SlingShot Interface Control Document

SEOPS, LLC/QUAD-M, INC.

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Table 1-1: Applicable Documents

Doc No.	Rev	Title
SSP 57000	R	Pressurized Payloads Interface Requirements Document
SSP 57003	L	External Payload Interface Requirements Document
SSP 51700		Payload Safety Policy and Requirements for the International Space Station
SSP 52005	F	Payload Flight Equipment Requirements and Guidelines for Safety-Critical Structures
SSP 30233	H	Space Station Requirements for Materials and Processes
SSP 30245	P	Space Station Electrical Bonding Requirements
JSC 20793	C	Crewed Space Vehicle Battery Safety Requirements
MSFC-SPEC-522	B	DESIGN CRITERIA FOR CONTROLLING STRESS CORROSION CRACKING

1. SlingShot Overview

1.1.0 SlingShot Description - The SlingShot system is modular CubeSat Deployer system designed to mount on the Passive Common Berthing Mechanism (PCBM) face of the Cygnus spacecraft and deploy CubeSats from the mounted Deployers. Each Deployer will accommodate 6U of satellite hardware. The launcher can deploy a satellite group inclusive of 1U satellites or a single 6U satellite or a combination of different size satellites totaling up to 6U as dictated by the satellite design. See Figure 1-1.

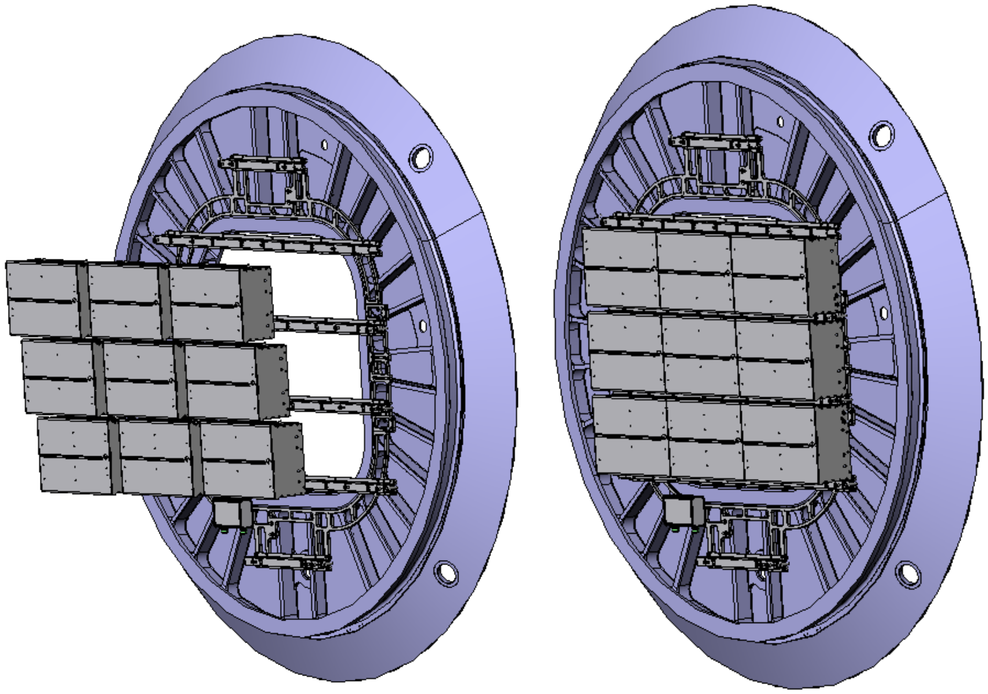


Figure 1-1 SEOPS SlingShot Deployer Configuration on Face of Cygnus PCBM

The SlingShot launcher system is composed of the following major components to be assembled on orbit.

1. Up to 9 preloaded 6U Deployers
2. Internal control box for launcher selection and control
3. Cygnus PCBM bulkhead brackets

The SlingShot components are launched via any ISS Visiting Vehicle (in the pressurized compartment) and soft-stowed as component elements to be assembled on orbit. Hosted payloads may also be accommodated by the SlingShot system which can provide external power/data and hard mounting via the SlingShot bulkhead brackets. Figures 3 through 5 illustrate the Deployer.

