.5 years, 2033**
[mm]	**[mils]**	**[g/cm^2]**	**[krad-(Si)]**
0.05	1.968	0.014	6.92E+02
0.1	3.937	0.027	4.14E+02
0.2	7.874	0.054	2.45E+02
0.3	11.811	0.081	1.79E+02
0.4	15.748	0.108	1.40E+02
0.5	19.685	0.135	1.15E+02
0.6	23.622	0.162	9.92E+01
0.8	31.496	0.216	7.63E+01
1	39.37	0.27	6.17E+01

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

1.5	59.055	0.405	4.21E+01
2	78.74	0.54	3.13E+01
2.5	98.425	0.675	2.49E+01
3	118.11	0.81	2.04E+01
4	157.48	1.08	1.47E+01
5	196.85	1.35	1.12E+01
6	236.22	1.62	9.13E+00
7	275.59	1.89	7.54E+00
8	314.96	2.16	6.36E+00
9	354.33	2.43	5.50E+00
10	393.7	2.7	4.75E+00
12	472.44	3.24	3.75E+00
14	551.18	3.78	3.01E+00
16	629.92	4.32	2.50E+00
18	708.66	4.86	2.12E+00
20	787.4	5.4	1.80E+00

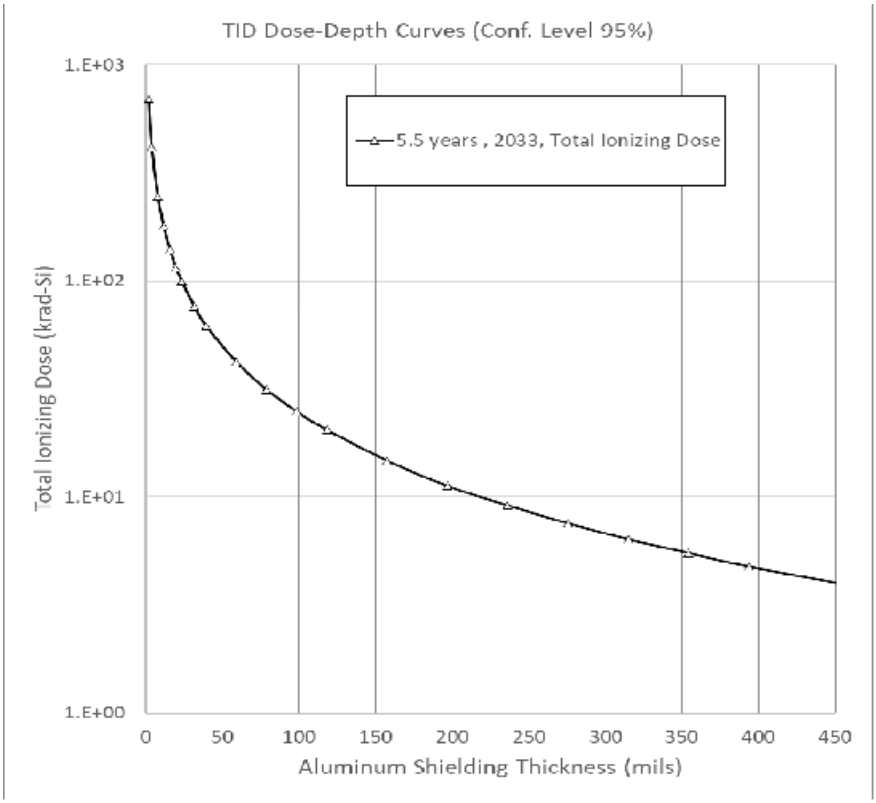

Figure 13. Total Ionizing Dose-Depth Curve

6.11.5 Displacement Damage Dose Levels

XRIS-809: The instrument **shall** meet all performance requirements after exposure to the 95th percentile mission displacement damage dose as specified in the bounding case Figure 14 and

Table 14 (Silicon materials), Figure 15 and Table 15 (GaAs materials), or equivalent for other materials.

Rationale: *To ensure survival of the radiation environment over during the life of the mission.*

Table 14. Displacement Damage Dose in Silicon vs. Shielding Thickness

Al absorber thickness			5.5 years, 2033
[mm]	[mils]	[g/cm^2]	[MeV/g-(Si)]
0.05	1.968	0.014	1.44E+10
0.1	3.937	0.027	8.46E+09
0.2	7.874	0.054	4.77E+09
0.3	11.811	0.081	3.39E+09
0.4	15.748	0.108	2.71E+09
0.5	19.685	0.135	2.22E+09
0.6	23.622	0.162	1.92E+09
0.8	31.496	0.216	1.49E+09
1	39.37	0.27	1.21E+09
1.5	59.055	0.405	8.38E+08
2	78.74	0.54	6.46E+08
2.5	98.425	0.675	5.14E+08
3	118.11	0.81	4.29E+08
4	157.48	1.08	3.12E+08
5	196.85	1.35	2.43E+08
6	236.22	1.62	2.01E+08
7	275.59	1.89	1.68E+08
8	314.96	2.16	1.41E+08
9	354.33	2.43	1.24E+08
10	393.7	2.7	1.09E+08
12	472.44	3.24	8.58E+07
14	551.18	3.78	7.07E+07
16	629.92	4.32	5.87E+07
18	708.66	4.86	5.03E+07
20	787.4	5.4	4.30E+07

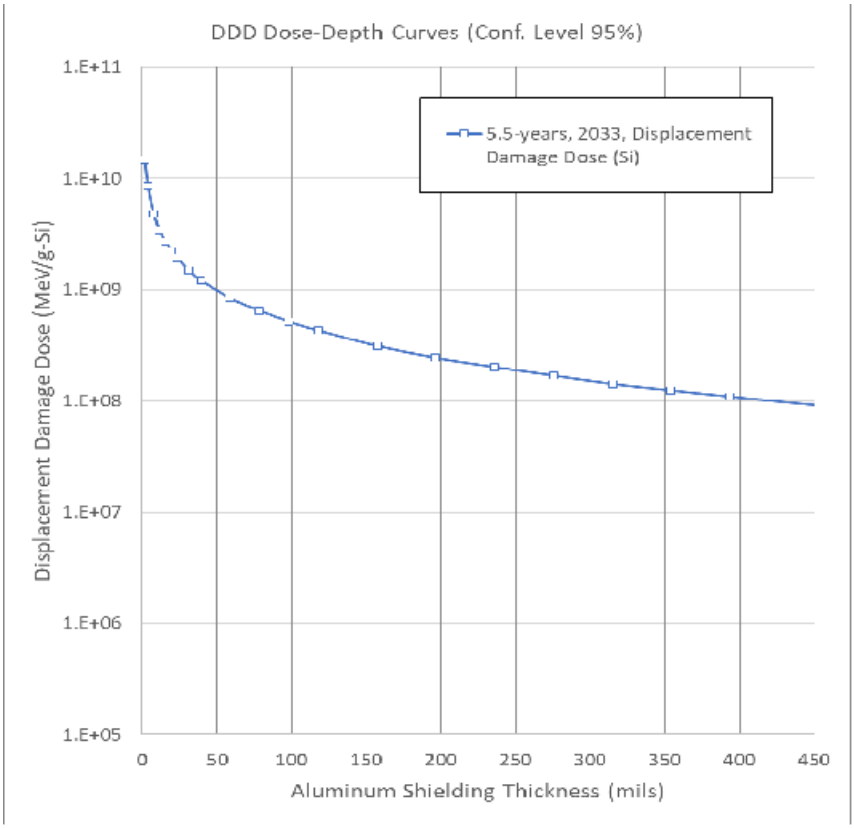

Figure 14. Displacement Damage Dose in Silicon vs. Shielding Thickness Dose-Depth Curve

Table 15. Displacement Damage Dose in Gallium Arsenide vs. Shielding Thickness

Al absorber thickness			5.5 years, 2033
[mm]	[mils]	[g/cm^2]	[MeV/g-(GaAs)]
0.05	1.968	0.014	1.19E+10
0.1	3.937	0.027	7.03E+09
0.2	7.874	0.054	3.97E+09
0.3	11.811	0.081	2.83E+09
0.4	15.748	0.108	2.26E+09
0.5	19.685	0.135	1.85E+09
0.6	23.622	0.162	1.60E+09
0.8	31.496	0.216	1.24E+09
1	39.37	0.27	1.01E+09
1.5	59.055	0.405	7.03E+08
2	78.74	0.54	5.45E+08
2.5	98.425	0.675	4.37E+08
3	118.11	0.81	3.67E+08
4	157.48	1.08	2.70E+08
5	196.85	1.35	2.13E+08
6	236.22	1.62	1.78E+08

7	275.59	1.89	1.50E+08
8	314.96	2.16	1.27E+08
9	354.33	2.43	1.13E+08
10	393.7	2.7	9.99E+07
12	472.44	3.24	7.99E+07
14	551.18	3.78	6.67E+07
16	629.92	4.32	5.62E+07
18	708.66	4.86	4.87E+07
20	787.4	5.4	4.22E+07

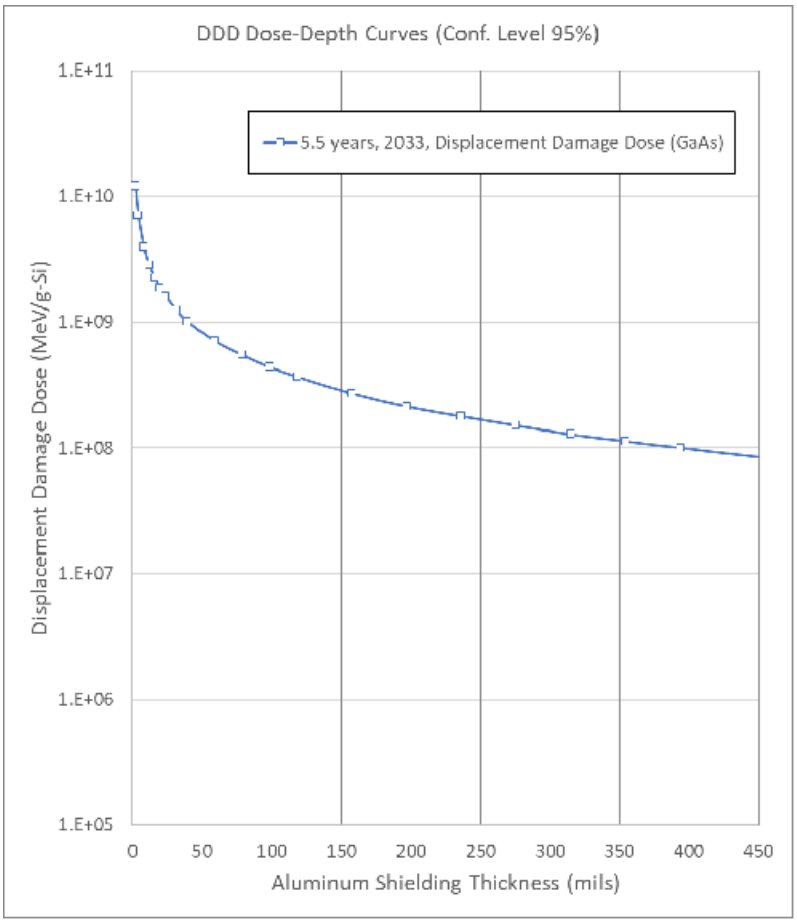

Figure 15. Displacement Damage Dose in Gallium Arsenide Dose-Depth Curve

6.11.6 Single Event Effects

SEE Environment calculations are based on a combination of the IRENE AP9/AE9 (trapped particles), ESP-PSYCHIC (solar particles), CREME96 (peak October 1989 solar particle event), and ISO-15390 (galactic cosmic rays).

6.11.7 Solar Cycle Performance

XRIS-1087: The instrument **shall** meet all performance requirements when operating in the bounding case assuming 100 mil Aluminum shielding of single-event effects background environment defined by Figure 16 and Table 16 (galactic cosmic rays), Figure 17 and Table 17 (solar particles), and Figure 18 and Table 18 (solar protons) or shielded flux values emergent from a shielding ray trace analysis or equivalent.

