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Korea Lunar Exploration Program

KPLO

Environmental Design and Test Specification

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REVISION / CHANGE RECORD

Revision	DOCUMENT DATE	REVISION / CHANGE DESCRIPTION	PAGES AFFECTED
Draft 00	11 August 2016	Initial Release	All
Preliminary 00	10 February 2017	Reference of thermal testing changed from to KOMPSAT EV spec to GK2-SP-320-006 F.00(GK2 EV spec) Updated bellows. - Launch vehicle requirements - Temperature extremes - External Pressure Extremes - Component Life - Radiation - Environmental Tests	All
Final 00	18 September 2017	. Launch Vehicle Requirements was updated . 140 Hz [TBD] in Minimum Natural Frequency of components was deleted . Equipment Temperature Limits was updated . TBD in Equipment Temperature Limits Table was deleted . Random Vibration level unit was updated - Payload, Generic instruments, CSSA, SBA, SADA . Table 3-7 was updated . Shock level of spacecraft was defined . 3.1.9 Orbital Radiation Dose levels was changed . 3.1.11 Micrometeoroids chapter was added newly for Solar Array design . Table 4-2 [TBD] delete . Table 4-3 [TBD] delete	1 6 9,10 11,15 16 18~19 23 30,31 43 44
Final 01	27 June 2018	. Table 3-1 note(A) at page9 was deleted . Below units temperature range was updated - ShadowCam, XTXU, HGAA, RWA, CSSA, STA-EU, SXP, SBA, PCA, Fill Drain Valve, Latching Isolation Valve, Pressure Transducer . Table 3-2 title was modified . CSSA, SBA, APMU and APEU random vibration level was updated . Table 3-7 was modified - Target frequency was modified 0-20 to 5-20 . Table number was modified. - Table 3-6 ~ Table 3-16 -> Table 3-7 ~ Table 3-17 - Figure 3-6 ~ Figure 3-11 -> Figure 3-7 ~ Figure 3-12 . Table 3-10 i) description was modified - OLD: +/- Y, Z Lower Closure Wall(b) NEW: +/- Y, +/- Z Lower Closure Wall (b) - OLD: +/- Y, Z Upper Closure Wall (c) NEW: +/- Y, +/- Z Upper Closure Wall (c) . Antenna deployment shock level was added in Table 3-10 i), ii) . Table 4-3 was updated	9 9, 10, 11 13~16 18 Refer to Table and Fig number. 20 21

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1. Introduction

This specification provides the environmental design and test requirements for flight components to be installed onto the Korea Pathfinder Lunar Orbiter (here after KPLO). These design and test requirements are guided by MIL-STD-1540 B/C and GEVS (General Environmental Verification Specification) of NASA. Based on the guided specification, tailoring is made for KPLO. The specific test environments and procedures required for a component shall be selected from these requirements based on an evaluation of its intended use, location, environmental sensitivity, and requirements stipulated in the applicable equipment (EQ) specification.

1.1 Scope

The requirements that flowed down from the KPLO system verification specification establish the basis for the environmental design requirements, and the proto-flight and acceptance test requirements for the flight component. Criteria set forth herein shall be incorporated in appropriate specifications for all components. All requests for deviation from the requirements or procedures of this specification shall be processed in accordance with section 1.3.

Overall launch environment of KPLO is summarized in Table 1-1.

Table 1-1. Launch Vehicle Requirements (TBR)

Item	Launch Vehicle
Quasi-Static Loads in LV Axis :	
• Axial	+ 2.8 g / -6.5g (- : in compression)
• Lateral	± 2.0 g
Acoustic Noise	143.0 dB
Shock, Maximum	2,000 g
Payload Frequency$^{(A)}$:	
• Lateral, Min.	14.0 Hz
• Longitudinal, Min.	37.0 Hz

(A) Payload frequency without separation system

1.2 Definitions

Terminology and associated definitions which relate to environmental testing are presented in the following sections.

1.2.1 Test Terms

1.2.1.1 Qualification Tests

Tests intended to demonstrate that the test item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch, and orbital operations. Their purpose is to uncover deficiencies in design and method of manufacture. They are not intended to exceed design safety margins or to introduce unrealistic modes of failure. The design qualification tests may be to either "prototype" or "proto-flight" test levels.

1.2.1.2 Acceptance Tests

Acceptance tests are the environmental, electrical, mechanical, and other tests which all components and assemblies scheduled for flight must pass before launch. The tests are planned to approximate the maximum environmental conditions expected in normal operation (i.e., the limit level defined in section 1.2.1.3). They are designed to verify that the components are free of workmanship and material defects.

1.2.1.3 Proto-flight Tests

Proto-flight tests are formal demonstrations that the design, manufacturing and assembly have resulted in hardware that conforms to the specification requirements. These tests subject the hardware to greater rigors of environment and operation than expected in normal operation. The levels are selected such that the components need not be refurbished prior to flight.

Note: Proto-flight (MIL-STD-1540B) and Proto-qualification (MIL-STD-1540C) model will have the same meaning in this specification. The terminology "Proto-flight" will be used hereafter in this specification.

1.2.1.4 Limit Level

The limit level is the maximum expected flight condition. Design and test margins are related to limit level.

1.2.1.5 Limited Functional Test

Limited functional tests serve to monitor interacting functions (basic end-to-end and go-no-go) and input-output validation. The objective is to demonstrate functional capability and flight-worthiness of hardware. Typically, they are periodic tests performed throughout the test program.

1.2.1.6 Full Functional Test

Full functional tests are conducted on a component for comprehensive performance measurements.

1.2.1.7 Test Margin

The factor by which a maximum operating flight environment is increased in order to establish a test environment which will ensure that the test objectives are achieved.

1.2.1.8 Test Discrepancy

A functional or structural anomaly which occurs during testing and which indicates a possible deviation from requirements for the test item. A test discrepancy may be a momentary, non-repeatable, or permanent failure to respond in the predicted manner to a specified combination of environmental and functional test stimuli. Such discrepancies may be due to a failure of the test component or to some other cause such as the test setup, test instrumentation and hardware, supplied power, the test procedures, or to the computer software used.

1.2.2 Classifications

1.2.2.1 Subsystem

It is a collection of components having similar mounting or functional relation.

1.2.2.2 Component, Equipment or Unit

It is a functional subdivision of a subsystem and generally is a self-contained combination of assemblies performing a function necessary to the subsystem's operation. Typical components include electronic equipment (e.g., black boxes), batteries, solar panels, etc., and mechanical subassemblies which perform specific functions.

1.2.2.3 Assembly

It is the next lower functional subdivision of a component and consists of parts or subassemblies which perform functions necessary to the operation of the component as a whole. Typical assemblies include filters, amplifiers, etc.

1.2.2.4 Subassembly

It is an element of an assembly in which two or more parts are joined together to form an item which is capable of disassembly or part replacement.

1.2.2.5 Part

It is an element of a component, assembly, or subassembly which is not normally subject to further subdivision or disassembly without destruction of its designated use. Typical parts include such items as resistors, integrated circuits, capacitors, gears, screws, etc.

1.2.2.6 Engineering Model

It is a component, which is used for verifying electrical performance, interface compatibility, and functional design.

1.3 Deviations

Only those conditions which are indicative of actual service usage and which can adversely affect the performance of the structure or component(s) shall be considered as the basis for design and testing. In the event that any specified requirement or tests do not apply, or if additional requirements or tests are necessary, or tests can be performed at a higher level of assembly, such recommendations shall be submitted in the form of CRFD (Components Request for Deviation) with substantiating analyses and documentation for consideration and approval.

1.4 Qualification by Similarity

In the event that the component selected may have been qualified previously to test environments equal to or greater in severity than the environments specified herein, a specific request for qualification by similarity may be submitted if data from earlier tests can be applied to this program. The written request shall include a summary of the applicable test data package and pertinent correlative analysis, and be submitted to the Change Control Board for review and approval.

All components, including those qualified by similarity, shall require acceptance testing. However, any unit which has been previously subjected to proto-flight test environments are not required to be acceptance tested again.

2. Applicable Documents

The following documents are a part of this specification to the extent specified herein. If no revision or date is specified, the latest issue in effect shall apply. In the event of a conflict between the referenced documents and details of this specification, the detailed requirements herein shall be considered superseding unless otherwise noted.

2.1 KPLO Documents

Document No.	Title
KPLO-D0-312-003	KPLO Verification Plan
KPLO-SP-410-001	KPLO Spacecraft Bus Specification
KPLO-SP-330-001	KPLO EMC Requirements Specification
KPLO-SP-5X0-001	KPLO Payload Specification
KPLO-D0-210-003	KPLO Subcontractor PA Requirement

2.2 Other documents (For Reference Only)

Document No.	Title
MIL-STD-1540B	Test Requirements for Space Vehicles
MIL-STD-1540C	Test Requirements for Launch, Upper-Stage, and Space Vehicles
GEVS-SE Rev A	General Environmental Verification Specification
PSS-01-801	Test Requirements Specification for Space Equipment
ECSS-E-10-03-A	ECSS Space Engineering-Testing

3. Requirements

3.1 Environmental Design

This section presents the design and interface requirements in mechanical, thermal and electrical points of views, which must be met by equipment to be used on KPLO.

The equipment shall follow the design requirements hereafter unless otherwise specified in the EQ (equipment) specification. A waiver shall be raised if there is any confliction and it will be processed by agreement between the equipment supplier and KARI.

The documents specified in section 2.2 could be referred for detailed design requirements.

3.1.1 Minimum Natural Frequencies

Components (black boxes) shall be designed such that the minimum natural frequency is greater than 140 Hz in any direction. Exceptions will be considered for component which, due to configuration and other design constraints, has inherently low stiffness. If specific exceptions are needed, those shall be documented in the relevant document.

3.1.2 Temperature Extremes

The acceptance and qualification temperature ranges of all payload and spacecraft components shall be derived from the thermal interfaces following the margin rules specified below.

The maximum and minimum temperature limits including operating, acceptance and qualification limits are shown in Figure 3-1.

All predicted temperatures are shown with ±5 ℃ or ±10 ℃ (internal or external units) calculation uncertainty except for the units controlled by heaters where cold end margin is removed. Uncertainty on the thermal H/W is taken into account in the determination of the heater set points.

The acceptance and qualification limits shall include a thermal margin 5 ℃ and 10 ℃ respectively above and below the worst case design temperature (operating) expected in flight.

Equipment acceptance temperatures are :

$T_A min = T_O min - 5℃$

$T_A max = T_O max + 5℃$

Equipment qualification temperature are :

$T_Q min = T_O min - 10℃$

$T_Q max = T_O max + 10℃$

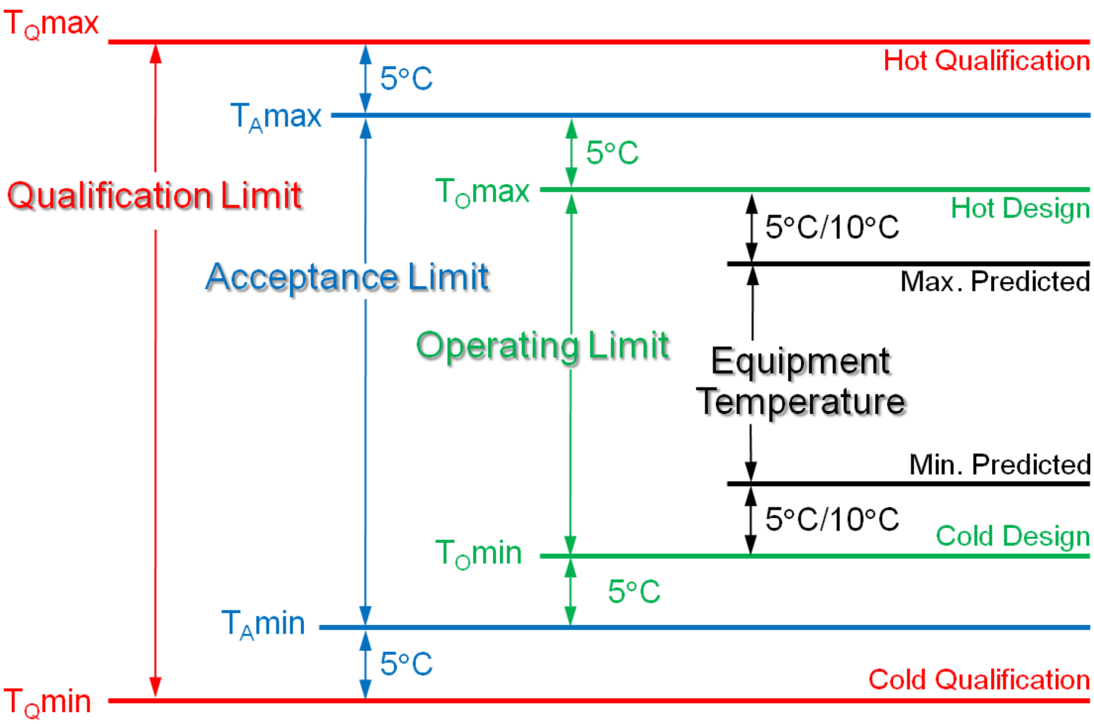


Figure 3-1. Temperature Design and Test Limits

- T_O (Operating Temperature) : The operating temperature limits specify the temperature extremes at and between which the functioning equipment shall be designed to operate whilst meeting all the specification requirements.
- T_A (Acceptance Temperature) : The acceptance temperature limits specify the temperature extremes to which flight model equipment must be subjected during thermal testing, in operating conditions, whilst fully meeting their performance requirements.
- T_Q (Qualification Temperature) : The qualification temperature limits specify temperature extremes to which qualification and proto-flight model equipment must be subjected during thermal testing, in operating conditions, whilst fully meeting their performance requirements.
- T_N (Non-operating Temperature) : The non-operating temperature limits specify the temperature extremes at and between which the equipment shall be designed to survive without functioning for any period of the mission and without performances degradation.