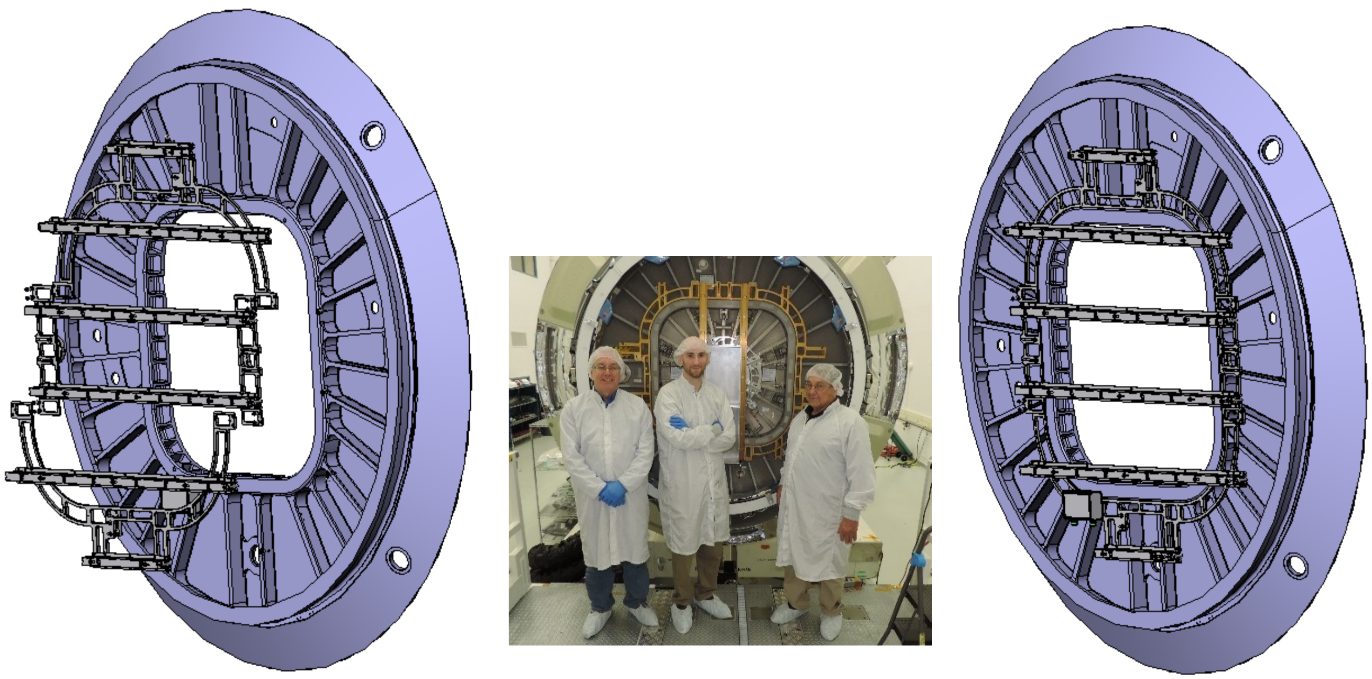


Figure 1-2 SlingShot PCBM Bulkhead Brackets

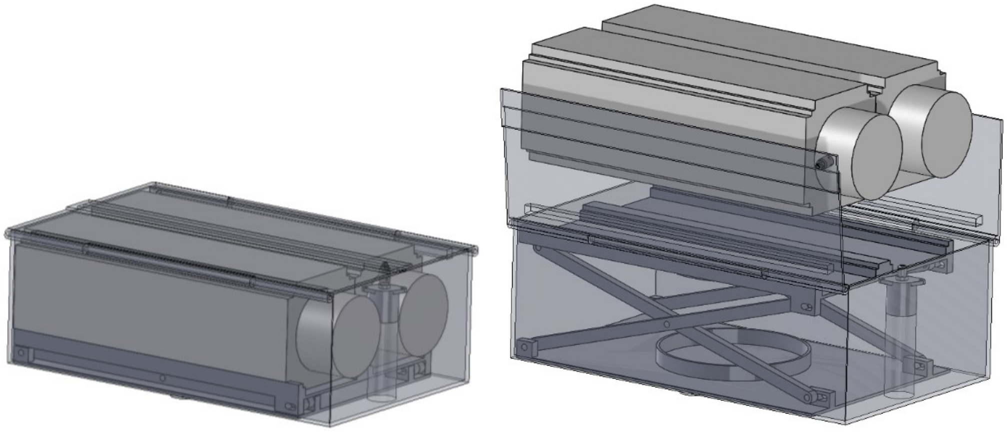


Figure 1-3 SlingShot Deployer Details

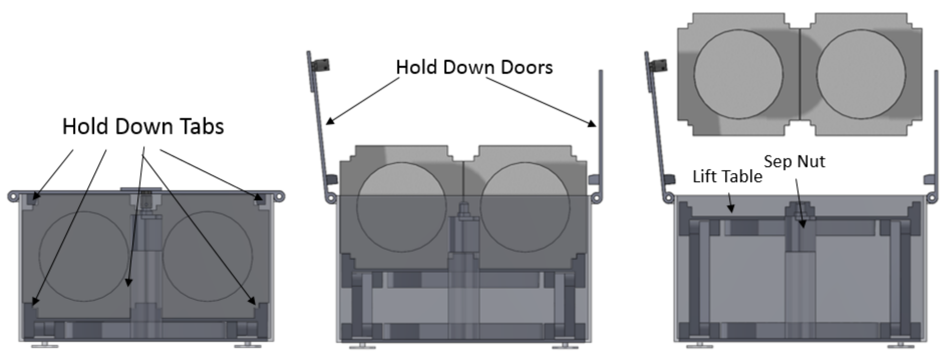


Figure 1-4 SlingShot Deployer Details

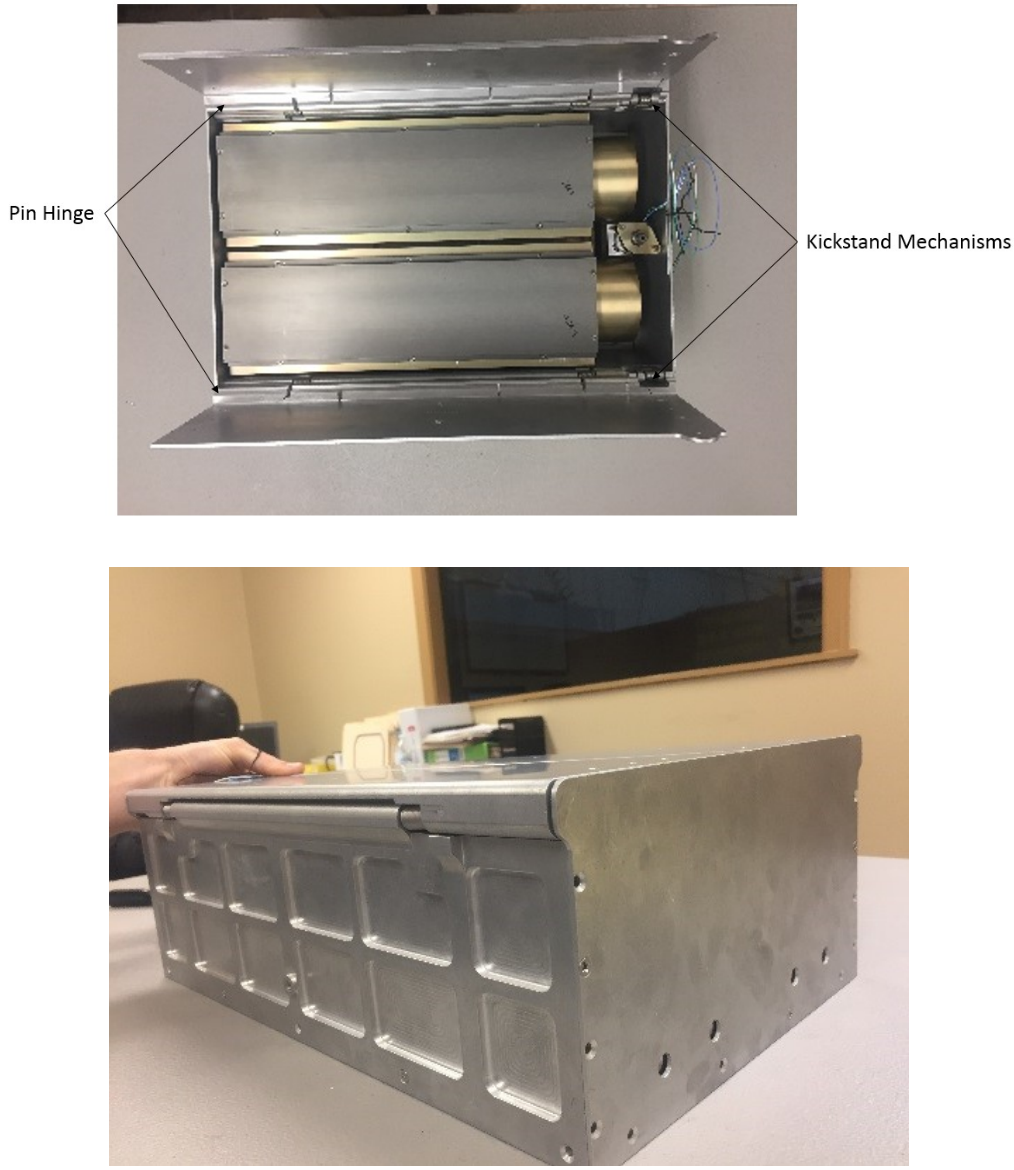


Figure 1-5 Door Hinge Details

Each SlingShot Deployer accommodates 6U satellites or combinations thereof. The SlingShot components and payloads can be transported on any pressurized visiting vehicle (e.g. SpaceX Dragon, Cygnus, Progress, Soyuz, HII Transfer Vehicle (HTV), etc.) to the ISS. Crew interaction with the SlingShot assembly involves transferring the components from the launch vehicle to the Cygnus and assembly of up to 9 Deployers per mission containing satellites for deployment on the Cygnus. Operation of the SlingShot will occur after transfer to the ISS and subsequent installation by the ISS crew on the PCBM bulkhead in the Cygnus. The SlingShot Assembly is not powered or active until deployment and power applied through the Cygnus for satellite deployment activity. This occurs far away from the ISS. Cygnus will maneuver SlingShot to the deploy location and orientation. See Figure .

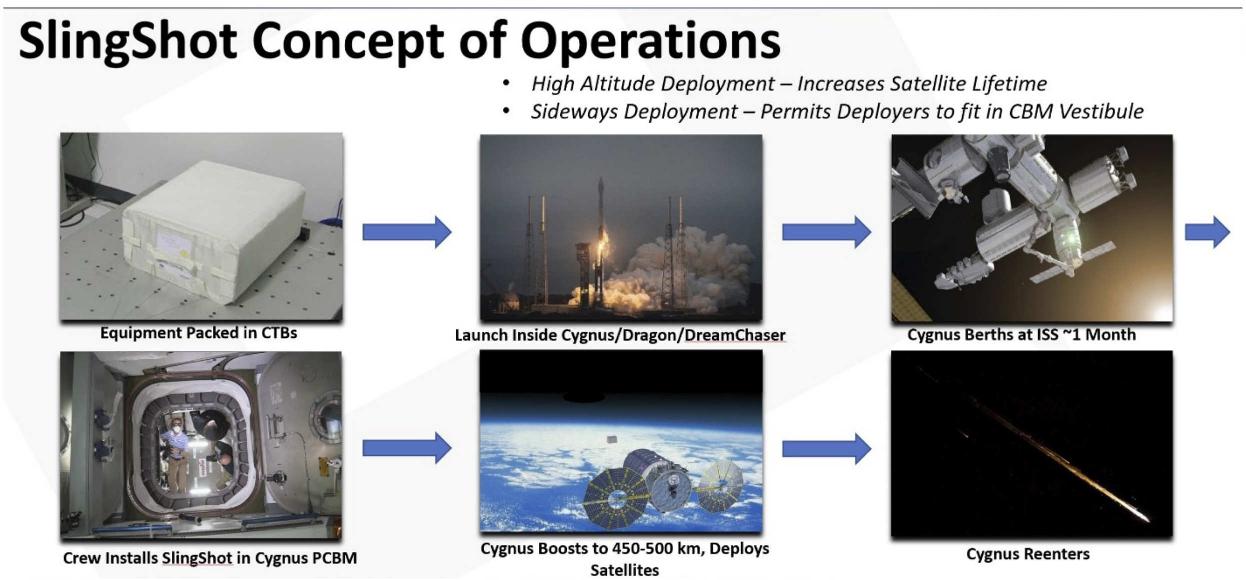


Figure 1-6 SlingShot Deployer Concept of Operations

1.1.1 SlingShot Integration Schedule - Table 1-1 is a standard SlingShot Integration Schedule. The detailed payload schedule will be coordinated between SEOPS and the Payload Developer.

Table 1-1: Standard SlingShot Integration Schedule

Milestone/Activity	Launch-minus Dates (months)
Regulatory Compliance Start (Spectrum Coordination License, Remote Sensing License)	L – 14
Feasibility Study/Contract Signing	L – 9
SEOPS/Customer Safety Data Call Start	L-8.5
SEOPS Safety Initial Assessment Complete	L-8.25
Baseline Interface Control Document	L-8
Phase 0/1/2 Support Data from PD complete	L-8
Phase 0/1/2 Safety Data Package Submittal to NASA	L-7
Phase 0/1/2 Safety Review	L-6
Satellite-Separation System Fit Check	L-5
Phase 3 Support Data from PD complete	L-4.5
Phase 3 Safety Data Package Submittal	L-4
Phase 3 Safety Review	L-3
ISS Program Required Flight Acceptance testing	L-2.5
Payload Delivery to SEOPS	L-2.5
SEOPS Delivery to NASA	L-2.5

1.1.2 SlingShot Fit Check - SEOPS will coordinate to complete mechanical interface checks between the satellite and the Deployer. Fit checks are conducted with the satellite flight hardware and a SEOPS mockup Deployer. Use of flight-like engineering qualification hardware in lieu of flight models must be coordinated with SEOPS.