Table 16. LET Spectrum of GCR Background

GCR	
LET	**5.5 years, 2033 start**
[MeV-cm^2/mg]	**Flux > LET [#/cm^{-2}/day]**
0.001609	3.6E+05
0.0022224	1.0E+05
0.0030698	6.3E+04
0.0042877	4.8E+04
0.0059225	4.1E+04
0.0081806	1.9E+04
0.011426	1.0E+04
0.015783	6.7E+03
0.022045	4.8E+03
0.03045	3.8E+03
0.058747	2.9E+03
0.081146	2.2E+03
0.11208	1.5E+03
0.15655	1.0E+03
0.21624	7.7E+02
0.30204	5.3E+02
0.4172	3.5E+02
0.57627	2.5E+02
0.8049	1.9E+02
1.1118	1.2E+02
1.5529	4.6E+01
2.145	2.3E+01
2.9628	1.1E+01
4.1383	5.4E+00
5.7161	2.6E+00
7.8956	1.3E+00
11.028	5.7E-01
15.233	2.4E-01
21.277	9.1E-02
29.389	3.8E-03
40.594	4.5E-05

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

56.7	1.5E-05
78.318	3.5E-06
109.39	1.2E-09

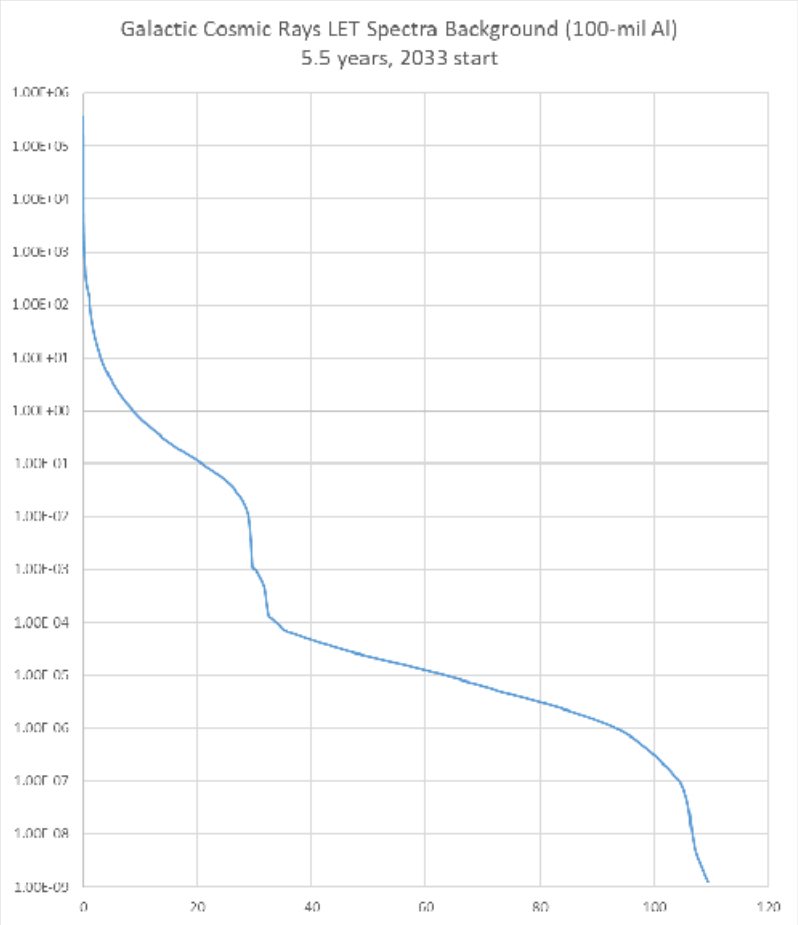

Figure 16. LET Spectrum of GCR Background

Table 17. Solar Background LET Spectrum

Solar	
LET	**5.5 years, 2033 start**
[MeV-cm^2/mg]	**Flux > LET [cm^{-2}/day]**
0.001609	3.4E+07
0.0022224	3.4E+07
0.0030698	3.4E+07
0.0042877	3.3E+07
0.0059225	3.2E+07
0.0081806	2.9E+07
0.011426	2.4E+07
0.015783	1.8E+07
0.022045	1.2E+07

0.03045	7.2E+06
0.04206	3.9E+06
0.058747	2.0E+06
0.081146	1.0E+06
0.11208	5.0E+05
0.15655	2.3E+05
0.21624	1.0E+05
0.30204	4.3E+04
0.4172	1.7E+04
0.57627	3.5E+03
0.8049	1.9E+03
1.1118	9.8E+02
1.5529	3.5E+02
2.145	2.1E+02
2.9628	1.2E+02
4.1383	6.0E+01
5.7161	2.7E+01
7.8956	9.7E+00
11.028	3.6E+00
15.233	1.1E+00
21.277	4.4E-01
29.389	2.3E-02
40.594	5.2E-05
56.7	1.7E-05
78.318	4.5E-06
109.39	1.0E-09

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

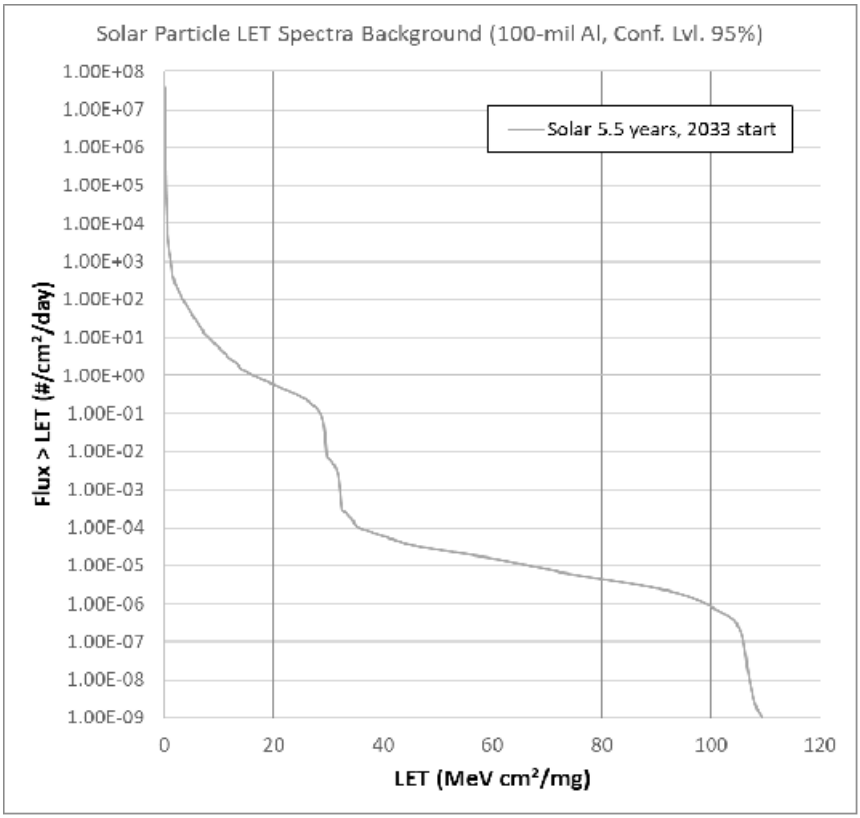

Figure 17. Solar LET Spectrum Background

Table 18. Solar Proton Background Flux

Solar	
Energy [MeV]	**5.5 years, 2033 start**
	Flux > E [cm^{-2}/day]
0.10	3.71E+07
0.30	3.70E+07
0.50	3.70E+07
0.70	3.70E+07
1.10	3.69E+07
1.40	3.68E+07
2.80	3.63E+07
3.50	3.59E+07
5.00	3.50E+07
6.30	3.41E+07
9.00	3.19E+07
12.00	2.93E+07
16.00	2.58E+07
22.00	2.10E+07
32.00	1.48E+07
40.00	1.13E+07

55.00	7.13E+06
71.00	4.59E+06
100.00	2.31E+06
140.00	1.08E+06
180.00	5.70E+05
250.00	2.24E+05
350.00	6.22E+04
450.00	8.06E+03

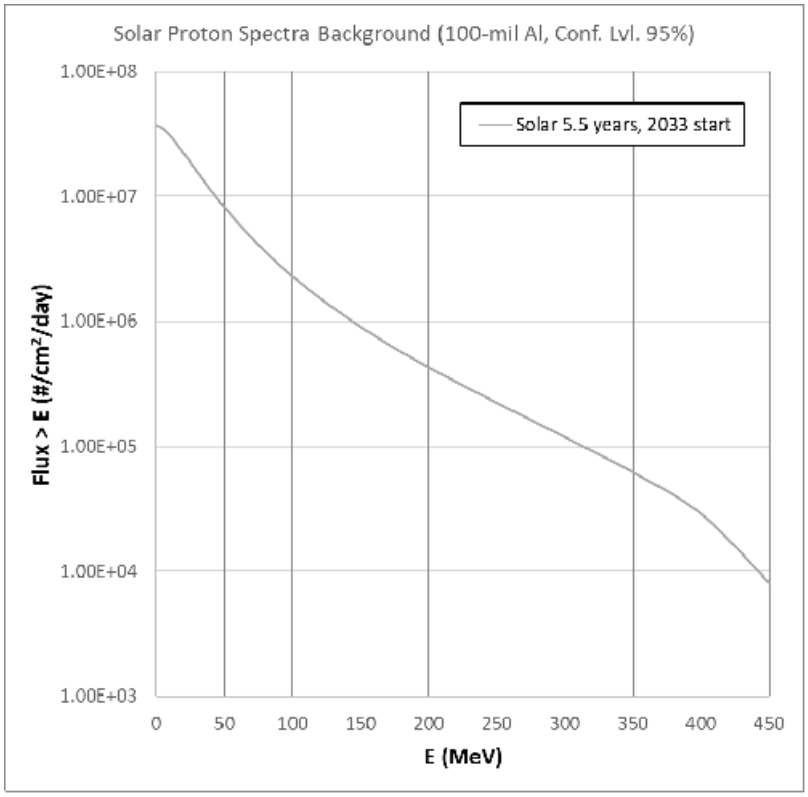

Figure 18. Solar Proton Background Spectra

6.11.8 Solar Particle Event Operations

XRIS-1392: The instrument **shall** meet data gap and availability performance requirements XRIS-198 and XRIS-199 during exposure to the worst week peak fluxes listed in Figure 19 and Table 19 (solar particles) and Figure 20 and Table 20 (solar protons).

XRIS-1393: The instrument **shall** function after exposure to the worst 5-Min peak fluxes listed in Figure 19 and Table 19 (solar particles) and Figure 20 and Table 20 (solar protons).

Table 19. Abbreviate Peak Solar Particle Event Flux

LET [MeV-cm²/mg]	Worst Week Flux > LET [cm⁻²/s]	Worst Day Flux > LET	Worst 5-Min Flux > LET [cm⁻²/s]

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

		[cm^{-2}/s]	
0.001609	1.05E+04	4.69E+04	1.74E+05
0.0022224	1.05E+04	4.69E+04	1.74E+05
0.0030698	1.05E+04	4.67E+04	1.73E+05
0.0042877	1.03E+04	4.63E+04	1.72E+05
0.0059225	9.90E+03	4.51E+04	1.67E+05
0.0081806	9.09E+03	4.23E+04	1.57E+05
0.011426	7.71E+03	3.68E+04	1.37E+05
0.015783	5.93E+03	2.89E+04	1.08E+05
0.022045	3.98E+03	1.98E+04	7.47E+04
0.03045	2.36E+03	1.19E+04	4.50E+04
0.04206	1.29E+03	6.61E+03	2.50E+04
0.058747	6.49E+02	3.34E+03	1.26E+04
0.081146	3.28E+02	1.70E+03	6.44E+03
0.11208	1.57E+02	8.20E+02	3.10E+03
0.15655	7.22E+01	3.80E+02	1.44E+03
0.21624	3.14E+01	1.66E+02	6.29E+02
0.30204	1.29E+01	6.87E+01	2.60E+02
0.4172	4.69E+00	2.50E+01	9.46E+01
0.57627	7.03E-01	3.93E+00	1.47E+01
0.8049	3.55E-01	1.89E+00	7.06E+00
1.1118	1.74E-01	8.65E-01	3.22E+00
1.5529	5.30E-02	1.86E-01	6.76E-01
2.145	3.41E-02	1.17E-01	4.26E-01
2.9628	2.18E-02	7.36E-02	2.67E-01
4.1383	1.30E-02	4.34E-02	1.57E-01
5.7161	7.10E-03	2.37E-02	8.57E-02
7.8956	3.49E-03	1.16E-02	4.17E-02
11.028	1.76E-03	5.83E-03	2.10E-02
15.233	8.00E-04	2.62E-03	9.41E-03
21.277	3.26E-04	1.07E-03	3.85E-03
29.389	1.64E-05	5.36E-05	1.93E-04
40.594	4.44E-08	1.05E-07	3.78E-07
56.7	1.48E-08	3.35E-08	1.21E-07
78.318	3.11E-09	6.83E-09	2.47E-08
109.39	1.15E-12	2.32E-12	8.40E-12