- T_S (Start-up Temperature) : The start-up temperature specifies the minimum temperature for the equipment to have power applied, or to be activated. The equipment design shall be such that upon such power application or activation with temperature moving into the operating range the equipment performance reaches specification requirements.

Spacecraft equipment temperature limits are specified in Table 3-1 and the temperature limits for required environmental tests are specified considering the relations defined in Figure 3-1. Equipment shall perform within the temperature ranges throughout the proto-flight and acceptance tests.

3.1.3 Thermal Flux

Heat removal from the component will occur by a combination of conduction from the base plate and radiation from the remaining faces of the unit. The proportion between radiation and conduction will depend on the individual box design and location. Since external radiation effects are location dependent, units shall be designed to maintain reliable internal component temperatures by conduction from the base plate only (i.e., thermal energy dissipation by conduction to the contact area of the mounting plate).

Unless otherwise defined in the relevant equipment specification, thermal dissipation through a component footprint shall not exceed a steady state maximum power dissipation of 775 W/m^2 (0.5 W/in^2) averaged over the entire component footprint, and the power dissipation at localized areas shall be limited to 2,325 W/m^2(1.5 W/in^2) over local areas of 1 square inch or less.

Equivalent transient heat dissipation of box is defined by orbit average + (peak value - average) $\times$ duty cycle.

Table 3-1. Equipment Temperature Limits

	EQUIPMENT	TEMPERATURE LIMITS			Temp limits definition
		Non-operating (T_N)	Start-up (T_S)	Operating (T_O)	
PAYLOAD	LUTI	-15 / +55	-10	-5 / +40	S/C I/F
	DTNPL	-35 / +70	-30	-20 / +50	Baseplate
	POLCAM OB	-30 / +65	-25	-20 / +60	S/C I/F
	POLCAM EB	-35 / +70	-30	-20 / +50	Baseplate
	KGRS SU	-35 / +65	-30	-20 / +50	S/C I/F
	KGRS EU	-35 / +70	-30	-20 / +55	Baseplate
	KMAG Boom/Hinge	-70 / +90	-65	-55 / +70	S/C I/F
	KMAG Actuator	N/A / +50	N/A	N/A / +50	TBD
	KMAG FCE	-35 / +70	-30	-20 / +50	Baseplate
	ShadowCam	-40 / +70	-40 / +40	-40 / +40	ShadowCam I/F
PDTS	RF Switch	-35 / +60	-25	-20 / +50	Baseplate
	X-band Transmitter	-40 / +75	-30	-20 / +55	Baseplate
	APMU	-80 / +150	-75	-30 / +70	S/C I/F
	APEU	-35 / +65	-30	-20 / +55	Baseplate

Table 3-1 (Cont'd). Equipment Temperature Limits

	EQUIPMENT	TEMPERATURE LIMITS			Temp limits definition
		Non-operating (T_N)	Start-up (T_S)	Operating (T_O)	
AOCS	RWA	-40 / +75	-20	-10 / +55	Baseplate
	CSSA	-190/+145	N/A	-150/+135	CSSA I/F
	GRA	-30 / +70	-34	-10 / +60	Baseplate
	STA-EU	-40 / +70	-35	-25 / +60	Baseplate
	STA-OH	-40 / +70	-35	-20 / +50	Foot
	SADM	-40 / +90	-30	-20 / +55	S/C I/F
	SADE	-40 / +65	-30	-15 / +55	Baseplate
TC & R	SBMU	-40 / +75	-30	-25 / +60	Baseplate
	S-band Transponder	-35 / +75	-30	-10 / +55	Baseplate
	S-Band Omni Antenna	-170 / +144	N/A	-170 / +134	Antenna I/F
	RFDU	-30 / +70	-25	-20 / +55	Baseplate
EPS	PCDU	-40 / +75	-40	-25 / +55	Baseplate
	Battery	+0 / +40	N/A	+10 / +30	Baseplate
	Solar Array	-190 / +145	N/A	-190 / +135	
PS	Propellant Tank	+3 / +49	+10	+10 / +47	
	Pressurant Tank	-40 / +60	-10	-10 / +50	Baseplate
	PCA	-30 / +55	-5	-5 / +45	
	Fill and Drain Valve	+3(A) / +70	+5	+5 / +49	
	Latching Isolation Valve	+3(A) / +70	+5	+5 / +49	
	Filter	+3(A) / +94	+5	+5 / +49	
	Pressure Transducer	+3(A) / +60	+5	+5 / +49	
	Cat-Bed Heater	+3(A) / +927	+5	+5 / +927	
	Thruster Valve	+3(A) / +121	+5	+5 / +116	
	Lines, Fitting, MTG HW	+3(A) / +94	+5	+5 / +49	

(A) Lower Temperature limit = -23°C when there is no propellant

3.1.4 Random Vibration, Acoustic, and Shock Requirements

3.1.4.1 Random and Sinusoidal Vibration

The payload and spacecraft components limit level vibration environment is specified in the following Tables 3-2, 3-3, 3-4, 3-5, 3-6, 3-7 and Figures 3-2, 3-3, 3-4, 3-5, 3-6.

a) Equipment except for those specified hereafter

Table 3-2. Random Vibration Level for Spacecraft Equipment and Propulsion Module Component (Except CSSA, S-Band antenna, SADM, APMU and APEU) [TBC]

i) Perpendicular to equipment interface plane (equipment longitudinal axis)

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
10 – 60	+9 dB/oct	+9 dB/oct
60 – 400	0.5	0.22
400 – 2000	-6 dB/oct	-6 dB/oct
Overall	18.4 grms	12.3 grms

ii) In equipment interface plane (equipment lateral axis)

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
20	0.026	0.013
50	0.16	0.08
800	0.16	0.08
2000	0.026	0.013
Overall	14.1 grms	10.00 grms

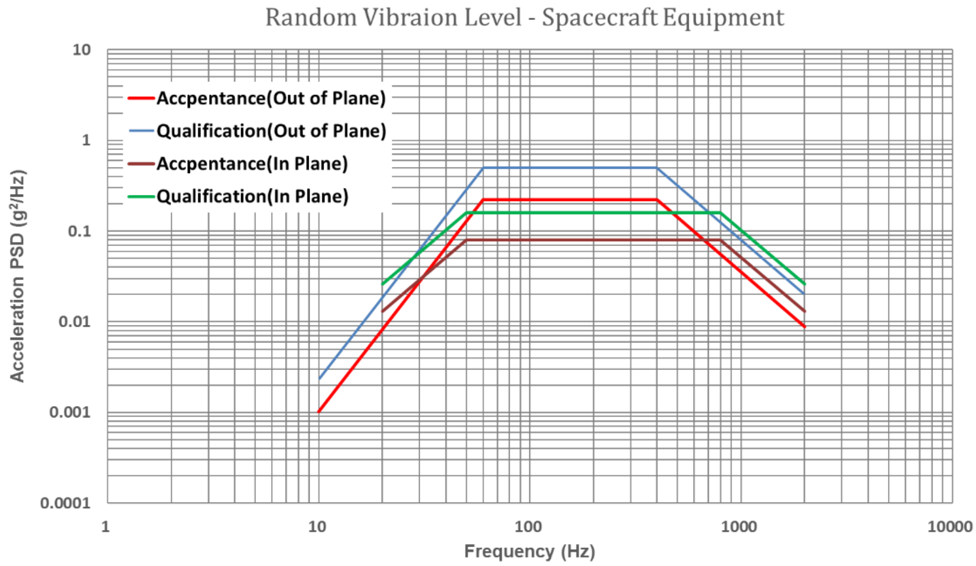


Figure 3-2. Random Vibration Level for Spacecraft Equipment and Propulsion Module Component (Except CSSA, S-Band antenna, SADM, APMU and APEU)

b) Coarse Sun Sensor Assembly (CSSA)

Qualification spectrum can be achieved in several parts if deemed necessary due to shaker limitations.

Table 3-3. Random Vibration Level for CSSA [TBC]

Frequency (Hz)	Frequencies [Hz]	Level
Qualification /Proto-flight	20	0.3
	70	12.5
	90	12.5
	100	10
	200	10
	300	1.875
	500	0.75
	2000	0.06
	Overall level	50.88 grms
Acceptance	20	0.15
	70	6.25
	90	6.25
	100	5
	200	5
	300	0.9375
	500	0.375
	2000	0.03
	Overall level	35.96 grms

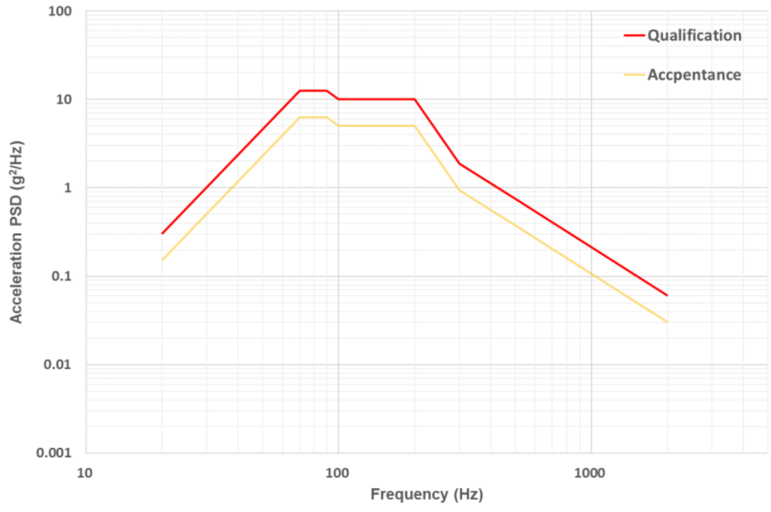


Figure 3-3. Random Vibration Level for CSSA [TBC]

c) S-Band antenna and equipment installed on brackets (SADM,…) in all three axes

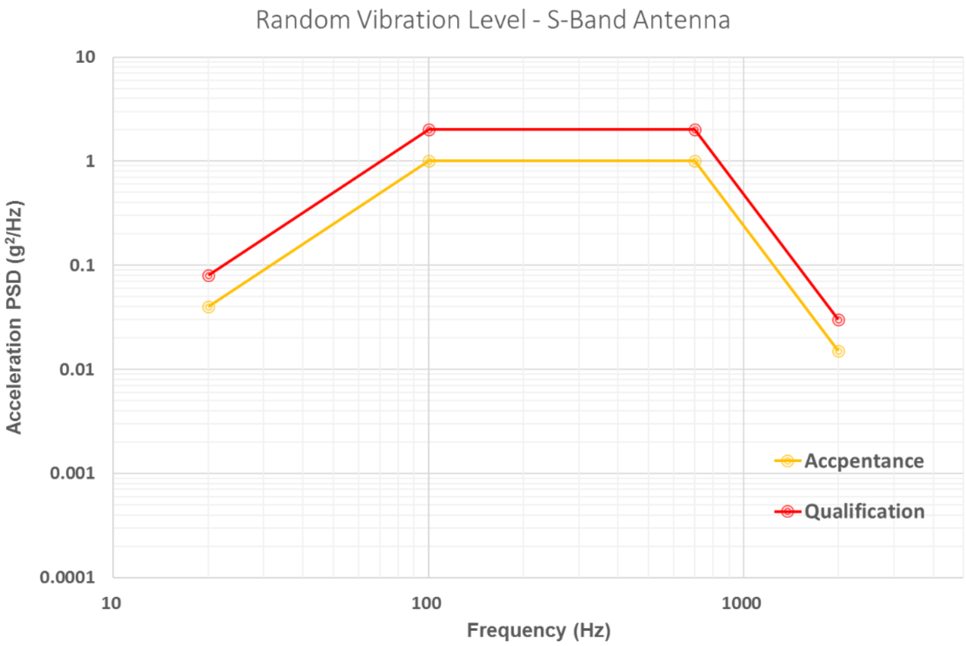
Table 3-4. Random Vibration Level for S-Band antenna and SADM

i) S-Band Antenna

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
20 – 100	0.04	0.08
100 – 700	1.0	2.0
700 – 2000	0.015	0.03
Overall	29.3 grms	41.4 grms

ii) SADM

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
10 – 60	+9 dB/oct	+9 dB/oct
60 – 400	0.5	0.22
400 – 2000	-6 dB/oct	-6 dB/oct
Overall	18.4 grms	12.3 grms