1.1.3 Satellite Delivery to SEOPS - The payload customer will deliver the integrated payload to the SEOPS Houston facility, or another facility as determined by the ICD, by the dates listed in the schedule. Any special requirements, such as lifting equipment, ground handling hardware, special handling instructions, ESD sensitivity, etc., will be documented in the payload specific ICD.

1.1.4 SEOPS Inspection - SEOPS will inspect the combined payload assembly to verify it meets the appropriate safety and ICD. This includes, but is not limited to, leak checks, mass properties and overall dimensions.

1.1.5 SEOPS Data Gathering for Operations - SEOPS will assess the combined payload assembly to develop products and procedures in support of crew interaction and on-orbit assembly. In order to efficiently assemble the payload, minimize crew time, and maximize mission success, SEOPS will gather information on the payload including an overall evaluation, pictures, and other products as needed. This information will be used to create an effective way for crew to assemble and install the payload, develop supporting procedures, and ensure successful deployment of the satellite.

1.1.6 SEOPS Testing - SEOPS will perform any agreed to testing of the completed assembly based on the Interface Control Agreement. This may include, but is not limited to, grounding checks, bonding checks, fit checks. Any special requirements will be documented in the payload specific ICD.

1.1.7 Customer Ground Servicing - The customer is allowed to perform last minute payload activities at the SEOPS facilities prior to final packaging, based on the agreements in the ICD, as long as these activities are part of the documented and verified payload design. No material or design chances are allowed at this phase of the processing. Once the payload has been delivered to the Cargo Mission Contract, no further payload servicing will be allowed. Any special requirements will be documented in the payload specific ICD.

1.1.8 SEOPS Packaging and Delivery - SEOPS will deliver the loaded Deployer assembly to the Cargo Mission Contracts area for incorporation into its final stowage configuration.

1.1.9 Launch – The NASA Cargo Mission Contract Team is responsible for delivering the final stowed Deployer to the appropriate launch site facility and integration into the ISS visiting vehicle for delivery to the ISS.

1.1.10 Deployer Specification - This specification is intended for the Slingshot deployment system and payload designers. This deployment system is unique in that it can accommodate current standards of CubeSats such as 1U, 2U, 3U, 2UX2U (4U), and 2Ux3U (6U), Tab & Rail type satellites, details are further in the document. The flexibility of this system is achieved through a deployment that is perpendicular to the traditional deployment direction (Figure 1-7).

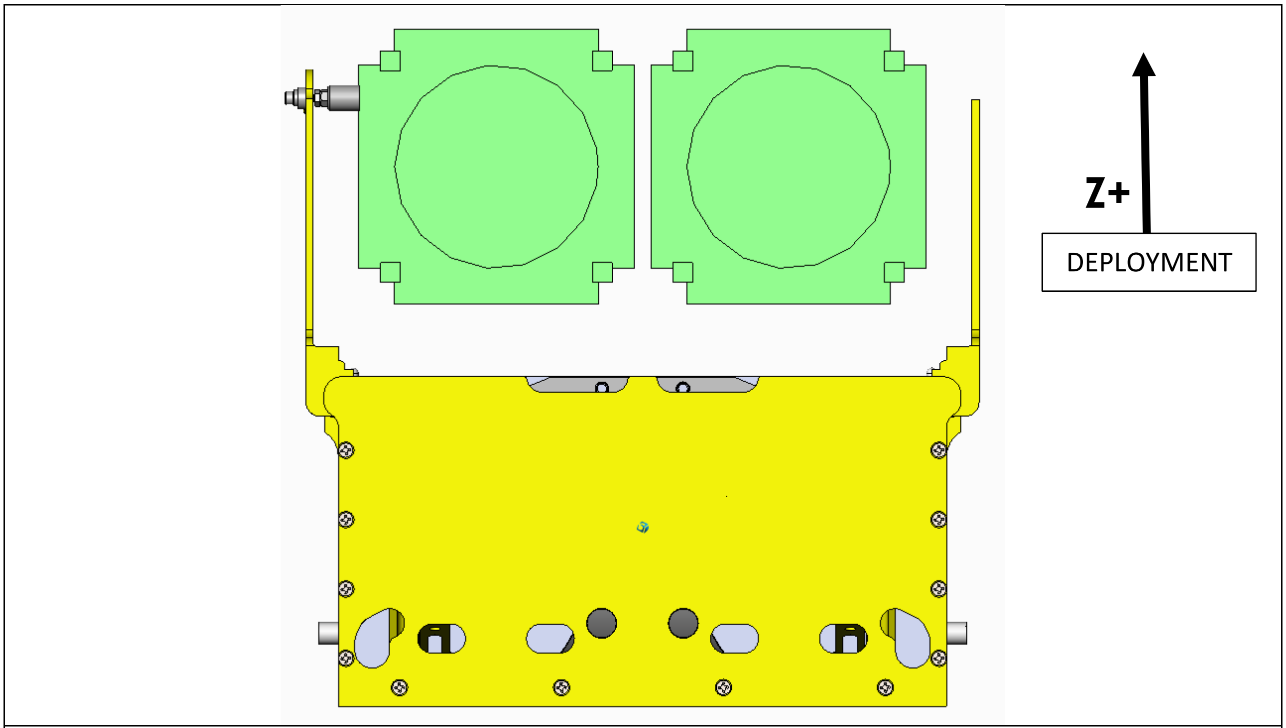


FIGURE 1-7: Deployment

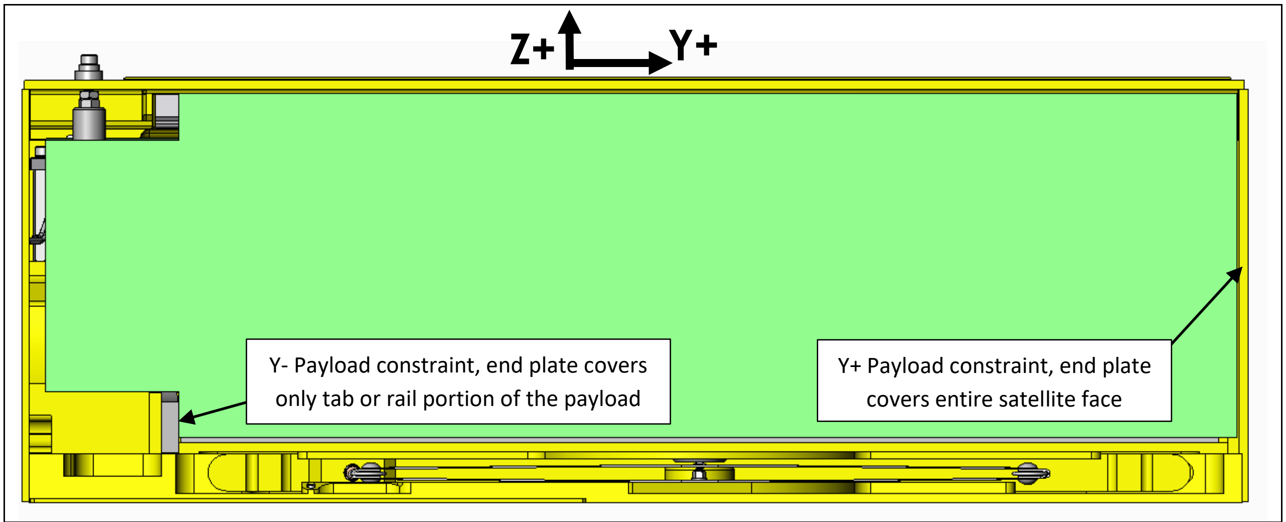
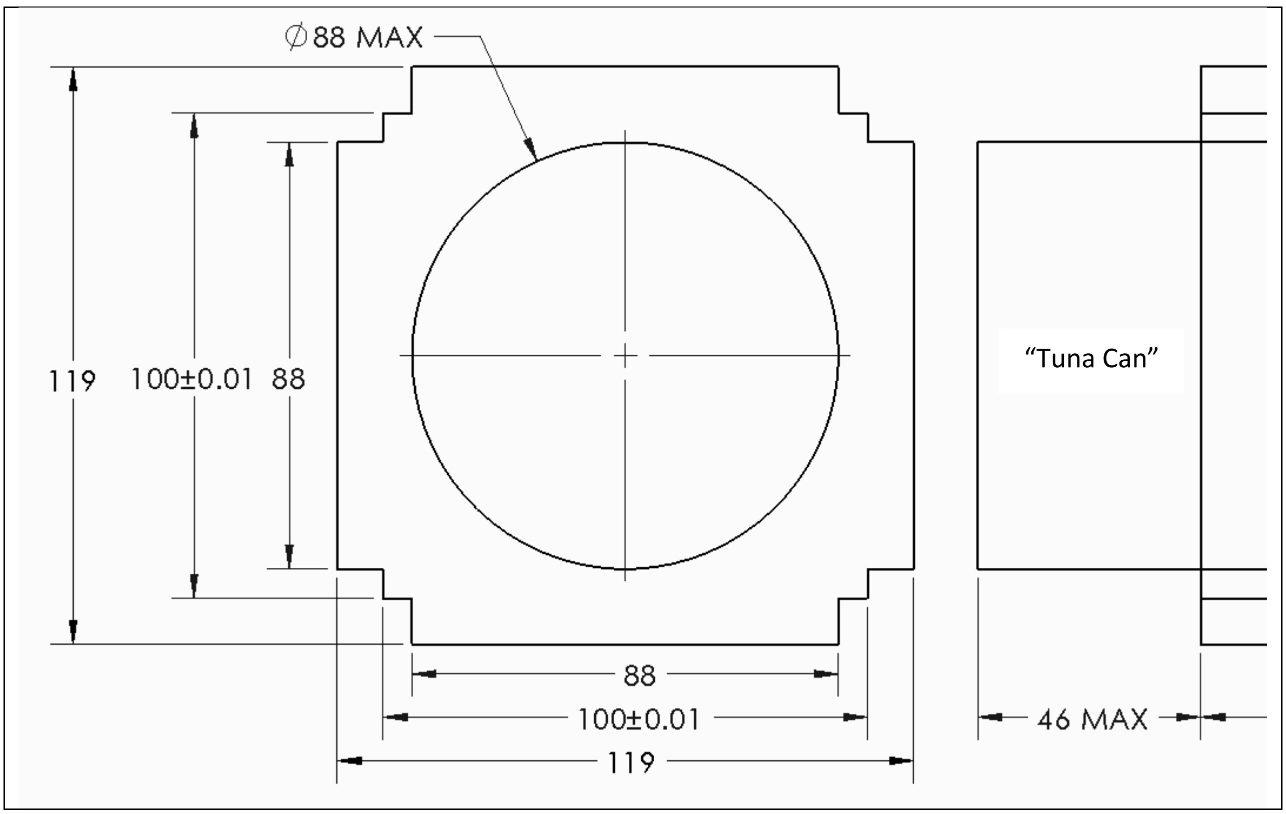