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

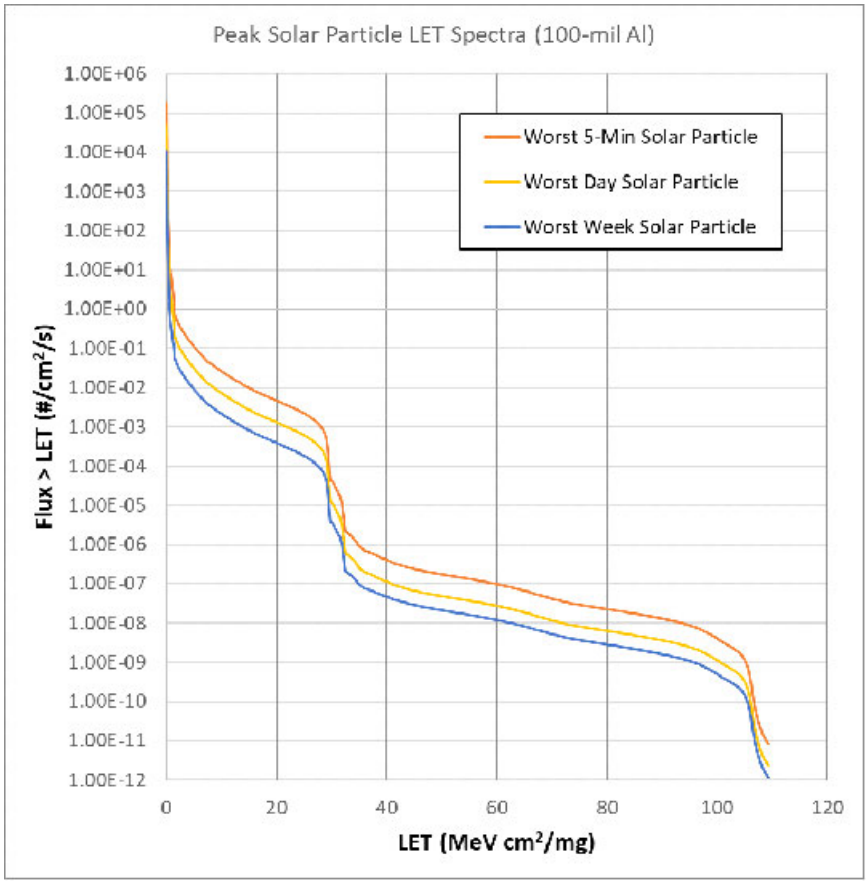

Figure 19. Peak Solar Particle Event Flux

Table 20. Peak Solar Proton Flux

Energy [MeV]	Worst Week Flux > E [cm^{-2}/s]	Worst Day Flux > E [cm^{-2}/s]	Worst 5-Min Flux > E [cm^{-2}/s]
0.1	1.05E+04	4.67E+04	1.73E+05
0.3	1.05E+04	4.66E+04	1.73E+05
0.5	1.04E+04	4.66E+04	1.73E+05
0.7	1.04E+04	4.65E+04	1.72E+05
1.1	1.04E+04	4.64E+04	1.72E+05
1.4	1.04E+04	4.62E+04	1.71E+05
2.8	1.02E+04	4.54E+04	1.68E+05
3.5	1.01E+04	4.49E+04	1.66E+05
5	9.84E+03	4.35E+04	1.61E+05
6.3	9.57E+03	4.21E+04	1.56E+05
9	8.91E+03	3.87E+04	1.43E+05
12	8.12E+03	3.48E+04	1.28E+05
16	7.07E+03	2.97E+04	1.09E+05
22	5.68E+03	2.30E+04	8.42E+04
32	3.91E+03	1.49E+04	5.43E+04

40	2.96E+03	1.08E+04	3.92E+04
55	1.83E+03	6.18E+03	2.24E+04
71	1.17E+03	3.69E+03	1.34E+04
100	5.87E+02	1.67E+03	6.09E+03
140	2.76E+02	7.07E+02	2.62E+03
180	1.49E+02	3.55E+02	1.34E+03
250	5.91E+01	1.30E+02	5.07E+02
350	1.70E+01	3.51E+01	1.43E+02
450	2.22E+00	4.43E+00	1.86E+01

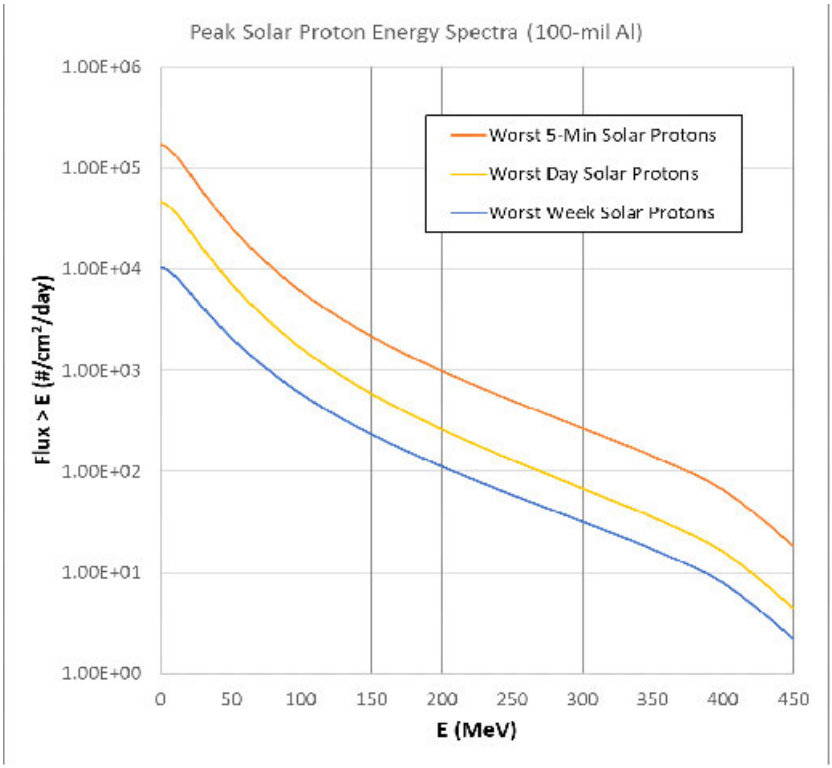

Figure 20. Peak Solar Proton Flux

6.11.9 Solar Flare Survival

XRIS-1693: The instrument **shall** remain functional after exposure to the worst day peak proton flux from 3 S4 or S5 class solar radiation storms of 4 days (i.e., 12 solar storm days) during the orbit lifetime and meet SC safe state requirements.

> ***Rationale:*** *S4 or S5 class solar radiation storms occur at a rate of 3 per solar cycle per NOAA. A reasonable outside estimate for the duration of these conditions is 4 days, hence a total duration of 12 days for this requirement.*

7 CONTAMINATION CONTROL

The information in this section addresses the cleanliness requirements for the instrument so that instrument performance will still be in compliance with requirements at the end of the designed mission life, and the instrument will not be a contamination threat to other instruments and the spacecraft. For this mission, the most contamination sensitive instruments will generally dictate the contamination requirements for other instruments and the spacecraft.

7.1 Surface Cleanliness Requirements

The following table presents instrument surface cleanliness requirements (per IEST-STD-CC1246) at the various mission phases. Please note that whereas the instrument may not require stringent contamination control, it is necessary to be compliant with the imposed contamination levels so that companion payloads and spacecraft subsystems will not be detrimentally impacted.

XRIS-1698: The instrument **shall** meet the contamination requirement at delivery to observatory in Table 21. Values shown in Table 21 for "post delivery" are for information only and are not verified requirements.

Table 21. Lifetime Contamination Requirements

INSTRUMENT SURFACE CLEANLINESS REQUIREMENTS (per IEST-STD-CC1246E)					
	Delivery to Observatory	Encapsulation	**Launch**	**BOL**	**EOL**
Particulate	VC-0.5-1000 UV (365)1	VC-0.5-1000 UV (365)1	0.02 PAC2	0.0805 PAC	0.0805 PAC
Molecular	R5E-1	R5E-1	R5E-1	R1	R1

1. Approximately equal Level 300 or PAC of 0.02 per IEST-STD-CC1246E
2. PAC: Percent Area Coverage

7.2 Contamination Generation

7.2.1 Particulate Generation

XRIS-1732: The instrument should not contain any of the following particle generating materials or construction unless specific steps are taken to minimize or eliminate the possibility of particulate generation:

Rationale: *Flowdown from IEST-STD-CC1246*

- Paints prone to shedding due to large paint pigment molecules, overspray, poor adhesion, etc.
- Dry lubricants (e.g. molybdenum disulfide).
- Surfaces prone to corrosion or oxides because of a lack of corrosion protection or dissimilar metals in close contact.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

- Fabrics with brittle constituents (e.g., composites, graphite or glass).
- Perforated materials when material is highly susceptible to tear propagation (e.g., multi-layer insulation (MLI)).
- Perforations must be properly created in materials like Kapton which tears easily. Perforated Kapton used as an inner layer sandwiched between outer layers with proper edge venting is acceptable.
- Perforated ITO coated Aluminized Silver Teflon which is sometimes required for charging reasons as an outer layer must have the material inspected for tears and cleaned to remove any chads, before assembly into the blanket.
- Metal oxides (bare [untreated] aluminum and magnesium, iron, non-corrosion resistant steel, etc.).
- Braided metallic or synthetic wires, ropes, slings, etc. unless measures have been taken to contain any broken filaments or fibers (sheathing, sealing with polymers, covering, etc.).
- Woven materials (metal braid, EMI shielding, lacing cord, thread, expando sleeving) should be minimized; when unavoidable they require special mitigations, such as:
- Special cleaning (ultrasonic, vacuuming, solvent, etc.) to remove molecular, particle or other contaminants from manufacture especially non-vacuum stable lubricants like silicone oils used to facilitate passage through machines;
- Special treatment of unfinished ends to prevent unraveling or generating fibers (cut with hot knife, seal with polymer, fuse with heat);
- Bagging/sleeving to prevent excessive contamination that becomes entrained in strands resisting removal
- Materials with thin films known to erode or crack or flake when subjected to normal handling (e.g., indium tin oxide [ITO] or other rigid or brittle semiconductor flexible substrates, Teflon, MLI, etc.).
- This pertains to ‘coated’ thin films – Erosion, cracking and flaking of coatings which are typically applied to MLI and other thin films. A test must be performed to determine how well the coatings are adhering to the item being coated.
- Foams, highly textured materials.
- Trapped debris in holes.

7.2.2 Molecular Contamination Generation

XRIS-1745: All materials used in the instrument **shall** have less than 1% TML and less than 0.1% CVCM.

Rationale: *The TML and VCM requirements are defined per NASA specification JSC SP-R-0022A.*

7.2.2.1 Molecular Material Restrictions

XRIS-1747: The instrument **shall** not contain Silicones.

- Waivers may be submitted to use silicones. The waiver will need to demonstrate that the quantity is minimal and that the silicone with the lowest TML and CVCM outgassing properties was selected. All waivers that are submitted to the instrument Material Review Board will describe how the top-level molecular surface cleanliness requirements can still be satisfied with the proposed material use.

XRIS-1749: Instrument designs **shall** limit the potential of outgassing contaminants from reaching sensitive optic surfaces, such as the Coronagraph. The use of organics in the XRIS interior should be limited to those applications in which an in-organic alternative is not possible or practical. Diffusion paths into the optical cavity should be limited whenever possible (e.g., designing adhesive joints such that diffusion paths are exterior to the optics cavity or the available surface area for outgassing is minimized). Organics used in optical cavities should have the lowest outgassing rate possible for the application.

Rationale: *To ensure mission success by limiting degrading of sensitive surfaces.*

7.3 Vacuum Bakeouts

XRIS-1753: Vacuum bakeouts will be performed on hardware at the box, unit or subsystem level in order to help deplete the contamination sources available for outgassing during the mission. All instrument hardware **shall** receive a bakeout to verify the outgassing rate (OGR) is met prior to the completion of instrument integration activities.

Rationale: *Standard best contamination control practice.*

7.4 Venting Requirements

XRIS-1763: The instrument **shall** vent away from observatory contamination sensitive surfaces to be defined in the Spacecraft ICD.

Rationale: *Inadvertent venting provides unintended long-term outgassing paths that can increase molecular mass deposition on surfaces that will change their optical or thermal characteristics.*

XRIS-1764: The instrument vents **shall** have filters or other means of preventing the egress of particles larger than 35 microns.

Rationale: *Inadvertent venting provides unintended long-term outgassing paths that can increase molecular mass deposition on surfaces that will change their optical or thermal characteristics.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

XRIS-1765: Dedicated molecular vents **shall** be designed to vent to a cryogenic plate or similar molecular trap during thermal vacuum testing, if the effluent from the vent is large enough to pose a contamination risk to the instrument during thermal vacuum.

Rationale: *To ensure that inadvertent molecular contamination does not occur.*

XRIS-1766: Instrument vents **shall** be sufficient to prevent a buildup of pressure during ascent that may result in mechanical damage.

Rationale: *To ensure the instrument survives launch and ascent.*

XRIS-1767: Instrument vents **shall** be sufficient to prevent a buildup of pressure during high vacuum due to outgassing that could result in discharges in higher voltage circuits.

Rationale: *To ensure the instrument survives launch and ascent.*

XRIS-1768: Instrument vents sizing **shall** be analyzed to determine the time at which it is safe to operate at high voltages after passing through the Corona region after launch.

Rationale: *To ensure the instrument survives launch and ascent.*

7.5 Reserved

7.6 Cleanability and Protection

XRIS-1777: All instrument external surfaces **shall** be capable of being cleaned with low particulate, low NVR, polyester or similar texture wipes, soft bristle brushes, and isopropyl alcohol.

Rationale: *To ensure that the instrument can be cleaned should molecular or particulate contamination occur during integration and test, etc.*

XRIS-1779: All external surfaces should be capable of being vacuumed.