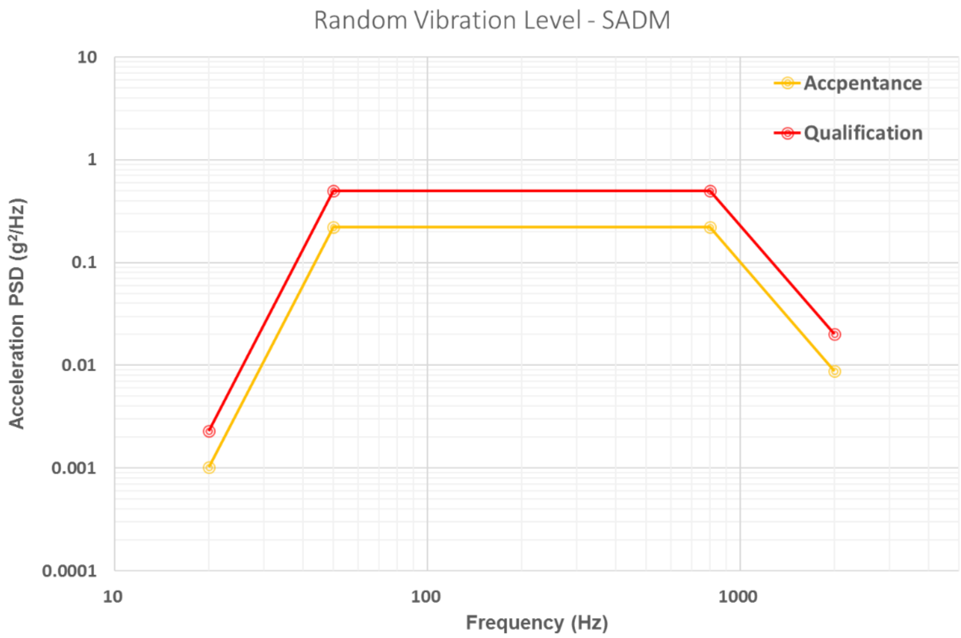


Figure 3-4. Random Vibration Level for S-Band antenna and SADM

d) APMU(Antenna Pointing Mechanism Unit), APEU(Antenna Pointing Electronics unit)

Table 3-5. Random Vibration Level for APMU and APEU [TBC]

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
20	0.026	0.013
50	0.16	0.08
800	0.16	0.08
2000	0.026	0.013
Overall	14.1 grms	10.00 grms

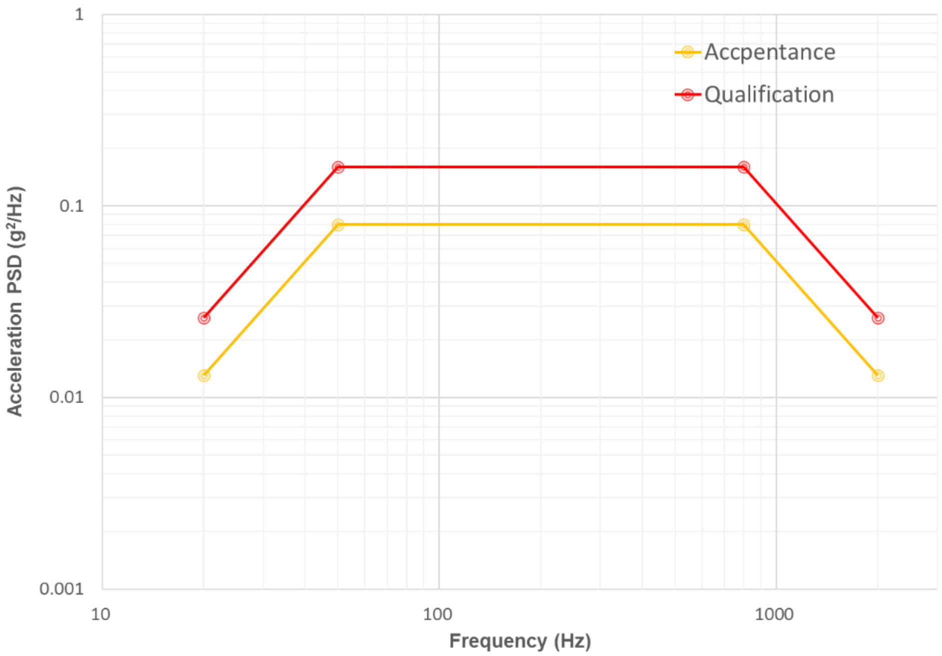


Figure 3-5. Random Vibration Level for APMU and APEU [TBC]

e) Payload

Table 3-6. Random Vibration Level for Payloads

i) Perpendicular to equipment interface plane (equipment longitudinal axis)

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
10 – 60	+9 dB/oct	+9 dB/oct
60 – 400	0.5	0.22
400 – 2000	-6 dB/oct	-6 dB/oct
Overall	18.4 grms	12.3grms

ii) In equipment interface plane (equipment lateral axis)

Frequency (Hz)	Qualification/Proto-flight PSD (g^2/Hz)	Acceptance PSD (g^2/Hz)
20	0.026	0.013
50	0.16	0.08
800	0.16	0.08
2000	0.026	0.013
Overall	14.1 grms	10.00 grms

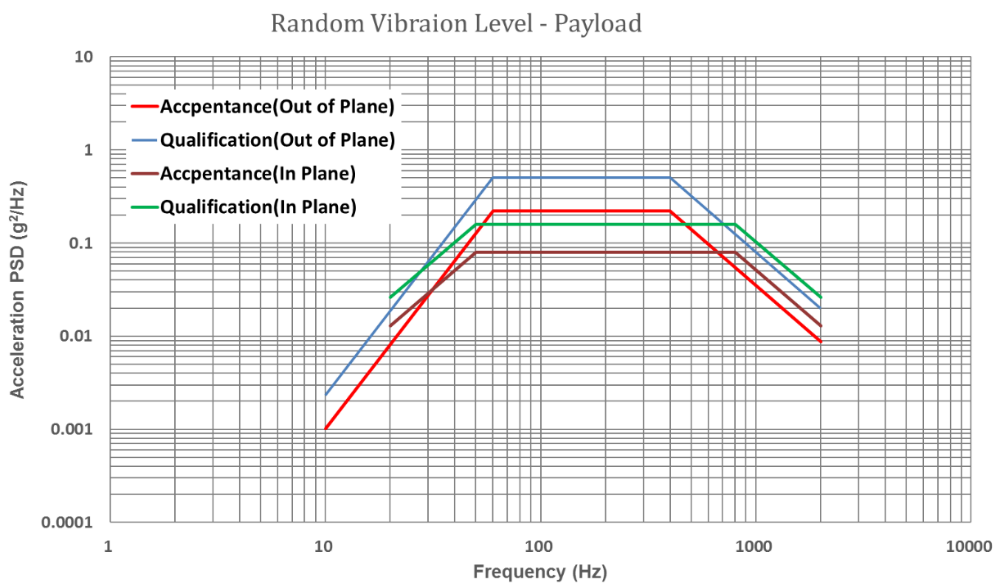


Figure 3-6. Random Vibration Level for Payloads

Table 3-7. Sinusoidal Vibration Level for Spacecraft Component and Payloads

Frequency [Hz]	Acceleration [g] Acceptance Level	Acceleration [g] Protoflight / Qualification Level
5-20	Max. shaker limit	Max. shaker limit
20-100	16 g	20 g
Sweep Rate	4 Oct/Min/Axis	4 Oct/Min/Axis 2 Oct/Min/Axis

The random vibration design margin and test range requirements for payload and spacecraft components are summarized in Table 3-8.

Table 3-8. Component Random Vibration Design and Test Requirements [TBR]

Component Status	Design Margin	Test Range
Payload		
LUTI and other payloads	Qualification level	Qualification level
Other Components	Qualification level	Qualification level
Spacecraft		
Existing Design	Qualification level	Limit levels
Modified Design	Qualification level	Qualification level
New Design	6 dB min. above limit levels	Qualification level

The spacecraft existing components shall be designed to Qualification level and tested to the limit level random vibration environment as specified in Figure 3-2 and Table 3-2. The spacecraft modified components shall be designed to Qualification level and tested to Qualification level those specified in Figure 3-2 and Table 3-2. All new spacecraft components shall be designed to levels 6 dB higher than and tested to Qualification level those specified in Figure 3-2 and Table 3-2.

The spacecraft CSSA, S-Band antenna, SADM, APMU and APEU shall be designed to a Qualification level and tested to the limit level (Acceptance level) random vibration environment as specified in Figures 3-3, 3-4, 3-5 and Table 3-3, 3-4, 3-5.

3.1.4.2 Acoustics

Any appendage such as reflectors and solar panels limit level acoustic environment will be as specified in Figure 3-7 and Table 3-9. Any appendage such as reflectors and solar panel shall be designed to withstand levels 6 dB higher than those specified in Figure 3-7 and Table 3-9.