FIGURE 1-8: Payload End Constraints

2. Payload Parameters

Parameter	Unit	1U		2U		3U		6U Rail		6U Tab	
		MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX	MIN	MAX
Mass	lb	-	5.3	-	7.9	-	10.6	-	26.4	-	26.4
Y+ axis activation switch force	lb	-	1.1	-	1.1	-	1.1	-	1.1	-	1.1
Z- axis activation switch force	lb	-	2.1	-	2.1	-	2.1	-	2.1	-	2.1
Payload separation from end plate necessary to change deployment switch state, Y+ axis	in	0.079	-	0.079	-	0.079	-	0.079	-	0.079	-
Payload separation from deployment table necessary to change deployment switch state, Z- axis	in	0.079	-	0.079	-	0.079	-	0.079	-	0.098	-

3. Rail Type Payloads 3U (Figure 3-1 & 3-2) and 6U (Figure 3-3 & 3-4) rail payloads are compatible with Slingshot. Optional "tuna can" is also compatible.



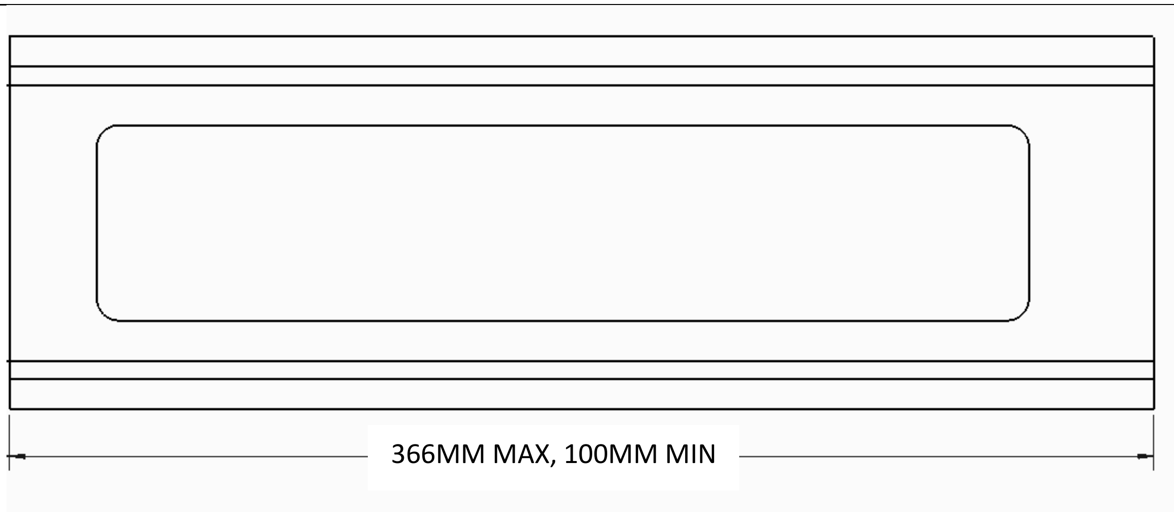
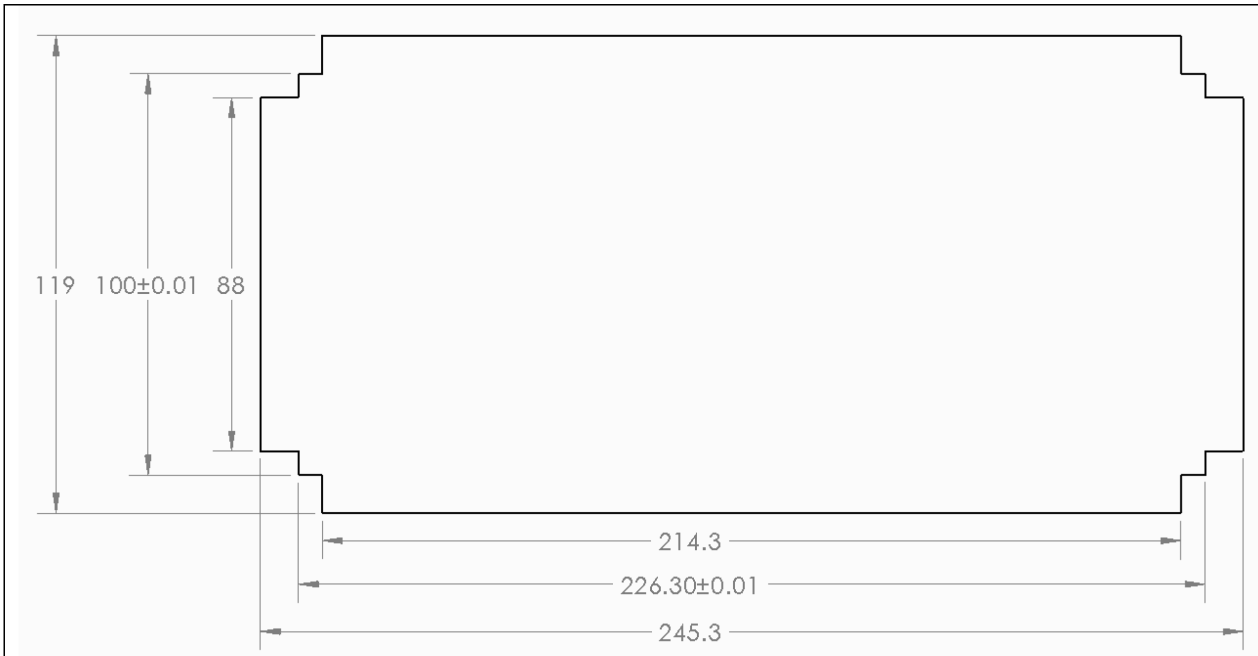