Rationale: *To ensure that the instrument can be cleaned should particulate contamination occur during integration and test, etc.*

8 DESIGN AND CONSTRUCTION REQUIREMENTS

8.1 Electrical

8.1.1 Connector Specifications

XRIS-1829: Selected connector types **shall** meet the Connector and Contact Requirements defined in Section C2 of EEE-INST-002 or equivalent Contractor Standard Practice.

> ***Rationale:*** *Best EEE practices for flight hardware. EEE-INST-002 provides a mechanism to assure that appropriate parts are used in the fabrication of space hardware that will meet mission reliability objectives*

8.1.2 Test Interfaces

Instrument test signals that require access during observatory-level testing should be protected or isolated from facility-induced noise, ESD GE malfunction, or other GSE malfunction.

8.1.2.1 Spacecraft Charging Limits

The following section provides requirements needed in order to minimize the charging of the surfaces and volumes of the instrument when exposed to the space plasma. Additional guidance for how the following requirements can be met can be found in "SOLAR Project Spacecraft Charging Guidelines (SOLAR-DESC-0003)."

XRIS-2646: All instrument surface potentials **shall** be less than 400 volts with respect to adjacent surfaces, spacecraft frame potential, or grounded substrates in the worst-case charging environments specified in Sections 6.11.3 and at the worst-case operating temperature in sun or shadow.

XRIS-2647: The internal electric field in all instrument materials **shall** be <1E7 V/m in the worst-case charging environments specified in Section 6.11.3 and at the worst-case operating temperature in sun or shadow.

Note: These charging requirements apply to all instrument materials, including, but not limited to blankets, coatings, PC boards and components, machined dielectrics, and cable insulation.

Context: verified by modeling / analysis. Enveloping environmental heritage (e.g. GEO) can be used in lieu of performing the analyses.

8.2 Safety

Applicable sections of NASA-STD-8719.24 ELV Safety Requirements are required in this section. The SOLAR Project will support determination of applicability for the requirements below. Sections of the NASA STD 8719.24 Annexes that are relevant to the hardware and GSE

being developed are included in Volume 3 and Volume 6. Volume 3 contains system/design requirements and Volume 6 contains operational requirements.

a) Mechanical Systems and GSE: Volume 3 Chapter 6 and Volume 6 Chapter 6

b) Hazardous Materials: Volume 3 Chapter 10 and Volume 6

c) Electronics (including Battery requirements): Volume 3 Chapter 14 and Volume 6 Chapter 14

d) Seismic Requirements – Volume 3 Chapter 17

XRIS-1860: All flight units **shall** meet the applicable sections of NASA-STD-8719.24 ELV Safety Requirements. See 8.2 for applicable sections.

Rationale: *Flow down of SSRD for LV Safety Requirements*

XRIS-1861: The instrument **shall** supply at least two inhibits for potential instrument catastrophic hazards when integrated to the observatory.

Rationale: *Want the instrument to provide 2 of 3 inhibits to prevent a catastrophic hazard. The spacecraft will provide the third inhibit.*

XRIS-1862: All Ground Support Equipment being used at the Launch site **shall** meet the applicable sections of NASA-STD-8719.24 ELV Safety Requirements. See 8.2 for applicable sections.

Rationale: *Flow down of SSRD for LV Safety Requirements*

8.3 Electromagnetic Compatibility

XRIS-1864: The instrument **shall** be electromagnetically compatible with itself and with all the external environments in which they are expected to operate.

Rationale: *Best practices for the control of electromagnetic interference (EMI) emission and susceptibility characteristics of electronic, electrical, and electromechanical equipment and subsystems.*

Table 22 summarizes the emission and susceptibility requirements.

Table 22. Emission and Susceptibility Requirements

Type	Test Description[1]	Component Applicability	Req Number	Comments
CE101	Conducted Emissions, Power Leads 30 Hz - 150 kHz	All	XRISSPEC-1935	
CE03	Conducted Emissions, Power Leads 150 kHz - 50 MHz	All	XRISSPEC-1937	

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

CMBCE	Common Mode Bulk Conducted Emissions, Power and Signal Lines 30 Hz - 200 MHz	All	XRISSPEC-1941 XRISSPEC-1942 XRISSPEC-1945	
CS101	Conducted Susceptibility, Power Leads, 30 Hz - 150 kHz	All	XRISSPEC-1951	
CS06	Conducted Susceptibility, Transients, Power Leads, 10 µsec transient	All	XRISSPEC-1970 XRISSPEC-1971 XRISSPEC-1974	
CS114DM	Conducted Susceptibility, Bulk Cable Injection, Differential Mode, Primary Power Leads, 150 kHz - 50 MHz	All	XRISSPEC-1957	
CS114CM	Conducted Susceptibility, Bulk Cable Injection, Common Mode, Interconnecting Power and Signal Cables, 10 kHz - 200 MHz	All	XRISSPEC-1965	
CS115	Conducted Susceptibility, Bulk Cable Injection, Impulse Excitation, Interconnecting Power and Signal Cables, 34 nsec @ 30 Hz	All	XRISSPEC-1976	
RE102	Radiated Emissions, Electric Field[2]	All	XRISSPEC-1982	
RS103	Radiated Susceptibility, Electric Field[3]	All	XRISSPEC-1988 XRISSPEC-2017	

Notes:
1. Test methods can be in accordance with the latest version of the Goddard Environmental Verification Specification (GEVS), MIL-STD-461G/462 or Contractor Standard Practice.
2. Default limit. The limit will be updated when all receivers, including instruments, if any, have been defined.
3. Default limit. The limit will be updated when all transmitters, including launch vehicle, have been defined.

8.3.1 Conducted Emissions

Conducted emission requirements are designed to prevent excessive noise from being induced on the spacecraft power bus by units that are connected to the bus.

8.3.1.1 Applicability of Conducted Emissions

The conducted emissions requirements are applicable for the primary power (+28V bus) and return leads that supply power to the unit. The requirements do not apply to secondary power wires or signal wires.

8.3.1.2 CE101 – Differential Mode Conducted Emissions Limits

XRIS-1935: The instrument **shall** not produce reflected ripple greater than the CE101 limits from 30 Hz to 150 kHz shown in Figure 21, when power leads are tested to a modified MIL-STD-461G CE101 (extended frequency range) test method as described in GSFC-STD-7000B, Section 2.5.2.1.1 or Contractor Standard Practice

8.3.1.3 CE03 – Differential Mode Conducted Emissions Limits

XRIS-1937: The instrument **shall** not produce reflected ripple greater than the limits from 150 kHz to 50 MHz shown in Figure 21, when power leads are tested to the MIL-STD-462 CE03 test method as described in GSFC-STD-7000B, Section 2.5.2.1.1 or Contractor Standard Practice.

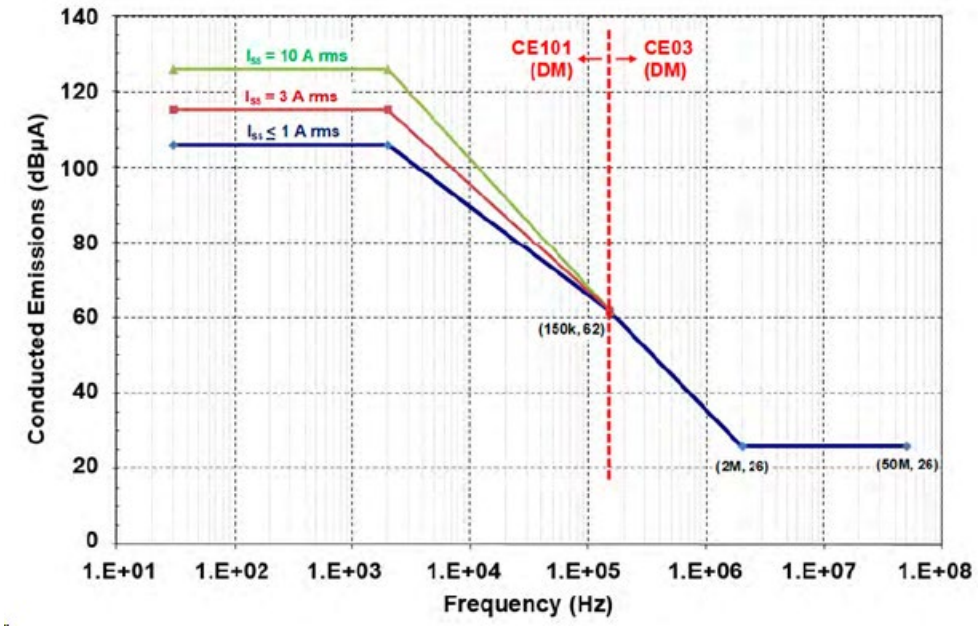

Figure 21. CE101/CE03 Differential Mode Conducted Emissions Limits

8.3.1.4 CMBCE – Common Mode Bulk Conducted Emissions Limits

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

XRIS-1941: A Common Mode Bulk Conducted Emissions (CMBCE) test **shall** be performed from 30 Hz to 10 kHz (as shown in Figure 22) on each component power and signal cables using the modified CE101 test method per GSFC-STD-7000B Section 2.5.2.1.2 or Contractor Standard Practice.

XRIS-1942: A Common Mode Bulk Conducted Emissions (CMBCE) test **shall** be performed from 10 kHz to 30 MHz (as shown in Figure 22) on each component power and signal cables using the modified CE03 test method per GSFC-STD-7000B Section 2.5.2.1.2 or Contractor Standard Practice.

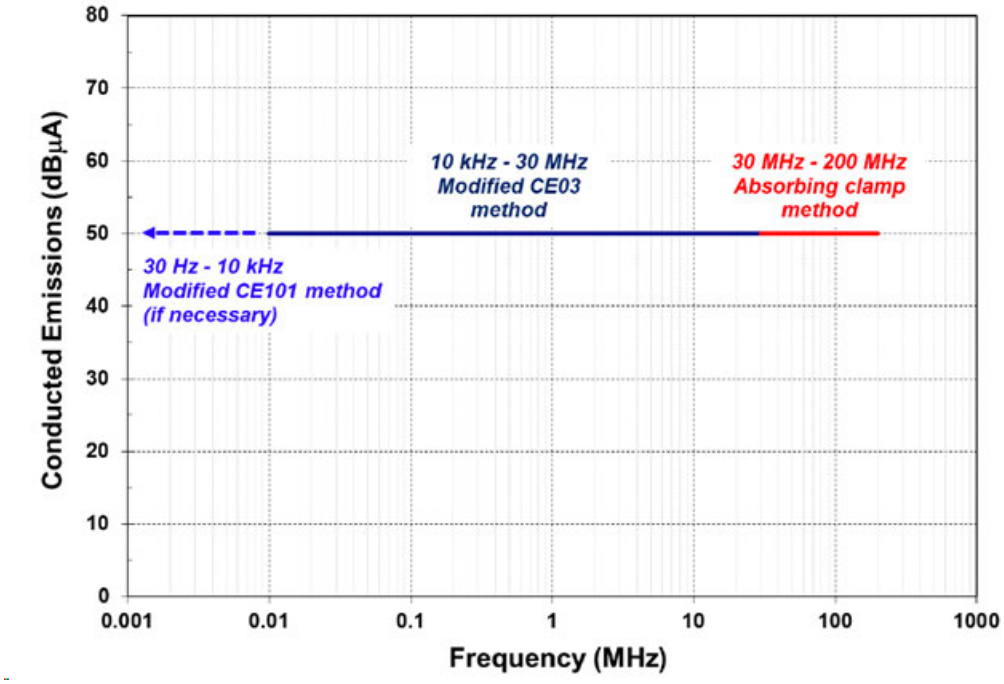

Figure 22. Common Mode Bulk Conducted Emissions Limit

XRIS-1945: A Common Mode Bulk Conducted Emissions (CMBCE) test **shall** be performed from 30 MHz to 200 MHz (as shown in Figure 22) on each component power and signal cables using the absorbing clamp test method per GSFC-STD-7000B Section 2.5.2.1.2 or Contractor Standard Practice.

8.3.2 Conducted Susceptibility

Conducted susceptibility requirements are designed to ensure that any unit connected to the spacecraft power bus is not unduly susceptible to noise expected to be present on the power bus.

8.3.2.1 Applicability of Conducted Susceptibility

The conducted susceptibility requirements are applicable for the primary power wires that supply power to a unit. Conducted susceptibility requirements do not apply to the power return wires, secondary power wires or signal wires.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

8.3.2.2 CS101 – Conducted Susceptibility Limit

XRIS-1951: The instrument **shall** meet performance requirements when power leads are subjected to the limits shown in Figures 23 and 24, from 30 Hz to 150 kHz per the MIL-STD-461G CS101 test method as described in GSFC-STD-7000B, Section 2.5.2.2.1 or Contractor Standard Practice.