Table 3-9. Acoustic Environment (TBR)

Frequency (Hz)	SPL (dB)
31.5	128.8
63	134.0
125	136.7
250	138.0
500	135.0
1000	132.8
2000	126.0
4000	123.0
8000	120.0
OVSPL	143.0

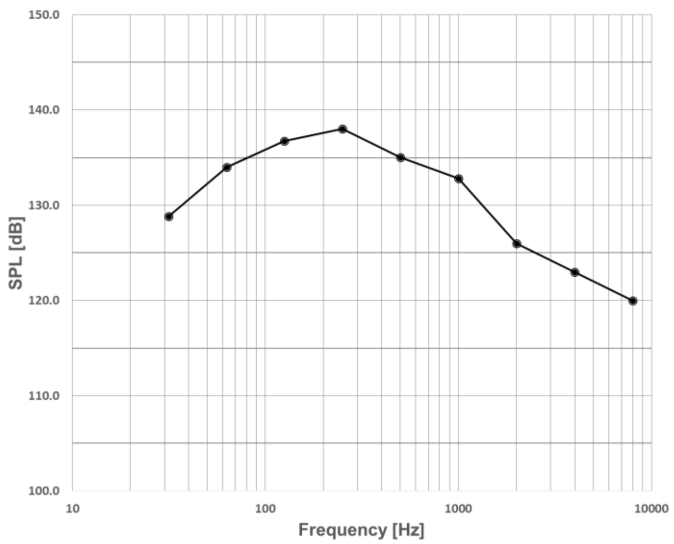


Figure 3-7. Acoustic Environment

3.1.4.3 Shock

The maximum expected (limit level) shock environment at the KPLO and LV interface is defined in the table below. KPLO and LV interface shock testing will be performed on the KPLO

Table 3-10. Spacecraft Separation and Antenna deployment Shock Level

i) Shock Response Spectrum (Q=10) [TBR]

	In plane		Out of plane	
Description	Freq. (Hz)	SRS (g)	Freq. (Hz)	SRS (g)
L/V Interface	100	20	100	20
	1000	2000	1000	2000
	5000	2000	5000	2000
Bottom Floor (a)	100	10	100	10
	400	300	400	400
	1000	700	1000	900
	2000	2000	1500	1300
	10000	2000	10000	1300
+/- Y, +/- Z Lower Closure Wall (b)	100	40	200	30
	1000	800	1000	900
	3000	800	2000	1500
	10000	800	10000	1500
+/- Y, +/- Z Upper Closure Wall (c)	100	10	100	10
	500	40	500	100
	1300	100	1000	300
	2000	700	3000	700
	10000	700	10000	700
+/- Y Lower Shear Wall (d)	200	10	200	10
	500	100	500	100
	1000	600	1000	600
	2000	1100	2000	1100
	10000	1100	10000	1100
+/- Y Upper Shear Wall (e)	100	10	100	10
	500	40	500	100
	1300	100	1000	300
	2000	700	3000	700
	10000	700	10000	700

Antenna Deployment Shock at S/C I/F (f)	100	10	100	10
	1000	140	1000	140
	2000	1000	2000	1000
	10000	1000	10000	1000

ii) SRS Level Applicable to S/C Payload and Bus Unit [TBR]

	EQUIPMENTS	Shock from S/C Separation	Shock from Antenna Deployment
P/L	LUTI	Table 3-10. i) (c)	N/A
	DTNPL	Table 3-10. i) (c)	N/A
	POLCAM OB	Table 3-10. i) (c)	N/A
	POLCAM EB	Table 3-10. i) (c)	N/A
	KGRS SU	Table 3-10. i) (c)	N/A
	KGRS EU	Table 3-10. i) (c)	N/A
	KMAG Boom/Hinge	Table 3-10. i) (c)	N/A
	KMAG Actuator	Table 3-10. i) (c)	N/A
	KMAG FCE	Table 3-10. i) (c)	N/A
	ShadowCam	Table 3-10. i) (c)	N/A
PDTS	RF Switch	Table 3-10. i) (c)	Table 3-10. i) (f)
	X-band Transmitter	Table 3-10. i) (c)	Table 3-10. i) (f)
	HGAA	Table 3-10. i) (c)	Table 3-10. i) (f)
AOCS	RWA	Table 3-10. i) (d)	N/A
	GRA	Table 3-10. i) (e)	N/A
	STA-EU	Table 3-10. i) (c)	N/A
	STA-OH	Table 3-10. i) (c)	N/A
	SADA	Table 3-10. i) (c)	N/A
	SADE	Table 3-10. i) (c)	N/A
TC&R	SBMU	Table 3-10. i) (c)	N/A
	S-band Transponder	Table 3-10. i) (e)	N/A
	S-Band Omni Antenna	Table 3-10. i) (c)	N/A
	RFDU	Table 3-10. i) (e)	N/A
EPS	PCDU	Table 3-10. i) (b)	N/A

	Battery	Table 3-10. i) (b)	N/A

The limit level shock environment at each platform of the spacecraft will be confirmed after the shock test of structure development model of KPLO

The shock design margin requirements for payload and spacecraft components are summarized in Table 3-11.

Table 3-11. Component Shock Design Margins

Component Status	Design Margin
Payload LUTI and other payloads Other Components	All payload units shall be designed to operate within performance spec. after being subject to the shock environment.
Spacecraft Existing Design Modified Design New Design	 Limit levels 3 dB minimum above limit levels 6 dB minimum above limit levels

The spacecraft existing components shall be designed to the limit level shock environment which will be updated after the detailed design of spacecraft structure. The spacecraft modified components shall be designed to a minimum of 3 dB higher than the limit level shock environment. All new spacecraft components shall be designed to levels 6 dB higher than the limit level shock environment.

3.1.5 External Pressure Extremes

All components shall be designed to meet their performance requirements after exposure to air transport ascent and descent pressure environments, and after exposure to launch vehicle boost and on-orbit pressure environments.

For the design of the equipments, the following assumptions have to be used :

- Maximum pressure decay rate of 70 mbars / sec.
- Pressure decay of 0.8 bar in 27 sec

(these values are more stringent than the envelope of the worst launcher conditions : 20 mbar/sec over 20 sec; peak at 69 mbar/sec of very short duration on some launchers; 0.8 bar in 75 sec min)

A demonstration by test (see section 4.10) is required. Alternately, a venting analysis showing a margin factor of 1.5 with the here above hypotheses, could be accepted after KARI approval.

For on-orbit operation, all equipment shall be designed to meet their performance requirements when exposed to the vacuum that is commensurate with the orbital altitude, as low as 10^{-10} Torr.

3.1.6 Transportation, Handling and Storage

Equipment shall be protected during fabrication, assembly, handling, transportation, storage and other pre-launch activities such that none of the acceptance test environments defined in Section 4.0 are exceeded. Environmental controls shall be employed when required to ensure no subsequent performance degradation. The shipping containers will be instrumented to record temperature, humidity shock and accelerations as required. Unit supplier shall have a responsibility of damage in transportation.

3.1.7 Component Life

Life is defined as the period of time/cycles during which the end item will not wear out or degrade such that it can't meet its specified requirements. All components shall be capable of supporting thirteen months minimum of orbital operation and four years of ground storage including AIT, for a total useful life of five years one month. The limited life items are hardware subject to degradation due to age, operating time, or cycles, and thus require periodic replacement, refurbishment, retest, or operation to assure that its mission performance will not be degraded beyond acceptable limits. These limited life items shall require time/cycle tracking during fabrication, test, storage, and mission operation.

3.1.7.1 Operating Life

The design shall accommodate all identifiable wear-out and time-dependent degradation factors so that specified performance can be maintained for a minimum of a 13-months mission life (after launch). The design shall also consider equipment operating times that occur prior to initial on-orbit operation, such as post-delivery integration tests and launch operations.

3.1.8 Natural and Induced Environments

All equipment shall be designed to perform in accordance with the appropriate equipment specification during and after exposure to any single or reasonable combination of the natural and induced environments representing the worst case conditions of the launch vehicle (i.e., acoustic, vibration, temperature, vacuum, etc.). The natural and induced environments are indicative of the test, transportation, handling, ground storage, pre-launch, launch/ascent, and operating orbital life.

3.1.8.1 Corona

SADA, RF power units and units powered during launch will be submitted to all pressure values between 1 bar and 0 bar. They have to be designed to operate in such an environment without any arcing. (see section 4.8)

3.1.8.2 Aero-Thermal Flux

Externally mounted units, shall be able to withstand a nominal aero-thermal flux of 1135 W/m2. In addition, for some particular launch configuration, they shall be able to withstand 1500 W/m^2 during 70 sec.

3.1.8.3 Micro-vibration

Sinusoidal vibration can be induced by the operation of the spacecraft actuators. The in-orbit flight excitations induced by all units at baseplate level, shall be equal or lower than the following spectrum:

Frequency (Hz)	Acceleration (g)
5 ~ 200	10^{-2}
200 ~ 450	Decrease based on $1/f^2$
450 ~ 2000	2×10^{-3}

The method used to compute or measure the micro-vibrations generated at equipment level shall be approved by KARI.

All equipments shall meet their performance requirements when submitted to this environment. (qualification levels are 1.5 times the above level)

3.1.8.4 Vacuum

Exposure to a hard vacuum commensurate with orbital altitude will occur at synchronous orbit, this will be 10^{-10} Torr in free space, but may be as high as 0.1 Torr within the spacecraft interior. Equipments shall be designed to operate in these environments, in particular without any arcing (see section 4.8)

3.1.9 Orbital Radiation Dose Levels

Any spacecraft components susceptible to damage from radiation shall be designed and constructed to operate within the levels shown in the paragraph 3.1.9.1 and 3.1.9.2 with sufficient margin defined in the PAR. Mission radiation environments of KPLO are derived from 1 year mission period of lunar orbit.

For the preliminary analysis, it shall be assumed that the spacecraft structure and its major components offer a shielding of : 0.6mm omni-directional for an equipment mounted inside the spacecraft; if an equipment mounted outside the spacecraft, 0.6mm on the baseplate direction with no shielding on the other directions shall be assumed.

3.1.9.1 Total Ionizing Dose

Figure 3-8 and Table 3-12 provides a mission ionizing dose depth curve calculated from SHIELDOSE model.

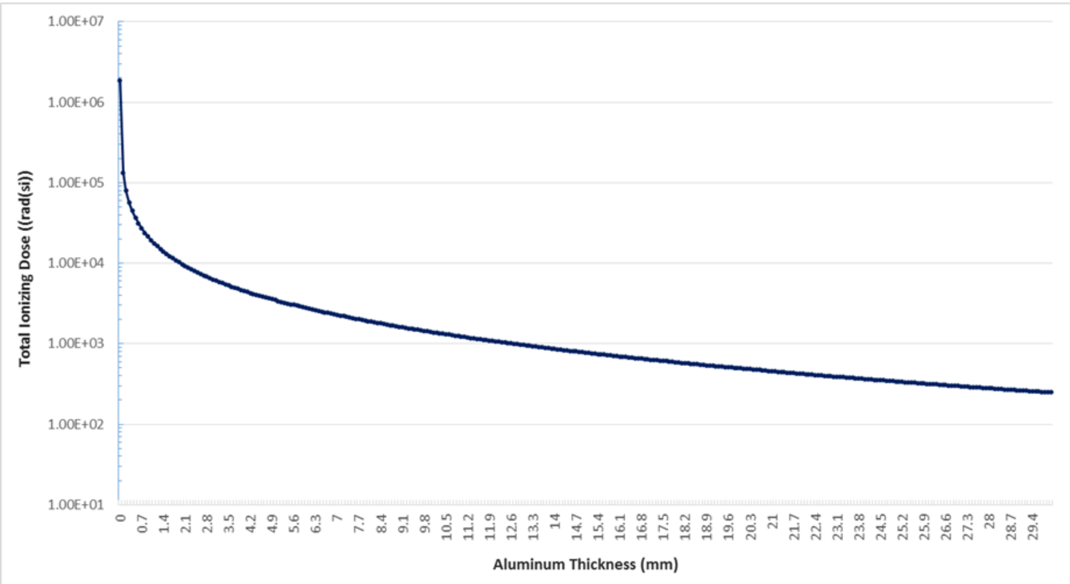


Figure 3-8. Mission Radiation Dose Depth Curve (for TID)

Table 3-12. Radiation Dose Depth values corresponding to Figure 3-7

Al Shielding Thickness (mm)	TID (rad(Si))	Al Shielding Thickness (mm)	TID (rad(Si))
0.5	3.68E+04	16	7.03E+02
1	1.95E+04	16.5	6.71E+02
1.5	1.31E+04	17	6.40E+02
2	9.77E+03	17.5	6.12E+02
2.5	7.68E+03	18	5.84E+02
3	6.29E+03	18.5	5.60E+02
3.5	5.30E+03	19	5.36E+02
4	4.56E+03	19.5	5.16E+02
4.5	3.96E+03	20	4.96E+02
5	3.49E+03	20	4.96E+02
5.5	3.10E+03	20.5	4.76E+02
6	2.78E+03	21	4.56E+02
6.5	2.51E+03	21.5	4.41E+02
7	2.28E+03	22	4.24E+02
7.5	2.08E+03	22.5	4.07E+02
8	1.91E+03	23	3.93E+02
8.5	1.76E+03	23.5	3.79E+02
9	1.63E+03	24	3.66E+02
9.5	1.51E+03	24.5	3.53E+02
10	1.41E+03	25	3.41E+02
10.5	1.31E+03	25.5	3.30E+02
11	1.23E+03	26	3.19E+02
11.5	1.15E+03	26.5	3.08E+02
12	1.08E+03	27	2.99E+02
12.5	1.02E+03	27.5	2.89E+02
13	9.65E+02	28	2.80E+02
13.5	9.10E+02	28.5	2.71E+02
14	8.62E+02	29	2.63E+02
14.5	8.18E+02	29.5	2.55E+02
15	7.78E+02	30	2.48E+02
15.5	7.39E+02		

3.1.9.2 Non-ionizing Dose (Displacement Damage Dose)

Long-term damage due to atomic displacements (displacement damage) degrades solar cells, optoelectronics, imaging devices such as CCDs, and some bipolar technologies. The displacement damage is caused by exposure of the components to protons and electrons. The threat ranges from moderate to severe depending on the amount of shielding surrounding the components. Solar cells and sensors that are fully exposed to the space environment are especially susceptible. No effect due to TNID (Total Non-ionizing Dose) shall cause permanent damage to a system or subsystem, or degrade its performances outside its specification limits.