FIGURE 3-1: “1U, 2U, 3U” Rail Payload



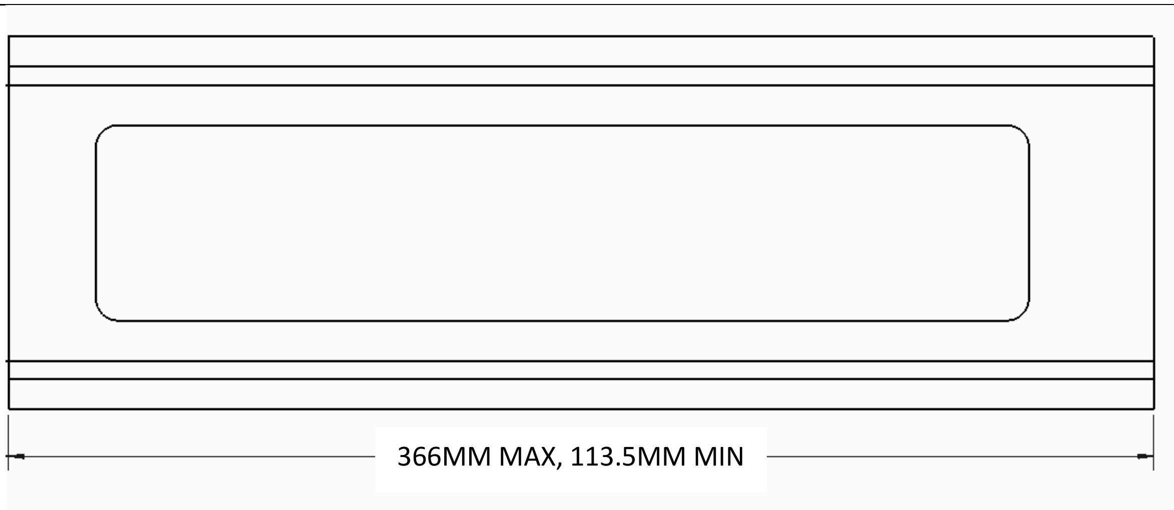


FIGURE 3-2: “6U” Rail Payload

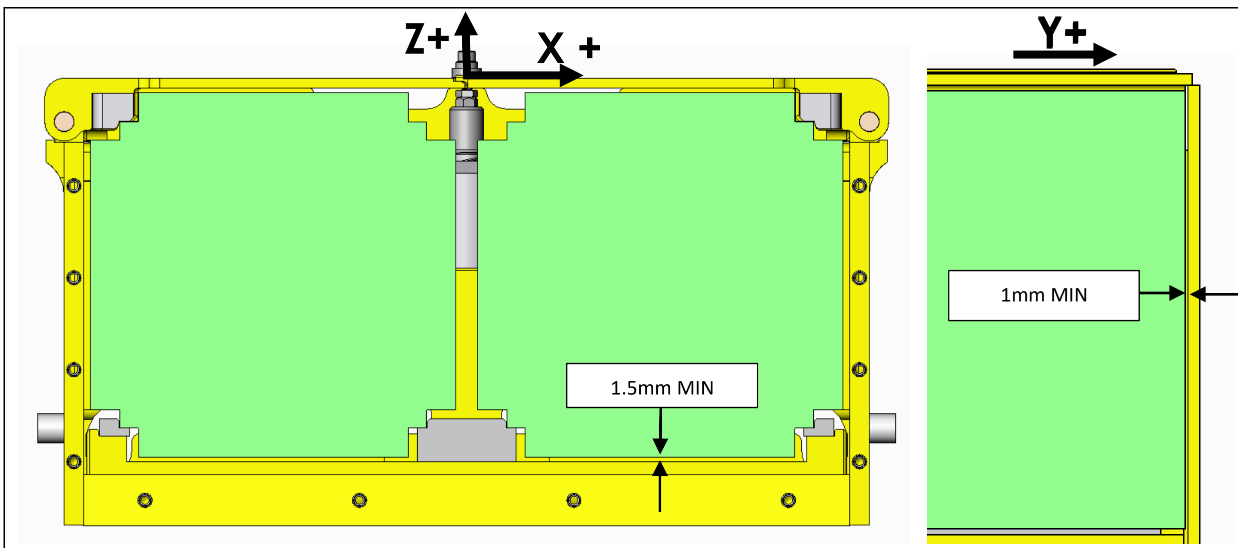


FIGURE 3-3: “3U” Rail Satellites Integrated Into Deployer

Z+ X + Y+

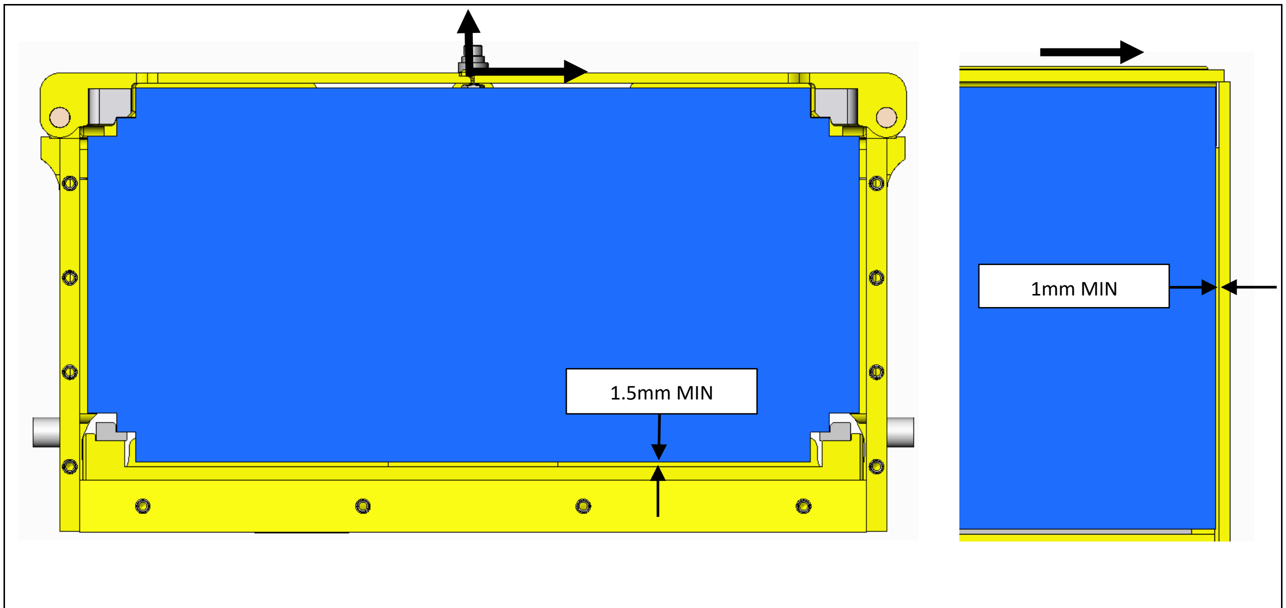
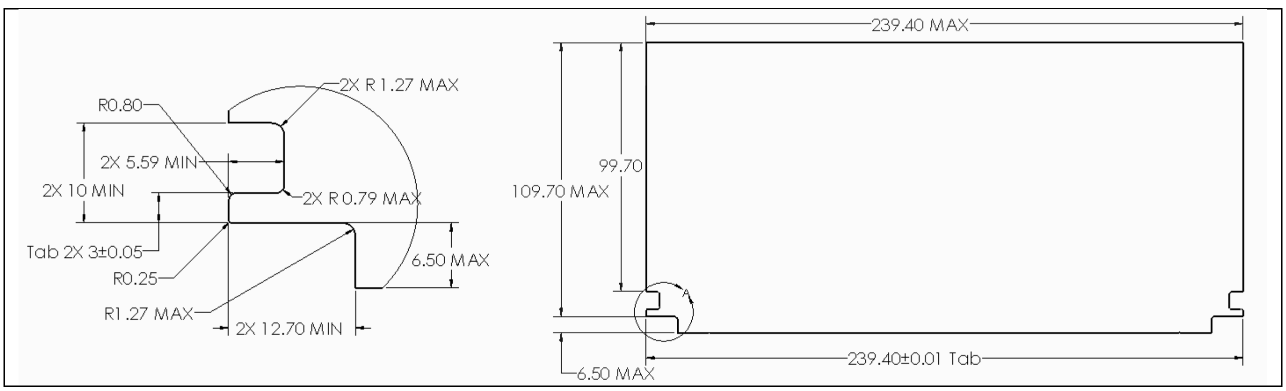


FIGURE 3-4: "6U" Rail Satellite Integrated Into Deployer

4. Tab Type Payloads - 6U (Figure 4-1, 4-2 & 4-3) tab type payloads are compatible with Slingshot. 1U, 2U & 3U tab type satellites are also compatible but not shown (contact SEOPS for more information).