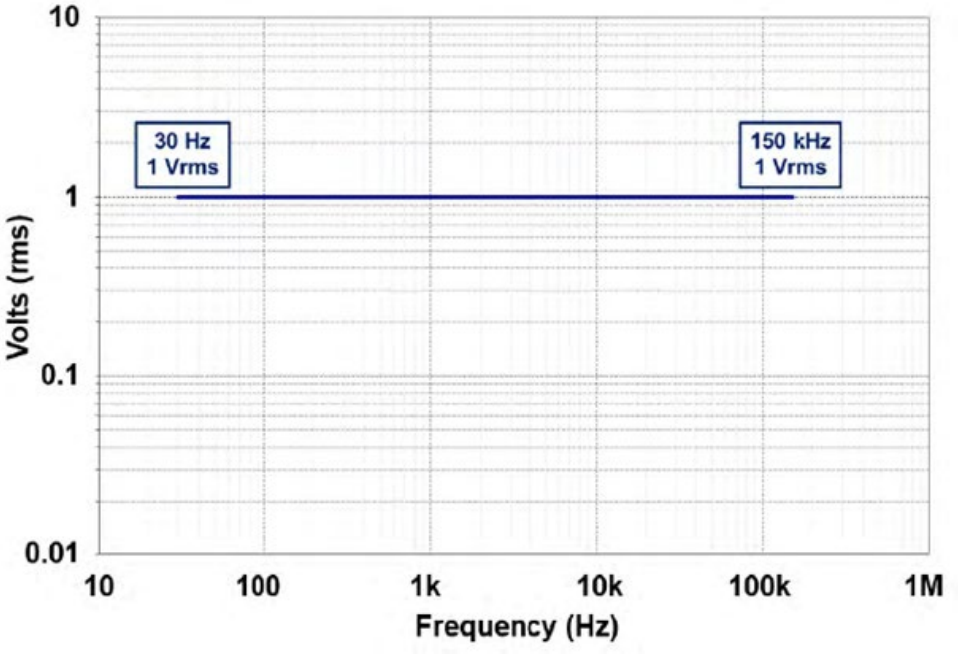

Figure 23. Power Lead Conducted Susceptibility (CS101) Voltage Limit

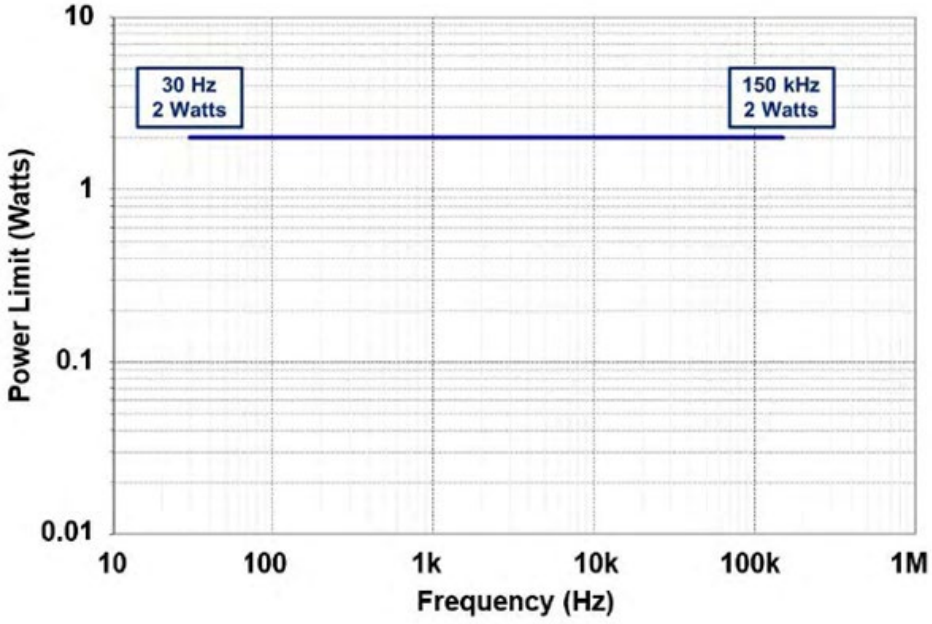

Figure 24. Conducted Susceptibility Power Limit (30 Hz to 150 kHz)

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

8.3.2.3 CS114 – Conducted Susceptibility Differential Mode Limit

XRIS-1957: The instrument **shall** not exhibit any malfunction, degradation of performance, or deviation from specified indications beyond the tolerances indicated in the individual equipment or subsystem specification, when its primary power lead is subjected to the injected potential limit shown in Figure 25 that is modulated as specified in MIL-STD-461G or Contractor Standard Practice.

Prior to injection onto the EUT, the injection signal must be pre-calibrated using the MIL-STD-461G CS114 test method to the current limit shown in Figure 26. This current limit is equivalent to an injected potential of 1 Vrms into the 100 ohms in the CS114 calibration fixture.

Requirements are also met if the source cannot establish the required level at the EUT and the EUT is not susceptible at the forward power levels determined during pre-calibration.

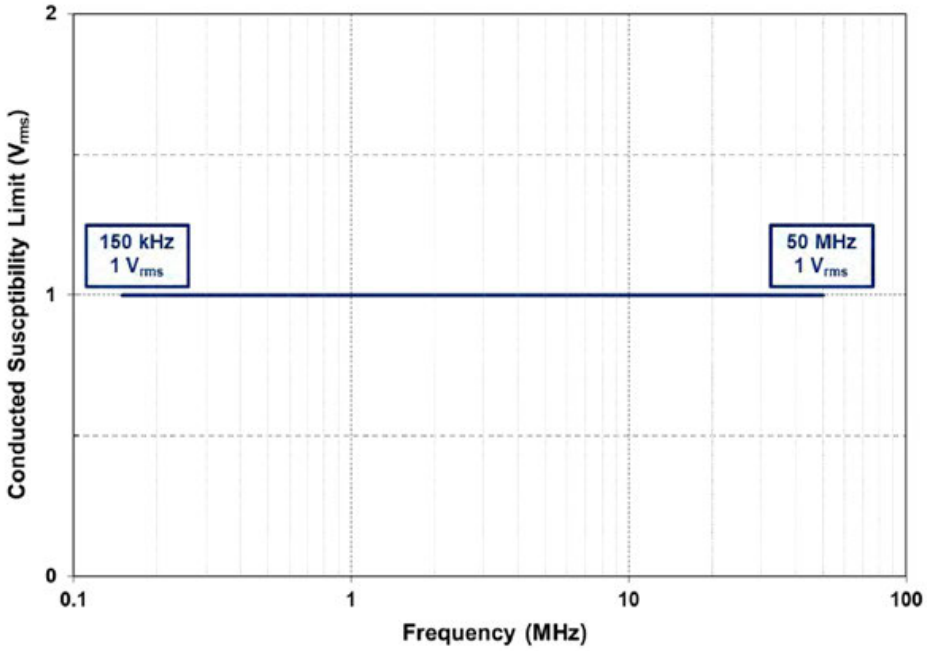

Figure 25. Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz Limit

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

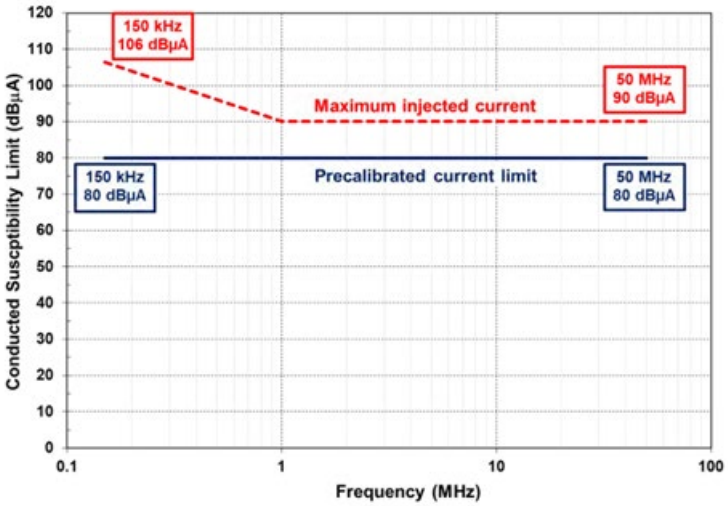

Figure 26. Conducted Susceptibility, Power Leads, 150 kHz to 50 MHz Current Limits

8.3.2.4 CS114 – Conducted Susceptibility, Common Mode, Interconnected Power and Signal Cables

XRIS-1965: The instrument **shall** not exhibit any malfunction, degradation of performance, or deviation from specified indications beyond the tolerances indicated in the instrument specification, when subjected to an injection probe drive level which has been pre-calibrated to the GEVS current limit shown in Figure 25 and is modulated as specified in MIL-STD-461G or Contractor Standard Practice.

Requirements are also met if the instrument is not susceptible at forward power levels sensed by the directional coupler that are below those determined during calibration provided that the actual current induced in the cable under test is 76 dBµA, i.e. 6 dB higher than the level in the plateau portion of the curve, over the entire frequency range, as shown by the red dashed curve in Figure 27. (This is the injection method specified by MIL-STD-461G, which is different from the method specified by MIL-STD-461F.)

Applicability: This requirement is applicable to all interconnecting power and signal cables.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

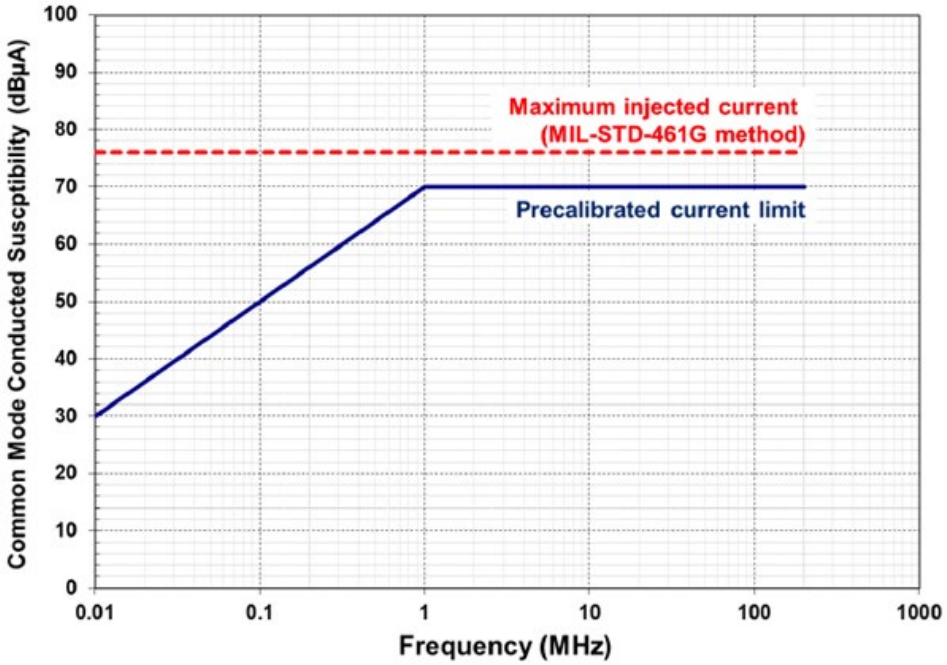

Figure 27. Common Mode Conducted Susceptibility Calibration Limit

8.3.2.5 CS06 – Conducted Susceptibility, Transients Limits

XRIS-1970: For all platforms, the negative transient **shall** be applied.

XRIS-1971: For platforms that include rapidly changing inductive loads that are expected to generate significant back EMF onto the bus, the positive transient **shall** be applied as well (Figure 28).

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

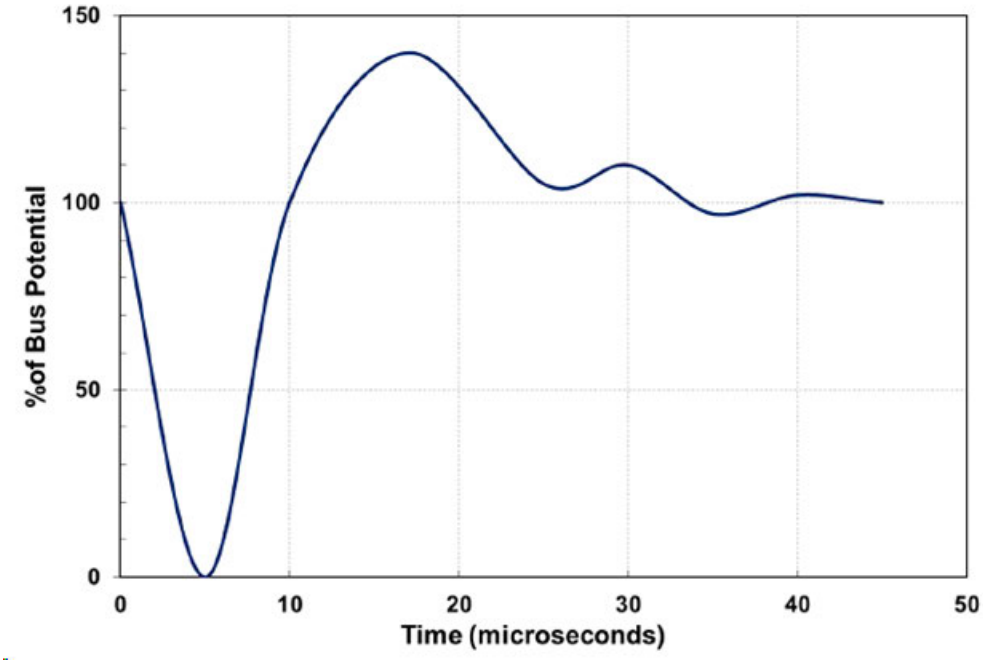

Figure 28. Conducted Susceptibility Transient Waveform

XRIS-1974: The pulse **shall** be applied per the MIL-STD-462 CS06 test method or Contractor Standard Practice.