Figure 3-9 and Table 3-13 show Displacement Damage Dose curve which is calculated for aluminum sphere shielding of various thickness. Damage equivalent 10 MeV proton fluxes as a function of spherical Aluminum shield radius is presented on Figure 3-10 and tabulated on Table 3-14 for Silicon target. Table 3-15 shows NIEL rates for proton in silicon.

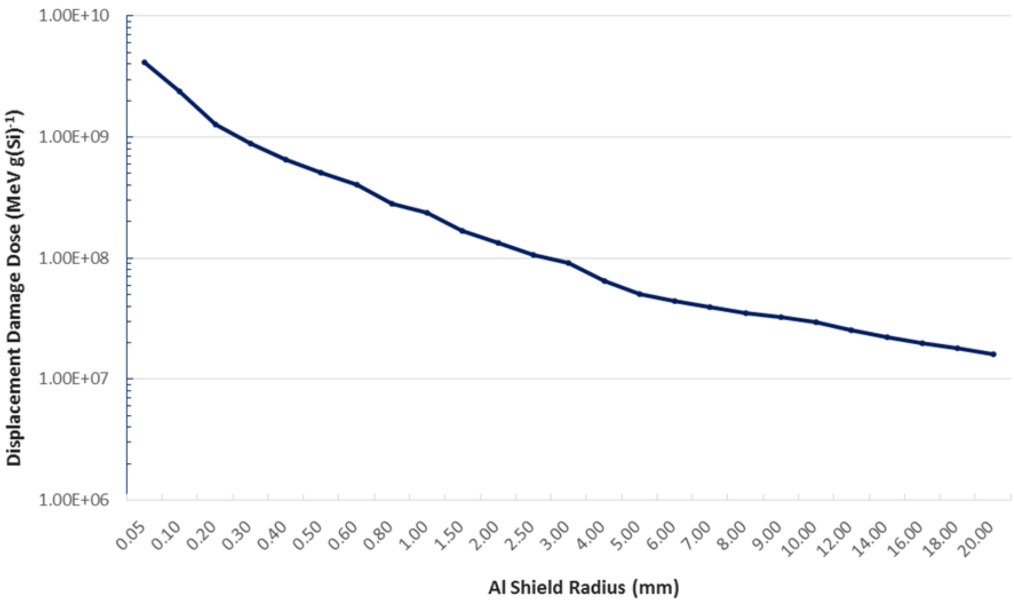


Figure 3-9. Displacement Damage Dose as a Function of Spherical Al Shield Radius, for Silicon Target

Table 3-13. Displacement Damage Dose values corresponding to Figure 3-9

Al Shielding Thickness (mm)	TNID (MeV/g(Si))	Al Shielding Thickness (mm)	TNID (MeV/g(Si))
0.05	4.15E+09	4.00	6.47E+07
0.10	2.38E+09	5.00	5.09E+07
0.20	1.27E+09	6.00	4.41E+07
0.30	8.77E+08	7.00	3.94E+07
0.40	6.56E+08	8.00	3.51E+07
0.50	5.11E+08	9.00	3.23E+07
0.60	4.06E+08	10.00	2.98E+07
0.80	2.80E+08	12.00	2.54E+07
1.00	2.35E+08	14.00	2.24E+07
1.50	1.67E+08	16.00	1.98E+07
2.00	1.33E+08	18.00	1.79E+07
2.50	1.07E+08	20.00	1.61E+07
3.00	9.05E+07		

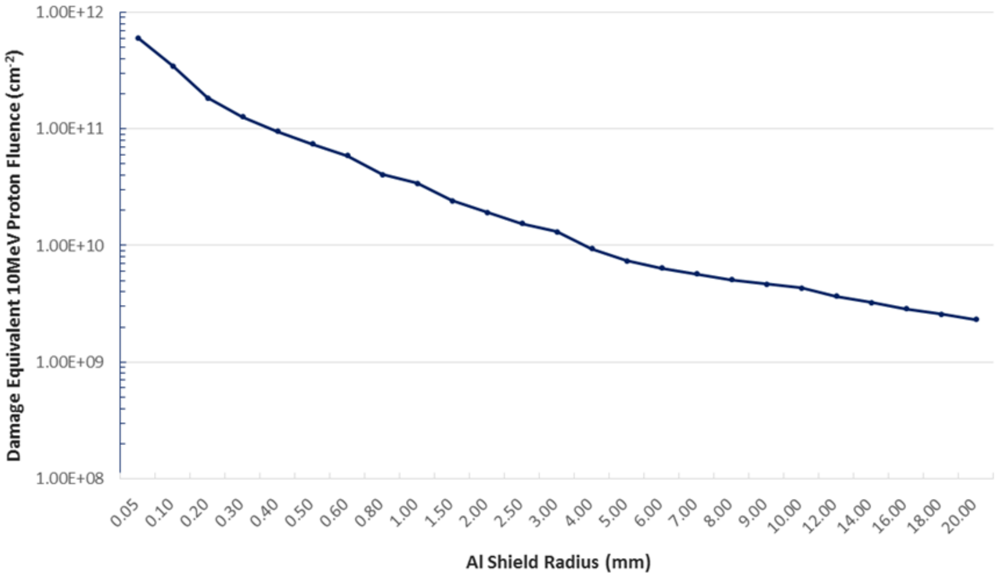


Figure 3-10. Damage Equivalent 10MeV Proton Fluence as a Function of Spherical Al Shield Radius, for Silicon Target

Table 3-14. DDEF values corresponding to Figure 3-10

Al Shielding Thickness (mm)	DDEF (/cm^2) (10MeV protons)	Al Shielding Thickness (mm)	DDEF (/cm^2) (10MeV protons)
0.05	6.02E+11	4.00	9.37E+09
0.10	3.46E+11	5.00	7.38E+09
0.20	1.84E+11	6.00	6.39E+09
0.30	1.27E+11	7.00	5.71E+09
0.40	9.51E+10	8.00	5.08E+09
0.50	7.40E+10	9.00	4.68E+09
0.60	5.89E+10	10.00	4.31E+09
0.80	4.06E+10	12.00	3.68E+09
1.00	3.41E+10	14.00	3.25E+09
1.50	2.42E+10	16.00	2.87E+09
2.00	1.93E+10	18.00	2.59E+09
2.50	1.55E+10	20.00	2.33E+09
3.00	1.31E+10		

Table 3-15. NIEL rates for Protons in Si

Energy (MeV)	NIEL (MeV cm^{-2} g^{-1})	Energy (MeV)	NIEL (MeV cm^{-2} g^{-1})
0.01	2.00E+00	7	9.80E-03
0.02	1.30E+00	10	6.90E-03
0.03	1.00E+00	15	5.40E-03
0.05	6.90E-01	20	4.70E-03
0.07	5.40E-01	30	4.30E-03
0.1	4.10E-01	50	3.70E-03
0.2	2.40E-01	70	3.30E-03
0.3	1.70E-01	100	3.00E-03
0.5	1.10E-01	150	2.50E-03
0.7	8.50E-02	200	2.40E-03
1	6.30E-02	300	2.20E-03
2	3.20E-02	400	2.10E-03
3	2.20E-02	500	2.00E-03
5	1.40E-02	1000	1.70E-03

DDEF (Displacement Damage Equivalent Flux) for other particles and materials and NIEL (Non-Ionizing Energy Loss) table can be used for calculation with KARI review.

3.1.10 Orbital Radiation for Single Event Effects (SEE)

Galactic Cosmic Ray (GCR) and solar flare radiation result in charge deposition inside EEE components, which can induce different kind of Single Event Effects (SEE), such as Single Event Upset (SEU), Multiple Bit Upset (MBU), Single Event Latch-up (SEL), Single Event Transient (SET), Single Event Burn-out (SEB), Single Event Gate Rupture (SEGR), Single Event Dielectric Rupture (SEDR), Single Event Functional Interrupt (SEFI) and others.

Subsystem and equipment shall be designed to ensure that nominal performance is maintained in the presence of SEE (non-destructive SEE). No destructive SEE shall cause damage to equipment and induce performance degradations or outages.

The equipment supplier shall evaluate the SEE occurrence probability and effects. SEE rate calculations and SEE characterizations are based on Linear Energy Transfer (LET) integral spectra for Galactic Cosmic Ray (GCR) and anomalous large Solar Particle Event (SPE) conditions. Applicable integral spectra for both conditions are shown on Figure 3.11 and Table 3.16.

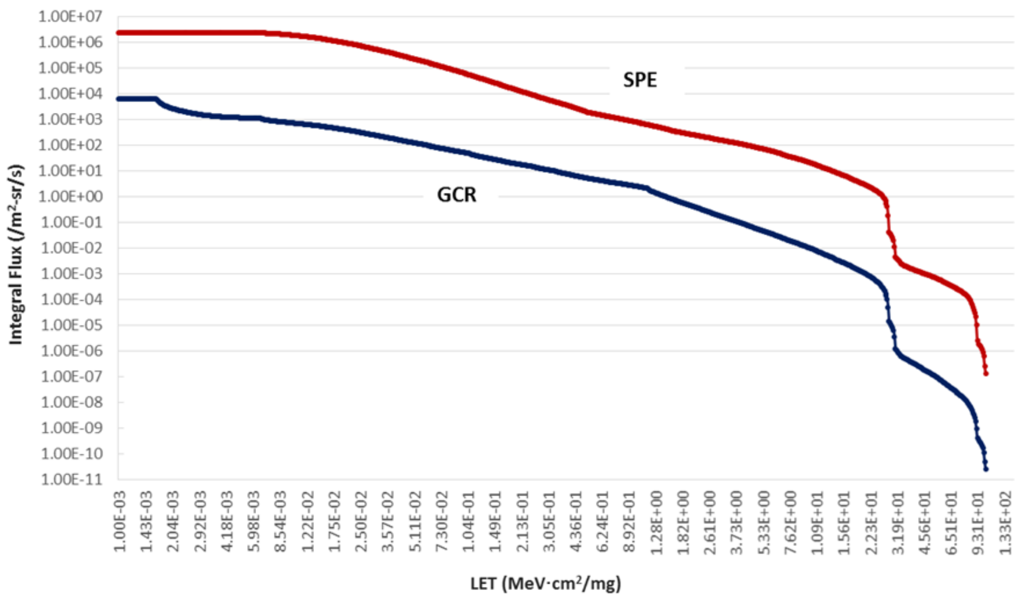


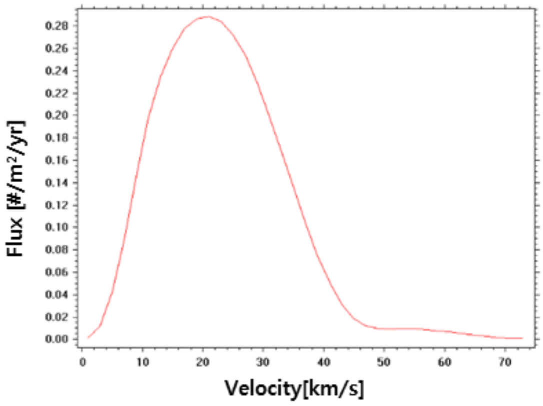
Figure 3-11. Integral Flux Spectrum for GCR and SPE (behind 100mils of Al Shielding)