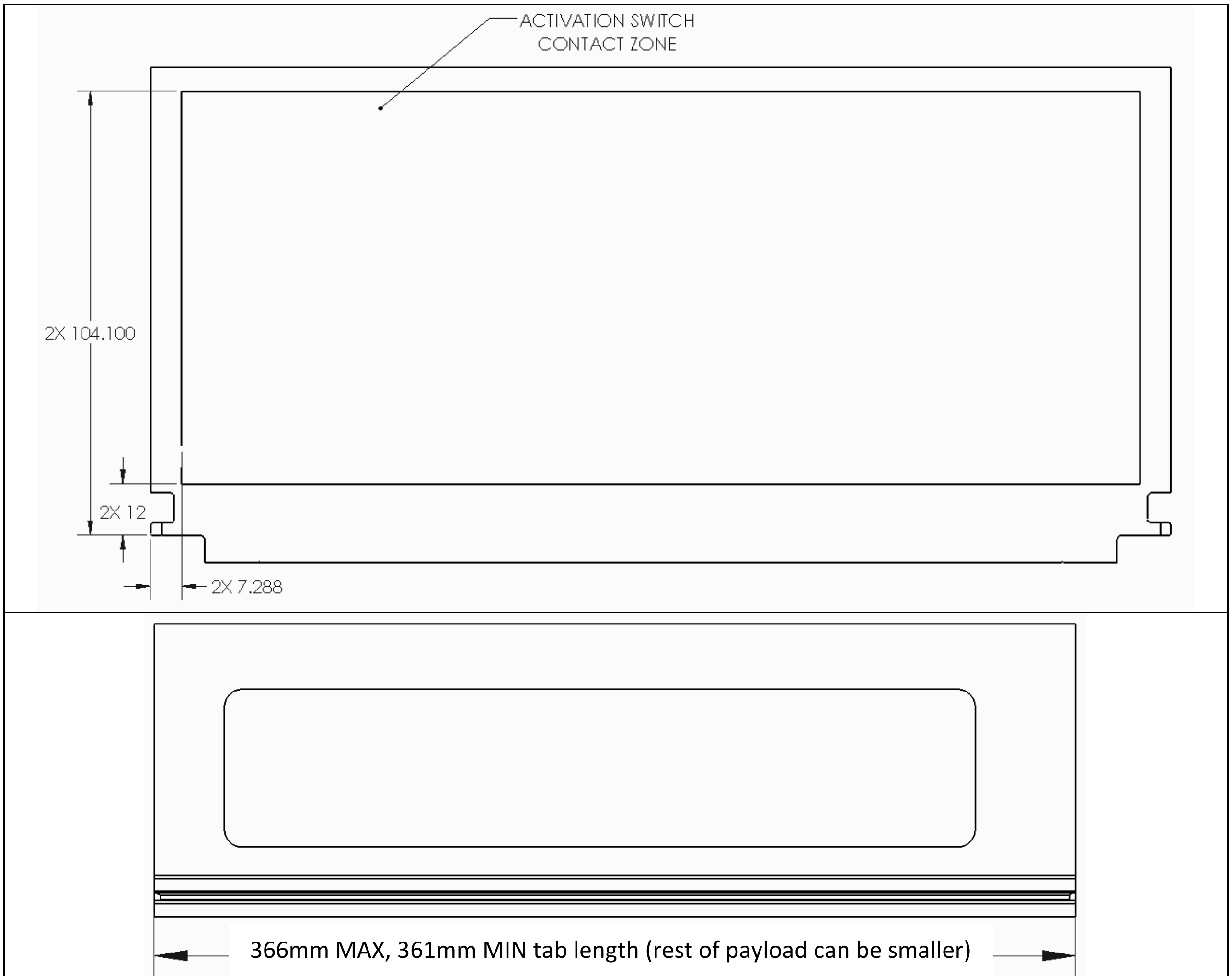


FIGURE 4-1: “6U” Tab Payload

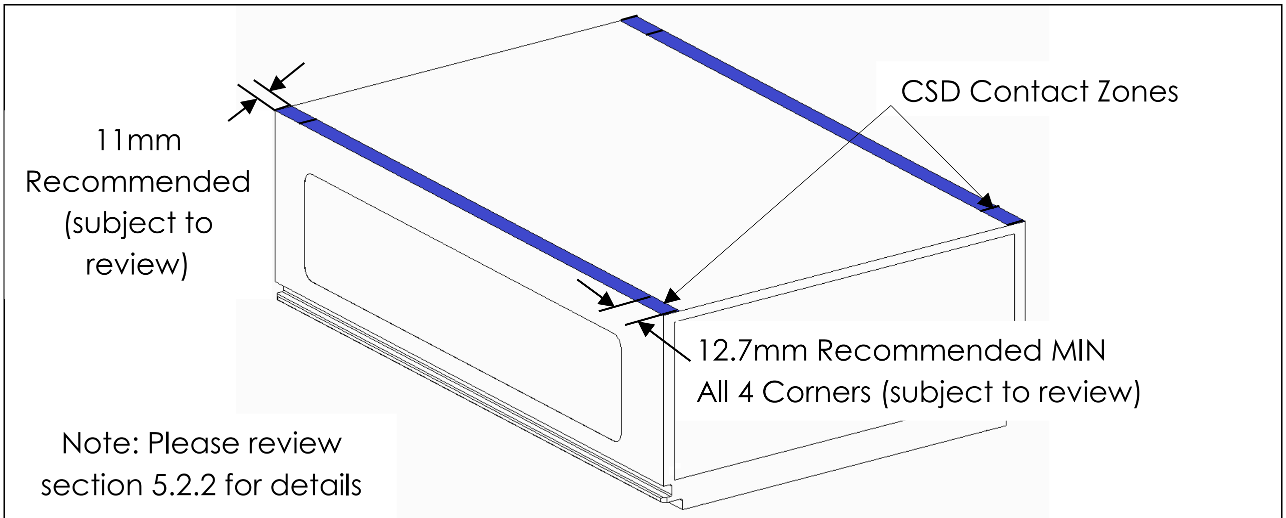


FIGURE 4-2: “6U” Satellite Deployer Contact

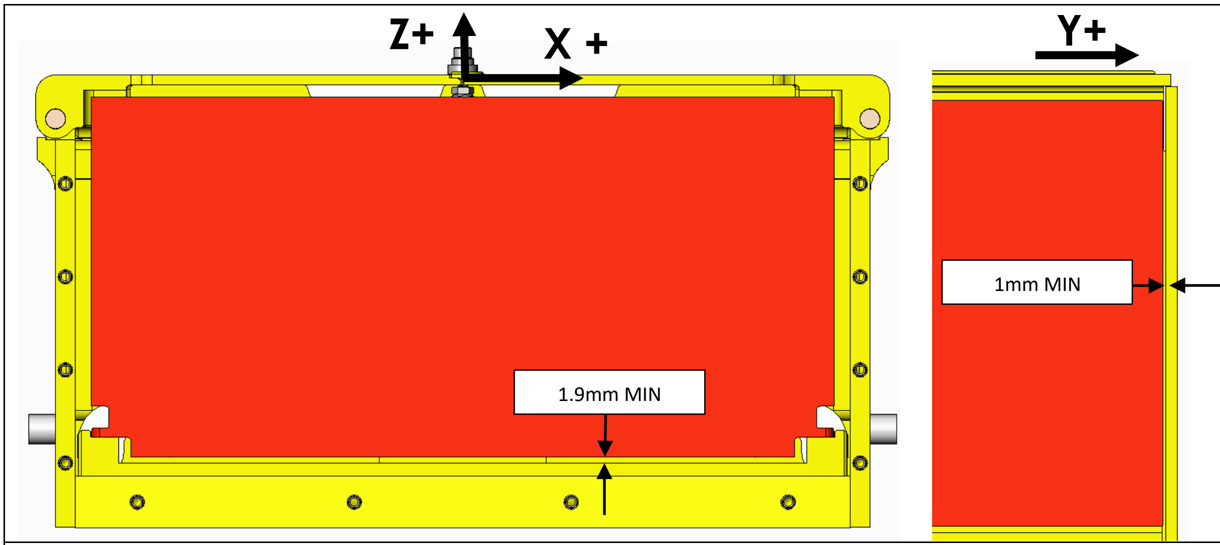


FIGURE 4-3: “6U” Tab Satellite Integrated Into Deployer

5. Payload Access - Payload access is provided via two (2) access panels in both sides of the Deployer. The area that is accessible is shown for both rail and tab configurations in Figure 5-1, 5-2 and 5-3. Please note that if the length of the satellite is less than the max your payload may be placed with other payloads and the access in the Y axis may be change. Example: A payload that is 200mm long may be placed with another payload that is 100mm long, being that the table is 366mm total length consider the potential changes to access.