8.3.2.6 CS115 - Conducted Susceptibility, Bulk Cable Injection, Impulse Excitation

XRIS-1976: The CS115 impulse defined in MIL-STD-461G **shall** be applied or Contractor Standard Practice. This waveform is reproduced in Figure 29. The amplitude, rise time, fall time, duration, and repetition rate of the applied impulse should be tailored according to the expected impulse amplitudes on the platform.

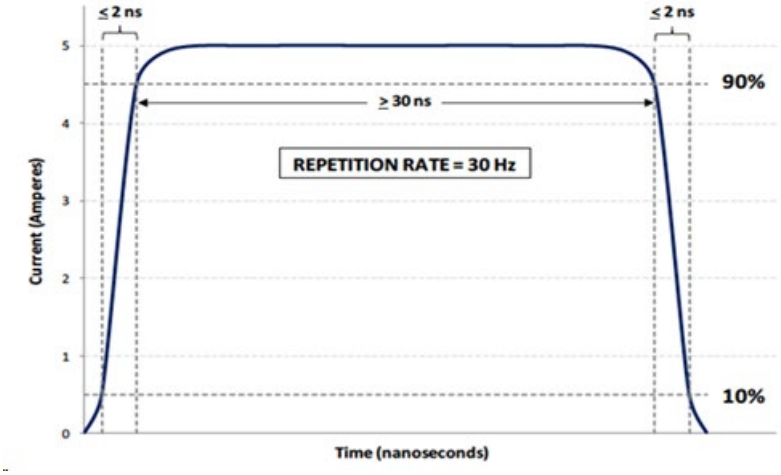

Figure 29. CS115 Impulse Waveform

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

8.3.3 Radiated Emissions

Radiated emission requirements are designed to prevent excessive noise from being radiated by the spacecraft units.

8.3.3.1 RE102 - Electric Field Emissions Limits

XRIS-1982: The instrument electric field emissions **shall** not exceed the on-orbit (blue curve) limits of Figure 30 when tested per the MIL-STD-461G RE102 test method as defined in GSFC-STD-7000B, section 2.5.2.3.2 or Contractor Standard Practice.

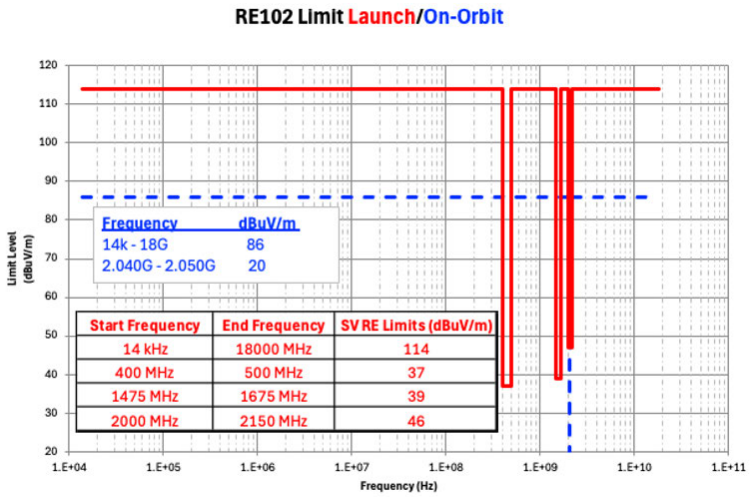

Figure 30. Unit Level RE102 Radiated Electric Field Emission Limits

8.3.4 Radiated Susceptibility

Radiated susceptibility requirements are designed to ensure that instrument units are not unduly susceptible to radiated noise emanating from the observatory and other sources.

8.3.4.1 RS103 - Radiated Susceptibility, Electric Field, Launch Limit

XRIS-1988: The instrument **shall** survive without performance degradation after being irradiated with electric field levels specified in Table 23 when in the launch configuration and tested per the MIL-STD-461G RS103 test method as defined in GSFC-STD-7000B, section 2.5.2.4.2 or Contractor Standard Practice.

Table 23. RS103 Radiated Susceptibility Levels

Start Frequency	End Frequency	LV and Range Electric Field Levels (V/m)	SV RS testing levels, includes 6 dB of margin (V/m)
14 kHz	18 GHz	10	20
400 MHz	500 MHz	18	36

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

2.2 GHz	2.4 GHz	18	36
5.4 GHz	5.9 GHz	20	40

8.3.4.2 RS103 - Radiated Susceptibility, Electric Field, On-Orbit Limit

XRIS-2017: The instrument **shall** not exhibit any malfunction, degradation of performance, or deviation from the specified indications, beyond the tolerances indicated in this specification, as a result of being irradiated with an electric field level of 2 V/m from 2 MHz to 18 GHz when tested per the MIL-STD-461G RS103 test method as defined in GSFC-STD-7000B, section 2.5.2.4.2 or Contractor Standard Practice.

8.4 Identification and Marking

XRIS-2019: Each unit **shall** be permanently marked with the part number and a unique sequential serial number in the area designated on the interface control drawing.

8.5 Workmanship

8.5.1 Connectors

XRIS-2023: GSE cable connectors that mate with flight test connectors **shall** be flight-approved connectors.

XRIS-2026: Circular connectors **shall** be positively keyed.

XRIS-2027: The instrument Spacecraft interface connectors **should** be spaced far enough apart to allow the mate and demate operations to be performed without a special tool.

XRIS-2028: The connector half that sources power to another component **shall** be female (socketed) to protect against inadvertent grounding prior to mating.

XRIS-2029: Instrument power **shall** not be applied or accessed at or through a test connector.

XRIS-2030: Each connector **shall** be labeled and clearly visible to facilitate proper mating.

XRIS-2031: Test connectors **shall** be capped with flight-approved RF and static control covers when not in use.

8.6 Reliability and Mission Lifetime

8.6.1 Mission Life

XRIS-2035: The instrument **shall** be designed for on-orbit mission life of 5 years plus 6 months of commissioning, plus an additional 2 years of integration and testing on the ground.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

XRIS-2036: The instrument **shall** be designed for a 4-year ground storage period.

8.6.2 Operating Time

XRIS-2038: A minimum of seven hundred (700) hours of operating/powered-on time **shall** be accumulated on all flight electronic hardware (divided between primary and redundant electronics) prior to shipping the instrument. Contractor Standard Practice for operating hours may be used in lieu of this requirement.

Rationale: *Accumulated power-on time that demonstrates trouble-free parts performance helps reduce the risk of failures after launch.*

8.6.3 Trouble-Free Time

XRIS-2040: At the conclusion of the performance verification program, the instrument **shall** have demonstrated trouble-free performance testing for at least the last 350 hours of operation prior to instrument shipment. Contractor Standard Practice for trouble-free hours may be used in lieu of this requirement.

Rationale: *Best practice to reduce risk of failures after launch.*

XRIS-2041: Trouble-free operation **shall** include 200 hours during the thermal-vacuum test with 100 hours being logged at the hot-dwell temperature and 100 hours being logged at the cold-dwell temperature prior to instrument shipment. Contractor Standard Practice may be used in lieu of this requirement.

Rationale: *Best practice to reduce risk of failures after launch.*

8.7 Ground Handling

8.7.1 GSE Cleanliness

XRIS-2050: All Ground and Test support equipment **shall** be compatible with the flight unit and the environment where the flight unit or test unit will reside (cleanroom, thermal vacuum chamber, vibration cell, etc.).

Rationale: *To maintain contamination control requirements compliance.*

8.7.2 GSE Bakeout

XRIS-2052: Thermal Vacuum GSE **should** be baked out and the outgassing rate certified prior to the test.

Rationale: *To maintain contamination control requirements compliance.*

8.7.3 Test Harness

XRIS-2054: Test harnesses that will be used in vacuum during ground operations **should** be vacuum compatible.

Rationale: *To maintain contamination control requirements compliance.*

9 MECHANICAL DESIGN REQUIREMENTS

For additional guidance on the design and analysis of threaded fastening systems in NASA spaceflight hardware, NASA-STD-5020B is recommended, “Requirements for Threaded Fastening Systems in Spaceflight Hardware”.

9.1 Structural Requirements

XRIS-2066: The instrument **shall** be capable of withstanding all worst-case load conditions to which it may be exposed, without requiring additional recalibration or realignment and by maintaining structural integrity (i.e., positive structural margins). This includes handling and transportation, test, pre-launch operations, launch, and on-orbit operations.

XRIS-2067: Structural margins **shall** be expressed using the following formula:

$$MS = \frac{(Allowable\ Load\ or\ Stress)}{(Factor\ of\ Safety)x\ (Limit\ Load\ or\ Stress)} - 1$$

XRIS-2069: For metallic and/or composite materials, selection and derivation of material design allowables **shall** follow the requirements defined in NASA-STD-6016, "Standard Materials and Processes Requirements for Spacecraft" or equivalent Contractor Standard Practice.

XRIS-2070: For glass/ceramic materials, selection and derivation of material design allowables **shall** follow the requirements outlined in NASA-STD-5001, “Structural Design and Test Factors of Safety for Spaceflight Hardware” or equivalent Contractor Standard Practice.

9.1.1 Component Fatigue

XRIS-2072: Thin-walled, highly loaded metallic instrument structural components serving as springs, flexures and other compliance devices under cyclic loading **shall** show sufficient fatigue strength for 4 times the number of mission cycles (including ground testing, launch and on-orbit loading) or equivalent Contractor Standard Practice.

9.1.2 Fracture Control Requirements

The use of materials that are susceptible to brittle fracture or stress-corrosion cracking require development of, and strict adherence to, special procedures to prevent problems.

XRIS-2075: If materials are used for structural applications that are not listed in Table 1 of MSFC-STD-3029, a Materials Usage Agreement (MUA) **shall** be approved by the NASA/GSFC COR.

Rationale: *To ensure that all structural applications are approved by NASA.*

XRIS-2076: Fracture control requirements (per NASA-STD-5019 or equivalent Contractor Standard Practice) **shall** apply to the following elements only:

(1) Pressure vessels, dewars, lines, and fittings (per NASA-STD 8719.24)

(2) Castings (unless hot isostatically pressed and the flight article is proof tested to 1.25 times limit load)

(3) Weldments

(4) Parts made of materials on Tables II or III of MSFC-STD-3029 if under sustained tensile stress

(5) Parts made of materials susceptible to cracking during quenching

(6) Non-redundant, mission-critical preloaded springs loaded to greater than 25 percent of ultimate strength

Rationale: *Common best practice per NASA-STD-5019*

XRIS-2077: All glass elements that are stressed above 10% of their ultimate tensile strength **shall** also be shown by fracture analysis to satisfy "Safe-life" or "Fail-safe" conditions or be subjected to a proof loads test at 1.0 times limit level or equivalent Contractor Standard Practice.

Rationale: *Common best practice per NASA-STD-5019*

9.2 Fastening Systems

9.2.1 Factors of Safety

XRIS-2080: Factors of safety for fastener strength analysis **shall** be as specified in Table 2 or equivalent Contractor Standard Practice.

9.3 Mechanisms

XRIS-2652: The instrument **shall** not require mechanisms or moving parts

10 LOGISTICS

10.1 Ground Support Equipment

10.1.1 Electrical System Test Equipment

XRIS-2151: The Electrical System Test Equipment (ESTE) **shall** operate the instrument and ground support equipment during performance verification and calibration testing.

Rationale: *ESTE serves the role of the spacecraft in providing power, telemetry, and command interfaces. In addition, it also archives the data.*

XRIS-2152: The ESTE **shall** simulate the spacecraft interface with power, clock pulses, command, and telemetry functions.

Rationale: *Needed to check the electrical interfaces to the spacecraft; risk reduction.*

XRIS-2153: The ESTE **shall** include all test equipment necessary to operate and control the instrument in all phases of I&T.

Rationale: *Enables the execution of "Test as you Fly" approach.*

XRIS-2154: The ESTE **shall** generate and maintain command logs.

Rationale: *Needed for development and any troubleshooting.*

XRIS-2155: The ESTE **shall** limit check all health and safety data.

Rationale: *Needed for development and any troubleshooting.*

XRIS-2156: The ESTE **shall** capture and archive all raw instrument data.

Rationale: *Needed for development and any troubleshooting.*

XRIS-2157: The ESTE **shall** provide near-real time and offline data analysis of all instrument data necessary to determine the performance characteristics of the instrument.

Rationale: *Needed for development and any troubleshooting.*

XRIS-2158: The ESTE **shall** interface with the Spacecraft Ground Support Equipment at the Spacecraft Contractor’s facility to extract instrument science and engineering data.