Table 3-16. Integral Flux values corresponding to Figure 3-11

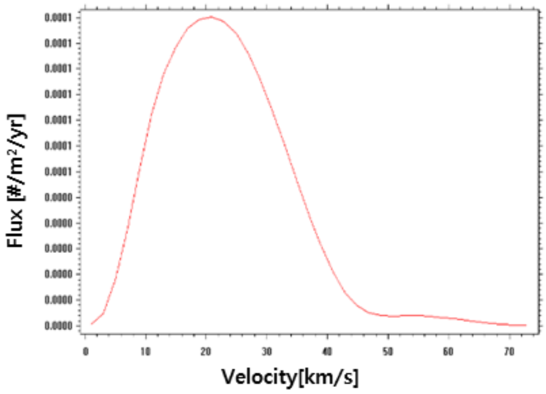
LET (MeV·cm^2/mg)	Integral Flux (#/m^2-sr/s)		LET (MeV·cm^2/mg)	Integral Flux (#/m^2-sr/s)	
	GCR (M=3)	SPE (M=8)		GCR (M=3)	SPE (M=8)
1.00E-03	6.26E+03	2.39E+06	2.42E+01	5.15E-04	1.62E+00
3.03E-03	1.49E+03	2.39E+06	2.60E+01	3.04E-04	1.08E+00
5.00E-03	1.15E+03	2.36E+06	2.63E+01	2.76E-04	9.82E-01
5.06E-03	1.15E+03	2.36E+06	2.83E+01	5.03E-05	1.79E-01
7.06E-03	9.61E+02	2.21E+06	3.00E+01	8.36E-06	2.53E-02
9.07E-03	7.73E+02	1.97E+06	3.22E+01	9.11E-07	3.61E-03
1.01E-02	7.19E+02	1.85E+06	3.42E+01	5.98E-07	2.30E-03
3.02E-02	2.43E+02	5.35E+05	3.63E+01	4.65E-07	1.90E-03
5.05E-02	1.24E+02	2.25E+05	3.81E+01	3.85E-07	1.63E-03
7.04E-02	7.89E+01	1.22E+05	4.00E+01	3.22E-07	1.42E-03
9.05E-02	5.74E+01	7.47E+04	4.24E+01	2.54E-07	1.22E-03
1.01E-01	5.08E+01	6.03E+04	4.40E+01	2.23E-07	1.13E-03
3.02E-01	1.11E+01	5.89E+03	4.61E+01	1.87E-07	1.01E-03
5.03E-01	5.29E+00	2.00E+03	4.83E+01	1.57E-07	9.06E-04
7.03E-01	3.62E+00	1.23E+03	5.01E+01	1.35E-07	8.29E-04
9.03E-01	2.77E+00	8.78E+02	5.32E+01	1.06E-07	7.07E-04
1.00E+00	2.46E+00	7.53E+02	5.58E+01	8.34E-08	6.08E-04
2.01E+00	4.64E-01	2.67E+02	5.78E+01	7.01E-08	5.38E-04
3.01E+00	1.80E-01	1.58E+02	6.06E+01	5.44E-08	4.49E-04
4.00E+00	9.23E-02	1.09E+02	6.36E+01	4.14E-08	3.79E-04
5.02E+00	5.40E-02	7.82E+01	6.67E+01	3.33E-08	3.27E-04
6.01E+00	3.52E-02	5.80E+01	7.00E+01	2.66E-08	2.79E-04
7.01E+00	2.41E-02	4.32E+01	7.34E+01	2.07E-08	2.34E-04
8.09E+00	1.69E-02	3.21E+01	7.70E+01	1.57E-08	1.89E-04
1.00E+01	9.81E-03	2.03E+01	8.07E+01	1.12E-08	1.44E-04
1.20E+01	5.97E-03	1.32E+01	8.37E+01	7.88E-09	1.06E-04
1.40E+01	3.71E-03	8.74E+00	8.77E+01	3.91E-09	4.85E-05
1.60E+01	2.52E-03	6.23E+00	9.09E+01	1.89E-09	2.23E-05
1.80E+01	1.75E-03	4.47E+00	9.42E+01	3.52E-10	1.87E-06
2.00E+01	1.22E-03	3.25E+00	9.77E+01	2.51E-10	1.36E-06
2.20E+01	8.35E-04	2.36E+00	1.00E+02	1.70E-10	9.19E-07

3.1.11 Micrometeoroids

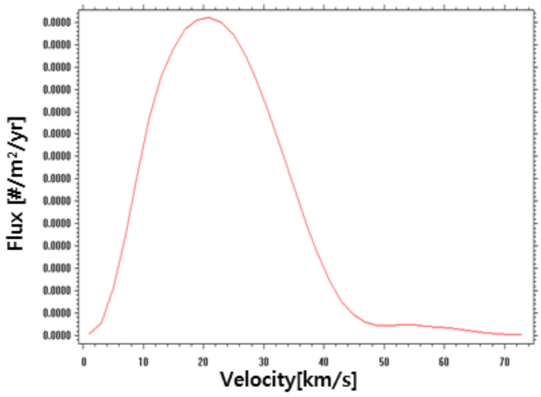
The spacecraft, or spacecraft components exposed to open space, shall survive the meteoroids environment for a period of at least thirteen (13) months, as shown in Figures 3-12 and listed in Tables 3-17.



[10^{-6}g Flux]



[10^{-3}g Flux]



[1g Flux]

Figure 3-12. Micrometeoroid Fluence

Table 3-17. Micrometeoroid Mass Fluence Spectrum

Mass (Diameter)*	+X Ram ($\#/m^2/yr$)	-X Wake ($\#/m^2/yr$)	+Y Port ($\#/m^2/yr$)	-Y Starboard ($\#/m^2/yr$)	+Z Zenith ($\#/m^2/yr$)	-Z Nadir ($\#/m^2/yr$)
> 10^{-6}g (0.12mm)	1.14E+00	8.00E-01	1.03E+00	9.69E-01	1.38E+00	1.42E-01
> 10^{-3}g (1.2mm)	4.75E-04	3.33E-04	4.28E-04	4.03E-04	5.74E-04	5.91E-05
> 1g (12mm)	5.58E-08	3.93E-08	5.07E-08	4.81E-08	6.75E-08	7.00E-09

3.2 General Test Requirements

Definitions of the various tests are described in section 1 and the descriptions of the corresponding test environments are given in section 4, along with test article and equipment criteria. This section defines the standard requirements for environmental testing to assure control for demonstrating hardware/software compliance to contractual requirements which require proto-flight qualification flight and acceptance testing.

Components subjected to the proto-flight and acceptance tests of section 4 shall be identical to the flight configuration equipment except for the differences required for test simulation, test external power requirements, and test instrumentation.

The test apparatus used in conducting test shall be capable of producing and maintaining the required test conditions with the equipment installed and operating as specified. Changes in test conditions from the nominal values specified by the applicable test procedures shall not exceed the tolerances specified in section 3.2.2.

3.2.1 Ambient Environment

Ambient conditions for conducting performance test prior to, during or after environmental exposures shall be within the range indicated below:

Temperature : (16 to 32)℃

Relative Humidity : (20 to 70)%

When tests are planned and conditions are outside these ranges, a go-no-go decision will be made by the subsystem responsible design engineer (RDE) or representative. At no time shall change ambient conditions within the range above, cause moisture condensation on a component.

3.2.2 Test Condition Tolerances

In the absence of a rationale for other test condition tolerances, the following values shall be used. These values include measurement uncertainties.

Acoustics
1/3 Octave Band:

Frequency (Hz)	Tolerance (dB)	
	Requirement	Reference (Goal)
f < 40	FC *	+3, -6
40 < f < 3150	±3	±3
f ≥ 3150	FC *	+3, -6
Overall Level	±1.5	±1

Note: * FC = Facility Capability : The reference frequency tolerances are outside of direct individual control and will be met closely as possible with available component.

Antenna Pattern Determination ± 2 dB

Electromagnetic Compatibility

Voltage Magnitude:	± 5 % of the peak value
Current Magnitude:	± 5 % of the peak value
RF Amplitudes:	± 2 dB
Frequency:	± 2 %

Distance: ± 5 % of specified distance, or ± 5 cm, whichever is greater

Humidity	± 5 % RH
Loads	
Steady-State (acceleration):	± 5 %
Magnetic Properties	
Mapping Distance Measurement	± 1 cm
Displacement of assembly center of Gravity (c.g.) from rotation axis:	± 5 cm
Vertical displacement of single probe Centerline from c.g. of assembly:	± 5 cm
Mapping turntable angular displacement:	± 3 degrees
Magnetic Field Strength:	± 1 nT
Repeatability of magnetic measurements (Short term):	± 5 % or +/- 2 nT, whichever is greater
Demagnetizing and Magnetizing Field Level:	± 5 % of nominal
Mass Properties	
Weight:	± 0.2 %
Center of Gravity:	± 0.15 cm (+/- 0.06 in.)
Moments of Inertia:	± 1.5 %
Mechanical Shock	
Response Spectrum:	+ 100 %, - 50 %
Time History:	± 10 %

Pressure

Greater than 1.3 x 104 Pa (Greater than 100 mm Hg):	± 5 %
1.3 x 104 to 1.3 x 102 Pa (100 mm Hg to 1 mm Hg):	± 10 %
1.3 x 102 to 1.3 x 101 Pa (1 mm Hg to 1 micron):	± 25 %
Less than 1.3 x 101 Pa (less than 1 micron):	± 80 %

Temperature	± 2.0℃
Temperature Stabilization	The component is within 2℃ of the specified temperature extreme and the rate of change of temperature is less than 1℃ per hour.

Vibration

Sinusoidal:	Amplitude	± 10 %
	Frequency	± 2 %
Random:	RMS level	± 10 %
	Accel. Spectral Density	± 3 dB

3.2.3 Vacuum Measurements

Vacuum measurements shall be made utilizing an instrument which is conductively connected to the same vacuum environment as the test article.

3.2.4 Temperature Measurements

Component test temperatures during thermal vacuum tests shall be controlled by a reference temperature sensor located on the component base plate (on the component side of the thermal interface) or on the component mounting panel. Additional sensors may be attached to external surfaces for monitoring and/or redundancy. Temperature reference point shall not be located in a high temperature or low temperature region of the base plate, but the area which will indicate the average temperature. Test sensors shall not generally be required internal to the component for test purposes. Sensors which are located internally, and are a normal part of the component functional/system design, shall be monitored throughout all test phases.

Temperature cycling tests at ambient pressure shall be controlled using appropriate controls of the temperature chamber in accordance with the test procedure. Equipment temperatures shall be monitored as in thermal vacuum testing. The combination of these environments shall be such to prohibit condensation on any test component. Temperature stabilization criteria shall be considered satisfied when requirements of section 3.2.2 are met.

3.2.5 Accuracy of Measurements

The accuracy of the measurement equipment shall be subject to controls and documentation which assures conformance of test requirements in this document. The calibration system utilized by any vendor shall provide for the prevention of inaccuracy by ready detection of deficiencies and timely positive action for their correction.

Each contractor shall be responsible for assuring that his subcontractors have calibration systems which meet the requirements of this specification.

3.2.5.1 Calibration System

Documentation shall exist which provides objective evidence of accuracy conformance. These shall include a description of the calibration system on equipment to be used; the measurement standards utilized; calibration intervals; sources; applicable calibration procedures; environmental conditions under which the measurement standards applied; and any transfer, nomenclature, identification numbers and vendor self audit practices utilized.

Standards established by the vendor for calibrating the test and measuring equipment used in controlling product quality shall cover the capabilities for accuracy, stability, and range required for the intended use. The test and measuring equipment shall be calibrated for the environments they are expected to experience (thermal, shock, vibration, pressure, acoustic, humidity, cleanliness, and solar).

Calibration periodic intervals shall have been established as required to assure continued accuracy on the basis of stability, purpose, and degree of usage. Written procedures shall exist for the calibration of all measurement and test equipment. The procedures may be a compilation of published standard practices or manufacturer's written instructions and need not be rewritten to satisfy the requirements of this specification.

3.2.5.2 Calibration Source

Measuring and test equipment shall be calibrated utilizing reference standards whose calibration is certified as being traceable to an internationally accepted Bureau of Standards, or accepted values of natural physical behavior for self-calibration technique. This certification shall be audible by a quality assurance representative of the program.

3.2.5.3 Calibration Labeling

Measuring and test equipment shall be labeled to indication the date of last calibration and when the next calibration is due. When physical size prevents such labeling, then an identifying code shall be used. If this is not practical, then suitable procedures can be used.