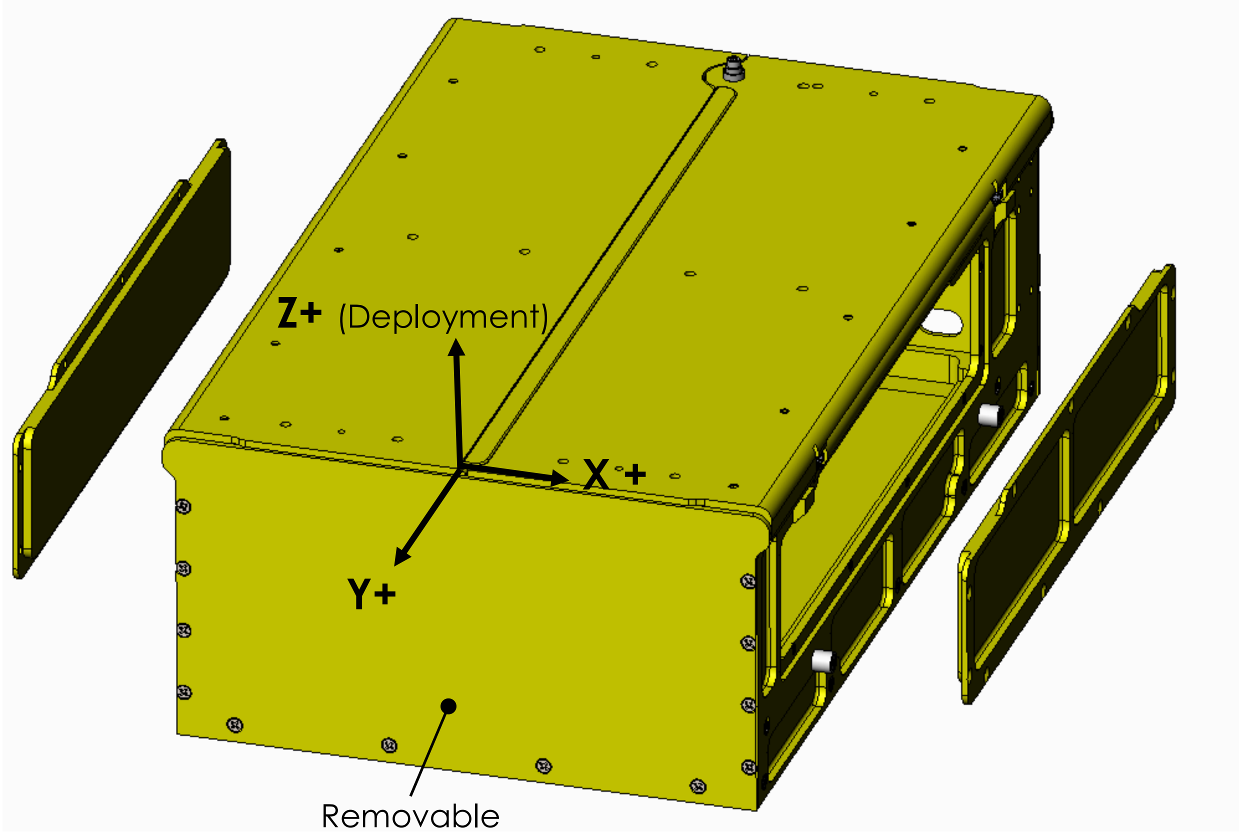


Figure 5-1: Payload Access

It is also possible to remove the end plate for satellite access depending on needs of the payload.

Removable

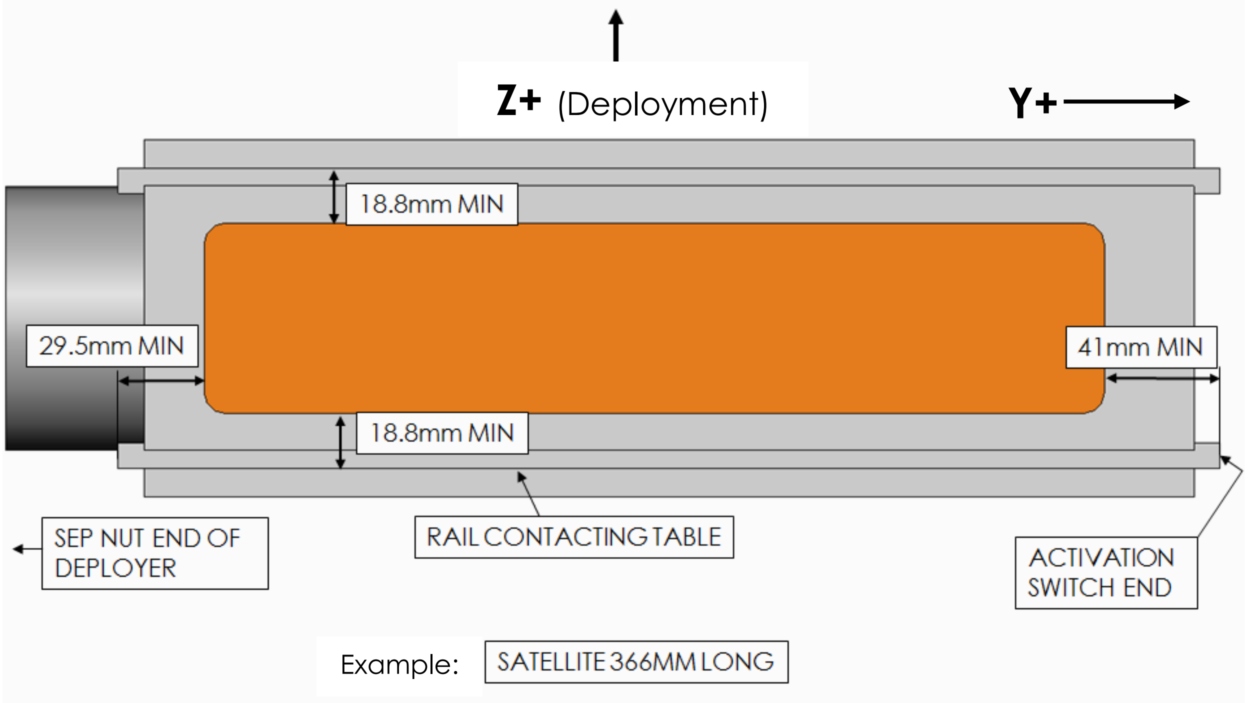


FIGURE 5-2: "3U & 6U" Rail Side Access (Both Sides)

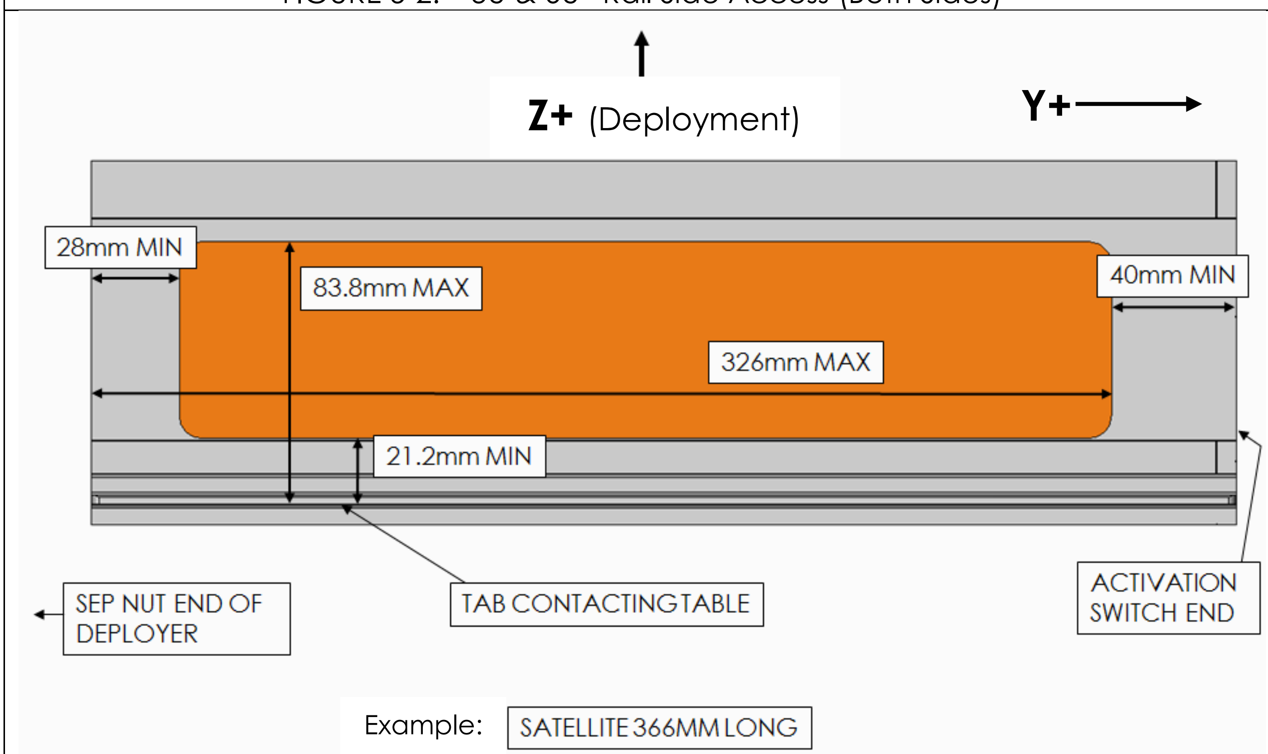


FIGURE 5-3: "6U" Tab Side Access (Both Sides)

6. Mechanical Requirements:

- 6.1.0 Safe/Arm plug, if necessary, shall reside in specified access zones.
- 6.1.1 All deployables must be constrained during deployment via burn wires or other mechanisms. This requirement may be waved after review from SEOPS.
- 6.1.2 No debris shall be generated that will inhibit separation.
- 6.1.3 The maximum dimensions stated in this document are the payload's dynamic envelope and shall include all load cases (vibration, thermal, acoustic, etc.).
- 6.1.4 Maximum dimensions stated in this document apply after plating on all surfaces.
- 6.1.5 Contact SEOPS to determine if payload inhibits (activation switches) are required. If required, locating on the –Z table face such that they contact the CSD's deployment plate is recommended. The deployable contact areas may also be used but consider the effect of tolerance build-up & frictional forces in the Deployer.
- 6.1.6 It is not desirable to have separation switches on the Y+ face of the satellite. Roller switches on the Y+ face are preferred. Roller/lever/plunger type switches on the Z- table face are preferred.
- 6.1.7 Activation switches must maintain a deactivated state until after deployment. The activation throw of the switch and the Deployer gap between the satellite and the Deployer Z-/Y+ face must be taken into consideration (see Figure 3-3, 3-4 & 4-3). The Y+ gap is part of the Deployers design and will not be fully contacting the satellite so deployment can be achieved. Assume there will be a gap as shown (Figure 3-3, 3-4 & 4-3).
- 6.1.7 No appendages or any part of the satellite shall contact the walls of the Deployer, pending SEOPS review.
- 6.2.0 Tab Specific:
  - 6.2.1 Tabs shall be aluminum alloy with yield strength ≥56ksi. 7075-T7351 is common but numerous other alloys also meet this strength requirement.
  - 6.2.2 The two tabs and the structure that contacts the CSD doors on the Z+ face (see Figure 4-2) are the only required features of the payload. The rest of the payload may be any shape that fits within the maximum dynamic envelope. The minimum contact shown can be reviewed by SEOPS to verify the payloads compatibility with the Deployer as it is possible to have contact anywhere along the indicated area so long as it gives contact to properly constrain the satellite during launch and does not interfere with deployment.
- 6.3.0 Rail Specific:
  - 6.3.1 Rail end separation plungers (example 3.2.17 Cal Poly CubeSat Design Specification Rev 13) must be removed before integration with the Deployer.