Rationale: *Needed for development and any troubleshooting.*

XRIS-2159: The ESTE **shall** prohibit hazardous or critical commands being sent to the instrument without operator verification.

Rationale: *Ensures that an inadvertent critical/hazardous cannot be sent without a “person in the loop”.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

10.1.2 Instrument Emulator

The instrument emulators will be integrated with the spacecraft simulator for data interface verification. The resulting observatory emulator will subsequently be used by the Mission Operations Center and other locations to support rehearsals, simulations, training, and command validation throughout the operational phase of the mission.

XRIS-2653: The instrument emulator **shall** process all commands received from the spacecraft simulator and change its telemetry observable state in response to those commands to emulate the actual instrument

> ***Rationale:*** *Being clear what we mean by an emulator. This functionality will be important when integrated into the OpsSim.*

XRIS-2654: The instrument emulator **shall** support the entire Command and Telemetry database (CDRL item OPS-1) for the instrument.

> ***Rationale:*** *If commands or telemetry are missing in the emulator, this will lead to inaccuracies and inefficiencies in conducting rehearsals or simulations.*

XRIS-2655: The instrument emulator telemetry transferred to the Spacecraft Simulator **shall** be in the exact format as specified in the command and telemetry database.

> ***Rationale:*** *If telemetry is not formatted correctly, it can't be processed by the ground system*

XRIS-2656: The instrument emulator **shall** reject any commands with invalid fields or incorrect formatting in a manner identical to the flight instrument

> ***Rationale:*** *Facilitates troubleshooting of commands, scripts, and interfaces*

XRIS-2162: The instrument emulator **shall** support scripting by end users to simulate predefined anomalies. These anomalies should be credible per the instrument FMECA observable either in telemetry or by the loss of telemetry, and should include software faults.

> ***Rationale:*** *To support mission rehearsals, contingency procedure development, and training.*

XRIS-2163: The instrument emulator **shall** communicate with a spacecraft simulator for instrument command, telemetry, and the 1 Pulse Per Second (PPS) signal using an interface that matches the instrument design electrically and logically.

> ***Rationale:*** *To be compliant with the spacecraft clock and data signals.*

XRIS-2657: The instrument emulator **shall** update its time in a manner identical to the flight unit when a time message is received from spacecraft simulator, unless this update has been disabled by a simulator command.

> ***Rationale:*** *to automatically bring instrument emulators in line with simulation time as the start of simulation*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

XRIS-2164: The instrument emulator hardware **shall** have all instrument to spacecraft data interfaces with the exception of DC power.

Rationale: *The emulator will support interface risk reduction.*

XRIS-2165: The instrument emulator **shall** use commercial power.

Rationale: *Provides flexibility to be used in multiple test/calibration facilities.*

XRIS-2166: The instrument emulator **shall** include all hardware and software required to enable end users to configure the emulator state and to issue simulation control commands.

Rationale: *Provides training and testing flexibility.*

XRIS-2167: The instrument emulator **shall** include a secondary TCP/IP interface for purposes of configuration and issuing simulation control commands.

Rationale: *Enables a "test as you fly" approach. Needed for development and any troubleshooting.*

XRIS-2168: The instrument emulator **shall** generate housekeeping data reflective of all instrument modes.

Rationale: *Needed for development and any troubleshooting. GOLD Rule 3.05*

XRIS-2169: The instrument emulator **shall** accept real-time simulation control commands to change simulated telemetry or modeling parameters.

Rationale: *Enables a "test as you fly" approach. Needed for development and any troubleshooting.*

XRIS-2658: The instrument emulator **shall** accept the same flight software and flight table load commands from the spacecraft simulator as the flight instrument, and update memory or tables accordingly.

XRIS-2659: The instrument emulator **shall** accept and process commands to dump memory or tables and respond with the appropriate dump packets in telemetry

XRIS-2170: The instrument emulator **shall** maintain a human readable log of all simulation control commands and flight commands received with both UTC wall clock and simulation time recorded.

Rationale: *Needed for development and any troubleshooting.*

XRIS-2660: The instrument emulator **shall** support making any required changes to emulator software on site.

10.2 Transportation Equipment

10.2.1 Shipping Container

XRIS-2174: The instrument shipping container **shall** meet all environmental requirements imposed on the instrument consistent with the storage and shipping environment.

> ***Rationale:*** *Environmental requirements compliance needs to be demonstrated during all phases through launch.*

XRIS-2175: The instrument shipping container **shall** provide internal temperature and humidity monitors with external indicators.

> ***Rationale:*** *To provide knowledge of the actual shipping environment.*

XRIS-2176: The instrument shipping container **shall** provide shock recorders.

> ***Rationale:*** *To provide knowledge of the actual shipping environment.*

XRIS-2177: The instrument shipping container **shall** meet all contamination control requirements imposed on the instrument units.

> ***Rationale:*** *To ensure that the contamination control requirements are met while shipping or storing the instrument.*

11 VERIFICATION REQUIREMENTS

XRIS-2180: The contractor **shall** conduct a verification program that demonstrates the unit design is qualified. Per the instrument SOW, the contractor will provide a verification matrix defining the method of verification for each specific requirement of this document.

11.1 Test Requirements

This section provides general test requirements on how testing is to be performed in the process of verifying that the deliverable item meets its requirements.

XRIS-2204: Performance parameter measurements **shall** be taken to establish a baseline that can be used to assure that there are no data trends established in successive tests that indicate a degradation of performance trend within specification limits that could result in unacceptable performance in flight.

XRIS-2205: Any requirement that exceeds previous qualification test data **shall** be presented to the NASA/GSFC COR as part of the verification planning process described in the SOW, for evaluation and a possible delta qualification test.

Rationale: *To ensure that all exceedances are reviewed and accepted by NASA.*

11.1.1 Definitions

The hardware definitions are reproduced here from Section 1.8 of GEVS (GSFC-STD-7000B).

Prototype Hardware: “Hardware of a new design; it is subject to a design qualification test program; it is not intended for flight.” The purpose of the tests on this hardware is to prove that a new design meets one or more of its design requirements. Qualification testing is performed at maximum expected flight levels plus a margin. Test durations are typically longer than for acceptance tests.

Protoflight Hardware: “Flight hardware of a new design; it is subject to a qualification test program that combines elements of prototype and flight acceptance verification; that is, the application of design qualification test levels and flight acceptance test durations.” The purpose of the test on this hardware is to prove that a new design meets one or more of its design requirements. Protoflight testing is performed at maximum expected flight levels plus a margin. Test durations are typically the same as for acceptance tests.

Follow-On (Acceptance) Hardware: “Flight hardware built in accordance with a design that has been qualified either as prototype or as protoflight hardware; follow-on hardware is subject to a flight acceptance test program.” The purpose of the test on this hardware is to prove that a particular flight unit has been manufactured properly. The design has already been proven during a qualification or protoflight test program. Acceptance testing is performed at maximum expected flight levels.

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

11.1.2 Test Factors

XRIS-2212: The following test factors and durations, shown in Table 24, **shall** be used for prototype, protoflight, and follow-on flight hardware. Equivalent Contractor Standard Practice Test Factors and Duration are acceptable.

Table 24. Test Factors and Durations

Test	Prototype Qualification	Protoflight Qualification	Acceptance
Structural Loads[1] Level	1.25 x Limit Load	1.25 x Limit Load	Limit Load
Duration Centrifuge/Static Load[4] Sine Burst	1 Minute 5 cycles @ full level per axis	30 Seconds 5 cycles @ full level per axis	30 Seconds 5 cycles @ full level per axis
Acoustics Level[2] Duration	Limit Level +3dB 2 minutes	Limit Level +3dB 1 minute	Limit Level 1 minute
Random Vibration Level[2] Duration	Limit Level +3dB 2 minutes/axis	Limit Level +3dB 1 minute/axis	Limit Level 1 minute/axis
Sine Vibration[3] Level Sweep Rate	1.25 x Limit Load 2 oct/min	1.25 x Limit Load 4 oct/min	Limit Load 4 oct/min
Mechanical Shock Actual Device Simulated	2 actuations 1.4 x Limit Level 2 x Each Axis	2 actuations 1.4 x Limit Level 1 x Each Axis	1 actuations Limit Level 1 x Each Axis
Thermal-Vacuum	Max./min. predict. ± 10°C	Max./min. predict. ± 10°C	Max./min. predict. ± 5°C
EMC & Magnetics	As Specified for Mission	Same	Same

1 If qualified by analysis only, positive margins must be shown for factors of safety of 2.0 on yield and 2.6 on ultimate. Beryllium and composite materials cannot be qualified by analysis alone.

Note: Test levels for weldments, beryllium, bonded and composite structure, including metal matrix, are 1.25 x Limit Level for both qualification and acceptance testing

2 As a minimum, the test level must be equal to or greater than the workmanship level.

3 The sweep direction should be evaluated and chosen to minimize the risk of damage to the hardware. If a sine sweep is used to satisfy the loads or other requirements, rather than to simulate an oscillatory mission environment, a faster sweep rate may be considered, e.g., 6-8 oct/min to reduce the potential for over stress.

4 Shorter durations may be used in static testing if necessary to protect the hardware from damage due to facility limitations. If a shorter duration is used then the dwell time at load should be sufficient to demonstrate that the target loading condition has been

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

achieved within the specified tolerances, all test measurements have been recorded, and the structure is stable under the applied loading condition.

11.1.3 Test Tolerances

The test condition tolerances given in GEVS section 1.13 may be used as a starting point for the instrument test program. Equivalent Contractor Standard Practices are acceptable.

XRIS-2265: Tolerances for test program parameters **shall** be specified in the individual component or instrument test plans or procedures.

11.1.4 Test Restrictions

11.1.4.1 Failure During Tests

XRIS-2268: When a failure (non-conformance or trend indicating that an out-of-spec condition will result) occurs, determination will be made as to the feasibility and value of continuing the test to its specified conclusion. The test **shall** be stopped if equipment fails during testing in cases where this failure will result in damage to the equipment.

XRIS-2269: Otherwise, the test **shall** be completed to obtain as much information as possible. If corrective action is taken, the test will be repeated to the extent necessary to demonstrate that the test item's performance is satisfactory. If corrective action taken as a result of failure affects the validity of previously completed tests (e.g., redesign of a unit), prior tests will be repeated.

If during a test sequence, a test item is operated in excess of design life and wears out or becomes unsuitable for further testing from causes other than deficiencies, a spare will be substituted, and previously completed tests will be repeated to the extent necessary.

XRIS-2271: No replacement, adjustment, maintenance, or repairs are authorized during testing. This requirement does not prevent the replacement or adjustment of equipment that has exceeded its design operating life during tests, provided that after such replacement, the equipment is tested as necessary to assure its proper operation. A complete record of any exceptions taken to this requirement **shall** be included in the test report.

11.1.4.2 Modification of Hardware

XRIS-2273: Once the formal acceptance test has started, cleaning, adjustment, or modification of test hardware **shall** not be permitted.

11.1.4.3 External Adjustment

XRIS-2275: The instrument **shall** be designed so that no external adjustments are required after start of acceptance or qualification testing.

11.1.4.4 Re-Test Requirements

XRIS-2277: If any event, including test failure, requires that the instrument be disassembled and reassembled, then all tests performed prior to the event **shall** be considered for repeat. If the unit has multiple copies of the same build, then all units must be examined to determine if the problem is common. If all copies require disassembly for repair, then each must receive the same test sequence.

11.1.4.5 Configuration

XRIS-2279: The instrument **shall** undergo mechanical environmental testing in a flight like configuration which includes mechanisms configured for flight/launch (if applicable), and with flight or flight-like blankets and harnesses present.

Rationale: *Common best practice.*

11.2 Required Tests

XRIS-2281: The following tests **shall** be performed for each flight unit to provide assurance that the instrument meets all of its requirements. This requirement does not require testing when other requirements give alternative paths to verification. Each test or demonstration is described below:

Performance Testing

Mass Properties

Static Load/Strength

Sine Sweep Survey

Sine Vibration

Random Vibration

Acoustics

Shock

Electromagnetic Compatibility

Thermal Vacuum Bake-Out

Thermal Vacuum

Thermal Balance

Magnetics Testing

11.2.1 Performance Tests

11.2.1.1 Comprehensive Performance Test

XRIS-2284: The Comprehensive Performance Test (CPT) **shall** be used to verify full compliance of each flight unit to all of its performance requirements, within the limitations of the environment and facilities.

XRIS-2285: The CPT **shall** be designed to verify unit performance in all modes and configurations and under varying input/output conditions.

11.2.1.2 Limited Performance Test

The Limited Performance Test (LPT) is a subset of the CPT and is designed to verify functionality of the flight unit under nominal input/output conditions. The LPT will be used to verify unit operation when it is not practical to use the CPT. Typically, it is used during and after certain environmental tests to demonstrate that the functional capability of the unit has not been degraded due to environmental exposure, handling, and transportation. It is also commonly used during some of the hot and cold plateaus during thermal vacuum testing.