3.2.6 Test Procedures and Reports

Prior to being subjected to the tests specified in this document, test procedures shall be prepared for each component. Components shall be operated and tested in accordance with approved procedures as developed from the appropriate equipment specification. The test procedures and final test report shall specify the following information as a minimum:

a) Configuration and identification numbers of major equipment items to be tested; whether they are flight equipment or any differences from flight configurations.
b) Designation of reference axis, if required for test.
c) Safety precautions or warnings to personnel or hardware.
d) Specific test required.
e) Required recording and measuring equipment.
f) Interface diagram for all test instrumentation.
g) Detail covering test setup and sequence of operations for each test.
h) Measurements to be made before, during, and after each test.
i) Pass or fail criteria in terms of the performance measurements to be made, including the definition of allowable test measurement tolerances and

corresponding design specification parameter limits, from the applicable equipment specification.

j) Operating temperature ranges and sensor locations.
k) Provisions to verify satisfactory performance of redundant elements.
l) Specific environmental test conditions and instrumentation defined in detail (the values and tolerances of which shall comply with this specification) including test levels and duration.
m) Test sequence requirements.
n) Procedure to follow in event of test failure.
o) Methodology for rounding off data numbers
p) Procedure to follow for required retesting.
q) Personnel required at the test (checklist signatures).
r) Photographs or documenting sketches made of instrumentation after being attached.
s) Provisions for quality verification of test instrumentation (calibration), test set-up and performance measurements.
t) Comparison of actual recorded environment with the required environment.
u) Equipment operating time log.
v) Quality Assurance acceptance.

3.2.7 Functional Test

These tests shall be performed prior to and following environmental exposures. Functional tests also shall be conducted during thermal vacuum or thermal cycle testing. Power-on vibration monitoring requirements for equipment shall be in accordance with the appropriate EQ specification. It is permissible to combine tests involving the application of like environments so that both operative and non-operative requirements may be verified during a single test sequence.

The final fabrication functional test shall satisfy the requirements for a pre-environmental functional test if performed within 60 days of the environmental exposure (components only).

3.2.7.1 Monitor Mode (Not Applicable to Solar Array Panels)

Key component performance parameters (including, as a minimum, DC power line current and/or RF signal and significant output voltage/current levels) shall be monitored continuously throughout the tests. The monitoring can be accomplished using multi-channel strip chart recorders or the equivalent during all power-applied (DC or RF) environmental tests; or it can use the normal factory test set capabilities for automated test and printout.

3.2.8 Rejection and Retest

Equipment shall be designed to achieve its specified performance requirements during and subsequent to exposure to specified environments. If a failure, a malfunction, or out-of-tolerance condition occurs during or after a test, the test shall be discontinued, the deficiency (including any design defect) corrected, and pertinent environmental tests repeated until successfully complete. When changes affect previously completed tests in the sequence, such tests shall also be repeated. All failures and corrective actions shall be documented in accordance with KPLO PAR.

3.2.9 Test Data Package

Upon completion of each environmental test, a data package will be prepared which includes, but is not limited to, a copy of the test procedure, test data sheets, list of equipment used (from data sheets in the procedure), the amount of accumulated operating time for each test item, a description/photograph of the test item configuration, and tabulation of all failure reports/damage. Other pertinent raw data, including log sheets, graphs, records, photographs, or incidents, shall be recorded in such a way as to allow copies or originals to be made. Test data packages will include the following identification:

a) Test article part number
b) Retest requirements document
c) Q.A. acceptance tag
d) Events log
e) Operating time record
f) All test procedures including diagrams of the test setups identifying axes and diagrams of thermal and thermal vacuum setup.
g) Test data
h) Summary test report and conclusions
i) All discrepancy documents.

3.2.10 Vibration Test Requirements

The vibration test fixture shall be designed to support the equipment and its supporting structure. Transmissibility between test items mounting point shall not exceed a factor of 6 dB between 5 and 2000 Hz. Cross axis response shall not exceed the excitation direction input.

3.2.10.1 Vibration Fixture Evaluation

A fixture evaluation shall be performed prior to performance of the specified qualification vibration test. When a fixture is to be used for more than one type of equipment, the worst case (fixture and equipment) shall be evaluated. Successful completion of the evaluation shall be a requirement for qualification. The transmissibility and crosstalk of the test fixture, loaded by the test item or a dynamically similar dummy unit, shall be determined by performing a low level random or sinusoidal vibration evaluation sequence along each of three mutually perpendicular axes.

The random vibration evaluation method shall be performed utilizing a low level broadband (20 to 2000 Hz) random signal with the spectrum shape flat at 0.01 g^2/Hz (4.5 g rms overall level). The exposure duration shall be sufficient to obtain a sample for analysis (approximately 30 seconds). The sinusoidal vibration method shall be performed by sweeping a sinusoidal signal through the 5 to 2000 Hz frequency range at a rate of 2 octaves/minute and at a peak acceleration level of 1 g. The low frequency input may be limited to 0.5 inch double amplitude.

3.2.10.2 Vibration Shaker System

All shakers shall be equipped with over-acceleration, over-travel (at 20 percent above the specified test tolerance levels) and loss of signal (6 dB below input) devices for automatic system shutdown. Shutdown should follow a multi-cycle decay to avoid shock.

Proper operation of the equipment shall be certified before the start of each vibration test.

3.2.10.3 Vibration Test Sequence

The sequence of the vibration tests shall be along the three orthogonal axes. There are no restrictions on which axis is tested first.

4. Environmental Tests

Environmental testing shall be completed on components prior to their delivery for installation on the spacecraft. Following any test, the test item shall be examined for evidence of damage and if it exists, such evidence shall be documented and corrective action initiated.

All test requirements shall be in accordance with this document and the applicable approved equipment (EQ) specifications. Table 4-1 lists the optional and required tests.

Table 4-1. Component Environmental Test Requirements

Test	Qualification /Proto-flight	Acceptance
Electronic Test Article:		
Preconditioning	Optional	Optional
Sinusoidal Vibration	Required	Not Required
Random Vibration	Required	Required
Shock	Required	Not Required
Thermal Vacuum (T/V)	Required	Required
EMI/EMC (ref. KPLO-SP-330-001)	Required	Required*
Acoustics (Payload Sensor Units & Appendages only)	Required	Required

* *Note : A Test is not required except for conducted/radiated emission. Critical conducted/radiated emission shall be tested on all models and limited conducted/radiated emission test("sniff" and "snort") shall be done as well.*

Preconditioning thermal cycling tests may be performed as an option prior to the start of formal proto-flight or acceptance testing for all components. These tests are typically conducted as a good cost effective practice to avoid failures during the final testing. The tests consist of functional and power-applied (DC or RF) thermal cycling tests. Temperature range during preconditioning testing shall be determined from the maximum and minimum acceptance test temperatures as shown in Table 3-1. In no case shall the lower temperature be less than the cold turn-on temperature. Operating time when the hardware is tested as a component can be counted against the accumulated operating time requirement described in section 4.1.

Qualification tests are intended to demonstrate that the test item will function within performance specifications under simulated conditions more severe than those expected from ground handling, launch, and orbital operations. Their purpose is to uncover deficiencies in design and method of manufacture. They are not intended to exceed design safety margins or to introduce unrealistic modes of failure. The design qualification tests may be to either “qualification” or “proto-flight” test levels.

Acceptance tests are intended to identify latent defects in components and demonstrate design margin before they are integrated into the flight spacecraft or before they are accepted as flight spares. Acceptance tests shall be nondestructive and shall not stress the component such that a significant amount of service life is expended (10 % or more).

The vendor has the option to conduct the formal test series with thermal testing first, then dynamic testing, or vice versa. A summary of thermal test requirements is shown in Table 4-2.

Table 4-2. Summary of Thermal Test Requirements

Condition	Qualification /Proto-flight	Acceptance
Thermal Vacuum (T/V)		
Temperature	T_Q(Table 3-1)	T_A(Table 3−1)
Pressure, Torr	$\leq 1 \times 10^{-5}$	$\leq 1 \times 10^{-5}$
Number of cycles	10 / 5	5
Dwell	Figure 4−1 /Figure 4-2	Figure 4-2

NOTE: Any exceptions to the environmental test requirements for components which have heritage to the COMS program or other KARI space programs shall be documented by the subsystem responsible design engineer (RDE) and specified in the applicable equipment (EQ) specification.

Table 4-3 summarizes the test levels and duration for structural mechanical tests, including, acoustics, random vibration, and mechanical shock.

The component test temperature extremes for equipment environmental tests are summarized in Table 3-1 of section 3.1.2. The component design should show positive margins for all ultimate failure modes.

Table 4-3. Structural Mechanical Requirements (TBD)

Test	Qualification	Proto-flight	Acceptance
Random Vibration			
Test Level	Qualification level	Proto-flight level	Limit Level
Duration	2 Minute/Axis	1 Minute/Axis	1 Minute/Axis
Sinusoidal Vibration			
Test Level	1.25 X Limit Level	1.25 X Limit Level	N/A
Sweep Rate*	2 Octaves/Minute	4 Octaves/Minute	N/A
Acoustics			
Test Level	Limit Level + 3 dB	Limit Level + 3 dB	Limit Level
Duration	2 Minute	1 Minute	1 Minute
Mechanical Shock			
Test Level	Qualification Level (Table 3-11)	Qualification Level (Table 3-11)	N/A

* Unless otherwise specified these sine sweep rates shall apply

Any exceptions to the environmental test requirements for components which have heritage to KOMPSAT or other KARI space programs shall be documented by the subsystem responsible design engineer (RDE) and specified in the applicable equipment (EQ) specification.

4.1 Operating Time

Each electronic and electrical component shall accumulate a minimum of 300 hours operating time (including acceptance testing) prior to delivery. This may include operating time at the assembly level. Internally redundant units must accumulate at least 100 hours on each redundant side. If accumulated operating time is less than 300 hours at the completion of acceptance testing, then additional operating time shall be accumulated after the completion of this testing such that the requirement is met. In this case, after the additional hours, a performance test shall be performed prior to delivery.

Accumulated operating time shall be maintained in an operating log. Solar arrays, antenna dishes, and batteries are exempt from the operating time requirements.

4.2 Sinusoidal vibration tests

Sinusoidal vibration test is used to demonstrate the ability of the equipment to withstand low frequency excitations by launchers considering a qualification factor.

4.2.1 Test Parameters

The following test parameters shall apply:

Sinusoidal vibration level: As specified in section 3.1.4.1 and Table 4-3
Test duration : As specified in Table 4-3

4.2.2 Test Procedures

The sinusoidal vibration levels shall be applied at the base of the equipment (interface plane) which shall be hard mounted on the vibration test fixture that has been verified in accordance with section 3.2.10.1.

A functional test shall be conducted before and after the test. The equipments which are active during launch shall be operating and monitored during the test.

Prior to frequency search and after check out of the vibration tests (sinus and random), a low level sine sweep shall be applied on all three axes to demonstrate that the tested equipment has not been degraded.

4.3 Random vibration tests

A full functional test shall precede and follow all random vibration tests. Random vibration proto-flight and acceptance tests shall be performed on components with the unit being tested in the electrical or RF configuration experience during launch. Functional monitoring shall be performed during vibration exposure. Monitor mode instrumentation (reference section 3.2.7.1) shall be required.

All random vibration tests shall be performed on a fixture that has been verified in accordance with section 3.2.10.1 and with the same shaker-fixture-unit clocking as done in the fixture evaluation. Unless otherwise stated, the levels are the monitored inputs at the control accelerometer(s).

4.3.1 Test Parameters

The following test parameters shall apply:

Random test spectra : As specified in Table 3-2, Table 3-3, Table 3-4, Table 3-5, Table 3-6 and Table 4-3

Test duration : As specified in Table 4-3

4.3.2 Test Procedures

Random vibration tests may be performed using a digital control system. A bare fixture random equalization shall have been performed prior to conducting the test in each axis. Verification of conformance with the random spectrum tolerances given in section 3.2.2 shall be made during each equalization test.

The component, including mounting flanges, shall be mounted on the test fixture. Cabling, tubing, etc., shall be attached to the component as required for operation or monitoring of the component. The component being tested shall be subjected to the specified random vibration along each of its three orthogonal axes.

4.4 Acoustic Test

Payload sensor units, antenna reflectors and solar panels shall be acoustically tested. If there has not been an acoustic test, it can be substituted by any other compatible test.

The hardware shall be mounted to the support stand, instrumented and installed in a reverberant acoustic chamber. The hardware shall be subjected to the test level and duration specified in Table 4-3.

A low level reference test as well as visual inspection shall be performed preceding and following the acoustic test.

4.5 Shock Test

Shock test demonstrates the ability of the equipment to withstand the shocks induced by launcher, solar array release mechanisms, antenna release mechanisms and etc.

The equipment shall be mounted to a fixture through the mounting holes of the equipment. The shocks shall be applied at the equipment mounting interface and the levels given in Table section 3.1.4.3 shall be used for qualification.

The equipments which are active during shock events (satellite separation, solar array or antenna deployments) shall be operating and monitored during the test.

4.6 Thermal Vacuum Tests

The thermal vacuum (T/V) test validates the component thermal design and verifies the performance of vacuum sensitive components (e.g., the absence of corona discharge or multipaction in RF components). For component T/V test, the minimum number of 5 thermal vacuum cycles is required as noted in Table 4-2.

QM or PFM equipment shall be tested over the qualification temperature range (TQ) and FM equipment shall be tested over the acceptance temperature range (TA). The number of cycles and the duration of a PFM equipment test shall be equal to the acceptance test number of cycles and duration.

Equipment performance shall be met at qualification temperature limits.

4.6.1 Test Parameters

The following test parameters shall apply:

Temperatures	:	As defined in applicable equipment specification and Table 4-2.
Temperature exposure	:	As described in Table 4-2 and Figure 4-1
*Pressure	:	As specified in Table 4-2

**The test (temperature cycling) may be commenced when the pressure falls below 10^{-4} Torr and a pressure of 10^{-5} Torr or less shall be achieved prior to switch ON the unit.*

**All units shall be able to be operated during spacecraft or subsystem thermal vacuum tests at pressure lower or equal to 10^{-3} mbar (1 Torr = 133 Pa = 1.33 mbar)*

4.6.2 Test Installation

The equipment shall be mounted in a vacuum chamber, in a thermally controlled environment. Temperatures shall be controlled, measured and selected such that it can be guaranteed that the test item experiences actual temperatures equal to or beyond the minimum and maximum qualification temperatures in the test environment. The equipment shall be qualified using the type of fixations and mounting as designed in the equipment specification. The equipment shall be fixed and mounted in accordance with one of the test methods specified in the section 5.1.15.4 of ECSS-E-10-03A.

4.6.3 Thermal Vacuum Qualification

At least one unit representing a series of equipment shall be qualified in thermal vacuum to the requirements and it is required to demonstrate full compliance with the requirement inside the qualification temperature range.

Temperature shall be controlled, measured and selected such that it can be guaranteed that the test item experiences actual temperature equal to or beyond qualification temperatures. Refer to section 3.2 for the tolerances and measurement accuracy.

Throughout the thermal testing, the performance shall be monitored to an extent necessary to detect any anomalous or transient behavior which could be prejudicial to proper functioning of the associated subsystems or of the equipment. In addition, limited performance tests shall be performed at least 4 times during this sequence i.e. initial ambient, hot, cold and final ambient.

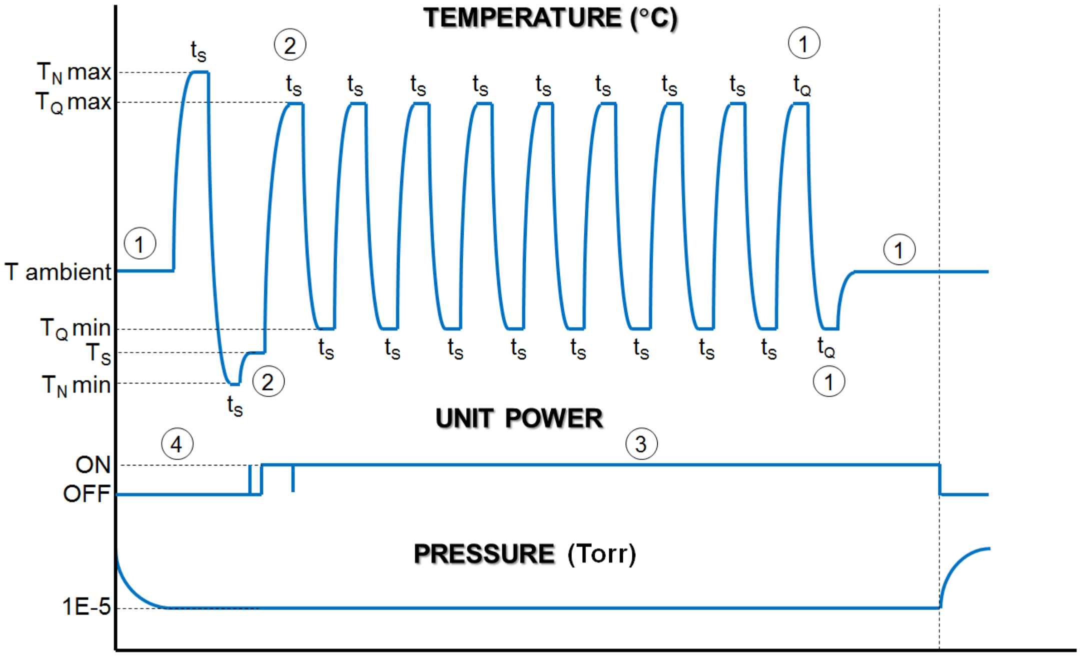
During qualification testing, the same item shall be tested, in the normal post lift off sequence, to the thermal environments appropriate to non-operating, switch-on (start-up) and operating qualification temperature limits. The performance of the test, i.e. number of cycles, temperature ranges and duration are shown in Figure 4-1. If preferred, the temperature cycle profile can be changed to give a hot phase first (for cycling with unit ON).

The minimum number of qualification temperature cycling is 10 for general equipments.

Note : dT/dt ≤ 2°C/min applies only to equipment within the satellite. For equipment outside the satellite higher gradients will be specified in the appropriate equipment specification.

Nomenclature (Figure 4-1) :

T_N max	Maximum non-operating temperature
T_N min	Minimum non-operating temperature
Tambient	Ambient temperature
T_Q max	Maximum qualification temperature
T_Q min	Minimum qualification temperature
T_S	Start-up temperature
t_S	Minimum stabilized temperature time : 1 hr
t_Q	Minimum stabilized temperature time prior to starting performance test : 4 hrs



① Limited performance test

② Unit ON/OFF(Verification of ON/OFF switching capability at T_S and T_Q max)

③ Whenever unit is ON, a selected set of major parameters (current…) are continuously monitored (typically 1 point per minute)

④ For Corona test, some units are ON during pressure drop (see section 4.8)

dT/dt ≤ 2°C/min except if specified in the equipment specification

Figure 4-1. Equipment Thermal Vacuum Qualification Test Profile

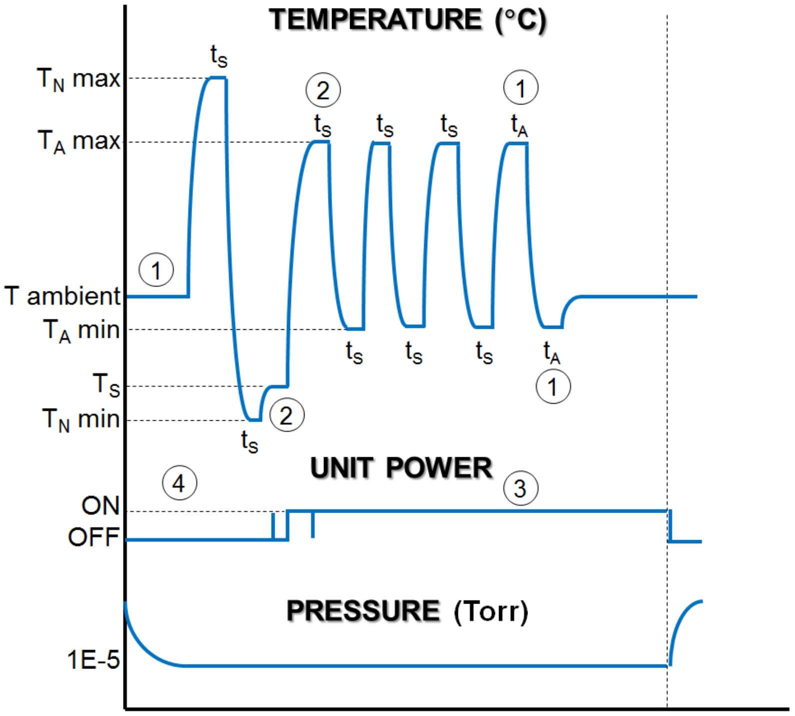
4.6.4 Thermal Vacuum Acceptance

A functional performance test shall be performed on the equipment in vacuum with the temperature of the equipment controlled, measured and selected such that it can be guaranteed that the test item experiences actual temperatures equal to or beyond the maximum and minimum acceptance operating temperatures. Refer to section 3.2 for tolerances and measurement accuracy. In order to ensure the application of the maximum stress condition, the unit shall be operated continuously throughout the test, with a limited performance test done 4 times at the first ambient plateau (reference test), at a hot and at a cold plateau and at the final ambient plateau, and adequate monitoring during the remainder of the test. The required sequence is shown in Figure 4-2. If preferred, the temperature cycle profile can be changed to give a hot phase first (for cycling with unit ON).

The minimum number of acceptance temperature cycling is 5 for general equipments.

Nomenclature (Figure 4-2) :

T_N max	Maximum non-operating temperature
T_N min	Minimum non-operating temperature
Tambient	Ambient temperature
T_A max	Maximum acceptance temperature
T_A min	Minimum acceptance temperature
T_S	Start-up temperature
t_S	Minimum stabilized temperature time : 1 hr
t_A	Minimum stabilized temperature time prior to starting performance test : 2 hrs



① Limited performance test

② Unit ON/OFF (Verification of ON/OFF switching capability at T_S and T_A max)

③ Whenever unit is ON, a selected set of major parameters (current…) are continuously monitored (typically 1 point per minute)

④ For Corona test, some units are ON during pressure drop (see section 4.8)

Figure 4-2. Equipment Thermal Vacuum Acceptance Test Profile

4.7 EMI/EMC Test

Electrical components shall undergo EMC/EMI testing or analysis as specified in Table 4-1. Testing shall be performed for the test parameters and components considered to have the greatest impact on achieving system level compatibility (both system self-compatibility and compatibility with specified external environments). Specific types of testing required for each component shall be specified in the appropriate equipment specification. The test shall verify that:

1) The hardware will operate properly if subjected to conducted or radiated emissions from other sources that occur or could occur during integration and test, launch, or in orbit (susceptibility tests), and
2) The hardware does not generate either conducted or radiated signals that could hinder the operation of internal subsystems or other systems (emission tests).
3) EMI/RFI test for the all the payload shall be performed according to the approved EMC control plans.

If developmental hardware is judged to be representative of flight hardware from an EMC test viewpoint, EMC testing may be performed on engineering models.

The final pass/fail criteria for component level EMC tests shall be the resultant effect on the EMC performance of the integrated spacecraft. Component test over-limit performance shall be analyzed to determine the resultant system level performance. If the analyses indicate acceptable system level performance, the unit design shall be considered acceptable for use.

4.8 Corona-Arcing Test

The aim of this test is to demonstrate the capability of the equipment to correctly operate (without arcing) in a low pressure environment, therefore no critical pressure exists :

- during ascent phase ; SADM, RF power units and units powered during launch will be submitted to all pressure values between 1 bar and 0 bar.
- on-station e.g during the initial outgassing phase or at any time during the operational life ; exposure to a hard vacuum commensurate with orbital altitude will occur at synchronous orbit, this will be 10^{-10} Torr in free space, but may be as high as 0.1 Torr within the spacecraft interior.

A corona test is mandatory for qualification models (EQM, QM, PFM) of SADM, RF power units and units powered during launch. The presence of corona and arcing shall be determined during reduction of pressure for thermal vacuum test of section 4.6. The pressure profile shall be such as to stay at least 1 hour between 10 Torr and 10^{-4} Torr. Corona and arcing shall not occur with the unit operating.

For flight models of SADM, RF power units and units powered during launch, either a test or an analysis (showing that corona is not sensitive to workmanship) has to be performed.

For all units, either a test or an analysis, has to be performed to demonstrate the non-sensitivity to satellite residual internal pressure.

As far as a thermal vacuum test is performed, a corona test is recommended.