11.2.1.3 Abbreviated Functional Test

An Abbreviated Functional Test (AFT) may be used when it is necessary to monitor the performance of the flight unit over a very short period of time. The AFT is typically used when a unit is being subjected to vibration testing.

11.2.2 Mass Properties Measurement

Measurement of the weight and center of gravity of each flight hardware unit will be made to show compliance with requirements and to provide accurate data for the observatory mass properties control program. Center of gravity at the unit level will be referenced to the unit to spacecraft mounting interface.

11.2.3 Static Loads/Strength Test

Strength testing is used to verify the unit strength and structural integrity (beryllium and composite materials cannot be qualified by analysis alone), and it can be done using a variety of techniques, such as sine burst, static pull, or centrifuge testing.

XRIS-2294: If required (i.e., qualification by analysis only is not planned), structural design loads **shall** be applied to prototype or protoflight hardware. There is no requirement to strength test flight hardware that has already been strength tested through a prototype or protoflight program (i.e., there is no “acceptance level” strength test requirement for flight hardware).

No permanent deformation may occur as a result of the loads test, and all applicable alignment requirements must be met following the test. Units that require alignment will have an alignment check following loads testing.

XRIS-2296: A performance test **shall** be conducted after the static loads test to verify that no damage occurred due to the loads test.

XRIS-2297: A functional test **shall** be performed before the start of testing and after a test in each axis if sine burst is used.

11.2.3.1 Sine Burst

If sine burst test is selected, the test will be performed in each of three orthogonal unit axes. A simple Sine Burst test following the random vibration test in each axis is a convenient method to conduct a structural loads test. Test frequency will be less than one-third of the resonant frequency of the unit to avoid dynamic amplification during test. This test applies a ramped sine input at a sufficiently low frequency such that the test item moves as a rigid body. The test will be conducted in a stepwise manner starting with lower-level sine-bursts that are fractions of the full load. An analysis is required to show that a base drive Sine Burst test will not cause over-test or under-test in some areas of the structure. The number of cycles at maximum level will be at least 5.

11.2.3.2 Static Pull

Static pull tests are another method to perform loads testing and can be applied at flight interfaces in a static test facility. The loads can be applied either as instrument loads applied simultaneously, or the single resultant vector load can be applied to the test point. Strain gages are generally positioned around the test point to verify deflection predictions from the analytical model. The test duration should be at least 30 seconds.

11.2.4 Sine Sweep Survey

XRIS-2303: The Sine Sweep **shall** be conducted on each XRIS before and after vibration testing in each axis. A low-level sine sweep is used to determine the modal signature of the flight item to verify compliance with fundamental frequency requirements, and to verify no change in structural integrity from testing. Test parameters will be as follows:

Frequency: 5 - 2000 Hz if performing random vibe testing; 5-150 Hz when performing sine vibe testing only (no random)

Acceleration: typically 0.1 g - 0.25 g, as required by hardware amplification

Sweep Rate: 2 octaves/minute

XRIS-2307: This test will be performed immediately preceding the sine/random vibration tests and will be repeated after the sine/random vibration tests to verify that the modal signature of the unit under test (first resonant frequency) is within 5% of the pre-test frequency. In this case, the

unit has passed and the test can continue. If the post-test frequencies are between 5% and 10% of the pre-test frequencies, the cause of the shift should be investigated (test bolt torques, etc.) before determining whether failure has occurred or success has been achieved and whether the test can continue. A shift of greater than 10% indicates failure, and the test should be terminated, and the failure documented. In either case when the shift is greater than 5%, NASA/GSFC **shall** be notified.

XRIS-2308: Frequencies **shall** be verified and reported up to 200 Hz. Any unit that fails to meet the fundamental frequency requirements specified in Section 6.3.1 is required to submit a test-correlated model. If a test-correlated model is required, all significant modes up to the required frequency must be determined both in terms of frequency and mode shape. A mode is considered significant if it has modal effect mass ≥ 5% of the total mass of the hardware at the level of assembly for which the modal survey is being considered. Modes that drive high responses of critical units from a coupled loads analysis should also be considered as target modes for a modal survey.

XRIS-2309: Cross-orthogonality checks of the test and analytical mode shapes, with respect to the analytical mass matrix, **shall** be performed with the goal of obtaining at least 0.9 on the diagonal and no greater than 0.1 off-diagonal for acceptable model correlation.

XRIS-2310: Frequencies between the corresponding test and analytical modes **shall** also match within 5%.

11.2.5 Sine Vibration

XRIS-2312: If required, the instrument **shall** be subjected to swept sine vibration testing to the appropriate levels and durations shown in the mechanical requirements section. Sine vibration test is intended to verify workmanship quality and to simulate launch vehicle loading conditions. The test will consist of a low frequency sine transient or sustained sine environment that would be present in launch by minimum modal frequency in each of three mutually perpendicular axes, one of which is normal to the mounting surface.

XRIS-2313: For the sine vibration tests, the unit under test will be attached to the vibration table using the same configuration, attachment points, preloads, and attachment hardware that will be used during launch and flight. Cross-axis responses of the fixture will be monitored to preclude unrealistic levels. During the test, the test input level will be reduced (notched) at critical frequencies, if required, to limit the vibration loads and/or acceleration responses to 1.25 times design limit levels. A functional test **shall** be performed before the start of testing and after a test in each axis.

11.2.6 Random Vibration

XRIS-2315: The instrument **shall** be subjected to a random vibration test in each of three orthogonal axes of the unit under test, one axis being perpendicular to the mounting surface to the appropriate levels and durations shown in the environmental requirements section. Random vibration test is intended to demonstrate workmanship quality and to simulate launch vehicle aerodynamic environmental levels. During the test, the test input level will be reduced (notched)

at critical frequencies, if required, to limit the random vibration loads and/or acceleration responses to 1.25 times design limit levels. Notching will be limited to -12 dB of the original input and to a bandwidth of less than 100 Hz to limit hardware responses to 3dB above design limit loads. Notching beyond these limits will require approval by the NASA/GSFC COR.

> For the random vibration test, the unit under test will be attached to the vibration table using the same configuration, attachment points, preloads, and attachment hardware that will be used during launch and flight. Prior to the test, a survey of the test fixture/exciter combination will be performed to evaluate the fixture dynamics and the proposed choice of control accelerometers. Cross-axis responses of the fixture will be monitored during the test to preclude unrealistic levels.

XRIS-2317: A functional test **shall** be performed before the start of testing and after each axis test.

11.2.7 Acoustic Test

XRIS-2319: If an acoustic test at the instrument level is required, it **shall** be conducted in an area large enough to maintain a uniform sound field at all points surrounding the unit under test.

The sound pressure levels at various frequencies are specified in section 6.6.

XRIS-2321: A functional test **shall** be performed before the start of testing and after the acoustic test.

11.2.8 Shock

Shock testing is designed to verify that flight hardware will survive expected shock events such as observatory separation, solar array deployment, etc.

The shock test at the unit level is not required if previous qualification data showing adequate margin exists or it can be shown by analysis that expected (attenuated) shock levels are enveloped by the curve presented in section 6.5. Items that are more likely to be shock sensitive include very brittle materials (e.g., glass, ceramics), wire bonds, sensors, transducers, and filters.

If the instrument is deemed shock sensitive by the NASA/GSFC COR and margin (per Table 24) cannot be demonstrated based on prior testing, then a unit level shock test should be performed.

Shock testing may be performed at the sub-unit or part level so as to limit the test risk to the flight hardware.

The unit shock testing approach should be coordinated with the project and consistent with the approach described in Section 2.4.4.1 of GSFC-STD-7000B.

If the instrument is not deemed shock sensitive by the Contractor, then qualification for the shock environment is not required, and the shock testing can be deferred to the level of assembly that allows for actuation of the actual shock-producing device.

11.2.9 Thermal Vacuum Bake-out

XRIS-2330: The instrument **shall** be baked-out prior to delivery to NASA GSFC to reduce outgassing of contaminants per the requirements in Section 7.3 of this document.

11.2.10Thermal Vacuum Test

11.2.10.1 Thermal Vacuum Test Parameters

XRIS-2333: Thermal vacuum testing **shall** be conducted in accordance with the requirements of Table 25 or equivalent Contractor Standard Practice.

Table 25. Thermal Vacuum Test Parameters

Parameter	Instrument Level
Chamber Pressure After Evacuation	$\leq 1 \times 10^{-5}$ Torr
Number of Cycles	8
Acceptance Temperature Limits (beyond operational limits)	5 °C
Proto-flight Temperature Limits (beyond operational limits)	10 °C
Minimum Dwell Time (at cold and hot plateaus)	12 hrs.
Total Number of Comprehensive Performance Tests (at least one each at cold and hot plateaus)	≥ 2
Expected Temperature Rate of Change	3 °C to 5 °C/min

[1] For subsystems/systems having units with redundant, fixed temperature (setpoint) operational heater control, the cold end margin for unit level testing of those units may be reduced from 10°C to 5°C, as long as the heater control duty cycle requirement (Requirement 635) is met. Higher level testing with the heater control enabled need not attempt to drive the hardware below the setpoint.

11.2.10.2 Thermal Vacuum Test Profile

XRIS-2359: The instrument **shall** be exposed to Thermal Vacuum Testing in accordance with Contractor Standard Practices or GEVS.

11.2.11 Magnetics Test

XRIS-2375: The magnetic field test **shall** measure the peak-to-peak change in magnetic field produced by each unit for all instrument operating modes.

Rationale: *To demonstrate compliance with the transient magnetic field requirement.*

XRIS-2376: Measurements **shall** be made in the X, Y and Z-axis of all units for both primary and redundant configurations.

Rationale: *To demonstrate compliance with the transient magnetic field requirement.*

The measured change in the magnetic field strength will be the difference between the ambient background level of the instrument and the maximum magnetic field induced by the change in the instrument's state.

XRIS-2378: A reference probe, or another technique, **shall** be used upon approval to exclude anomalous data caused by external events such as opening and closing of doors.

Rationale: *To ensure measurement of magnetic field of the instrument does not include ambient background field variations.*

XRIS-2379: The magnetic field test **shall** measure the permanent magnetic field produced by each unit.

Rationale: *To demonstrate compliance with the static magnetic field requirement.*

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

Appendix A Abbreviations and Acronyms

Abbreviation/ Acronym	Definition
AC	Alternating Current
AFT	Abbreviated Functional Test
Al	Aluminum
BER	Bit Error Rate
CCB	Configuration Control Board
CCP	Contamination Control Plan
CCR	Configuration Change Request
CE	Conducted Emissions
CG	Center of Gravity
CM	Configuration Management
CMO	Configuration Management Office
CMOS	Complementary Metal Oxide Semiconductor
COR	Contracting Officer's Representative
CPT	Comprehensive Performance Test
CS	Conducted Susceptibility
CVCM	Collected Volatile Condensable Mass
DA	Double Amplitude
DC	Direct Current
DDD	Displacement Damage Dose
DILS	Deliverable Items List and Schedule
EED	Electro-Explosive Actuators
EEE	Electrical, Electronic, and Electromechanical
ELDR	Enhanced Low Dose Rate
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
EOL	End of Life
ESD	Electrostatic Discharge
EUT	Element Under Test
FS	Factor of Safety
GeBK	Germanium Black Kapton
GSE	Ground Support Equipment
GSFC	Goddard Space Flight Center
HAP	High Availability Product
ICD	Interface Control Document
I&T	Integration and Test
ITO	Indium Tin Oxide
LET	Linear Energy Transfer
LOS	Line Of Sight
LPT	Limited Performance Test
LVDS	Low Voltage Differential Signal

Check https://ipdtdms.gsfc.nasa.gov/frontmenu_dsp.cfm to verify that this is the correct version prior to use.

MBU	Multi Bit Upset
MICD	Mechanical Interface Control Document
MLI	Multi-Layer Insulation
Mohms	Megaohms
MOP	Maximum Operating Pressure
MOSFET	Metal Oxide Semiconductor Field-Effect Transistor
MS	Margin of Safety
NASA	National Aeronautics and Space Administration
NEA	Non-Explosive Actuators
NIEL	Non-Ionizing Energy Loss
OSR	Optical Solar Reflector
PACE	Plankton, Aerosol, Cloud, ocean Ecosystem
PDL	Product Design Lead
QCM	Quartz Crystal Monitor
RE	Radiated Emissions
RF	Radio Frequency
RS	Radiated Susceptibility
S/C	Spacecraft
SCoRe	Signature Controlled Request
SEB	Single Event Burnout
SEGR	Single Event Gate Rupture
SEE	Single Event Effects
SEFI	Single Event Functional Interrupt
SEL	Single Event Latchup
SEM	Scanning Electron Microscope
SERDES	Serializer/Deserializer
SEU	Single Event Upset
SHE	Single Hard Error
SOW	Statement of Work
SPL	Sound Pressure Level
TBD	To Be Defined
TBP	To Be Proposed
TBR	To Be Reviewed
TID	Total Ionizing Dose
TML	Total Mass Loss
UUT	Unit Under Test
VDA	Vapor Deposited Aluminum
VDC	Voltage, Direct Current
XRIS	X-Ray Irradiance Sensor