7. Materials:

- 7.1.0 Stress corrosion resistant materials used from MSFC-SPEC-522 are preferred. Table II materials will be reviewed by SEOPS and Table III materials shall be avoided.
- 7.1.1 Beryllium, cadmium, mercury, silver or other materials prohibited by SSP-30233 shall not be used. Any hazardous materials must be coordinated with SEOPS.
- 7.1.2 Payloads shall comply with NASA guidelines for selecting all non-metallic materials based on available outgassing data. Satellites shall not utilize any non-metallic materials with a Total Mass Loss (TML) greater than 1.0 percent or a Collected Volatile Condensable Material (CVCM) value of greater than 0.1 percent. A list of all non-metallic parts (which may be included in the Bill of Materials below) should include all projected areas of non-metallic parts. Outgassing data can be found at https://outgassing.nasa.gov/.
- 7.1.3 A bill of materials (BOM) must be provided to SEOPS to verify the type of materials used and material masses.

8. Electrical:

8.1.0 Electrical System Design and Inhibits. All Electrical power shall be internal to Satellites. Satellite systems must be safe without Electrical services. Satellite electronics systems design shall adhere to the following requirements.

1) The Satellite operations shall not begin until a minimum of 30 minutes after deployment. Only an onboard timer system may be operable during this 30-minute post deploy time frame. Any timer operation initiated by satellite inhibits must automatically reset should inadvertent separation switch operation occur.
2) If activation of the satellite creates a hazard (e.g. activation of a powerful radio transmitter, activation of propulsion system, activation of a non-eye safe laser, etc.) the Satellite Electrical system design shall incorporate a minimum of three (3) inhibit switches actuated by physical deployment switches as shown in Figure . If activation of the Satellite does not present a hazard to crew or hardware one or two inhibit switches are satisfactory. Contact SEOPS for further clarification.
3) The Satellite Electrical system design shall not permit the battery charging circuit to energize the satellite systems (load), including flight computer (see Figure). This restriction applies to all charging methods.
4) Remove Before Flight (RBF) pins are required. Arming switch or captive jumpers may be an acceptable alternative; contact SEOPS.
5) The RBF pin shall prevent any power from any source operating any satellite functions except for pre-integration battery charging.
6) RBF pins must be capable of remaining in place during integration with the Deployer.

7) All RBF pins, switches, or jumpers utilized as primary Electrical system and RBF inhibits must be accessible for removal at the completion of flight integration with the Deployer.

8) All solar arrays must use an inhibit to preclude flow of current from the solar arrays into the bus in the event they are illuminated prior to deploy where bus actuation presents a hazard.

9) All spacecraft components shall be Electrically bonded per SSP 30245 to ensure the spacecraft is free from Electrical shock and static discharge hazards. Typically, spacecraft components may be bonded by either nickel plating or chemical film treated faying surfaces or dedicated bonding straps. A designated Single Point Ground shall be accessible on the Payload's exterior conductive surface.

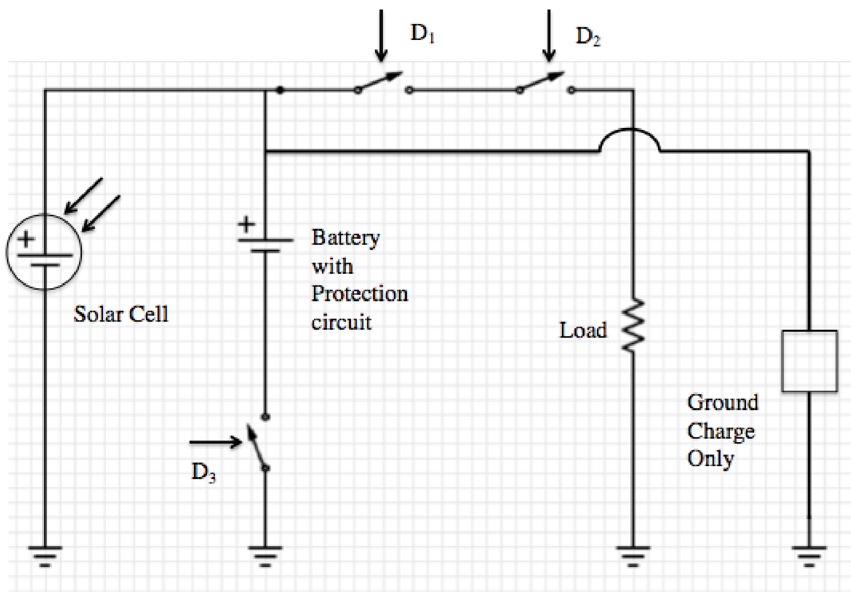


Figure 8-1: Satellite Electrical Subsystem Block Diagram (note: RBF pins switches not shown)

- Reference Example -

8.2.1 Batteries - Battery requirements for spacecraft flight onboard or near the ISS are derived from the NASA requirement document JSC 20793 Crewed Space Vehicle Battery Safety Requirements. Specific provisions for battery use are designed to assure that a battery is safe for ground personnel and ISS crew members to handle and/or operate during all applicable mission phases and particularly in an enclosed environment of a crewed space vehicle. These NASA provisions also assure that the battery is safe for use in launch vehicles, as well as in unpressurized spaces adjacent to the habitable portion of a space vehicle. The required provisions encompass hazard controls, design evaluation, and verification. Evaluation of the battery system must be complete prior to certification for flight and ground operations. To support this objective information on the battery system must be provided to SEOPS as soon as possible. For example, certain battery cell chemistries and battery configurations may trigger higher scrutiny to protect against thermal runaway propagation. It is imperative that SEOPS receive all requested technical data as early as possible to assure the necessary safety features are present to control the hazards associated with a particular battery design. True in nearly every case, redesign efforts greatly impact the payload developer both in cost and schedule. This can often be avoided by consulting with SEOPS before hardware is actually manufactured (if possible). Cell/Battery testing associated with the verification of the safety compliance must be completed prior to safety certification of the spacecraft. To be compliant to the requirements herein, every battery design, along with its safety verification program, its ground and/or on-orbit usage plans, and its post-flight processing should be evaluated and approved by the customer and SEOPS.

8.2.2 Battery Hazards - The possible sources of battery hazards are listed below and shall be identified for each battery system. Applicable hazards will be evaluated to determine and to identify design, workmanship, and other features to be used for hazard control (Electrical, mechanical, and/or thermal).

Potential Battery Hazards:

- Fire/Explosion Hazard
- Flammability
- Venting of Battery Enclosure
- Burst of Pressurized Battery Chemistries
- Overcharge Failure/Over-discharge Failure
- External Short Circuit
- Internal Short Circuit Failure
- Thermal Runaway Propagation
- Chemical Exposure Hazards
- Mechanical Failure
- Seals and Vents
- Electrical Hazards
- Extreme Environment Temperature Hazards

8.2.3 Battery Types - Although any battery may be made safe to fly in the crewed space vehicle environment there are some batteries that are impractical to make safe. For example, lithium-sulfur dioxide cells have built-in overpressure vents that will release SO2 (sulfur dioxide) gas and other electrolyte components that are highly toxic; thus, these are unacceptable in the habitable area of a space vehicle. However, these chemistries have been used safely in the non-pressurized areas of crewed spacecraft. Often the cells used in batteries for crewed space vehicle are commercially available.

Battery types Typically used in spacecraft include:

- Alkaline-manganese primary
- LeClanche (carbon-zinc) primary

- Lead-acid secondary cells having immobilized electrolyte
- Lithium/lithium-ion polymer secondary (including lithium-polymer variation)
- Lithium metal anode primary cells having the following cathodic (positive) active materials:
- Poly-carbon monofluoride
- Iodine
- Manganese dioxide
- Silver chromate
- Sulfur dioxide (external to habitable spaces only)
- Thionyl chloride
- Thionyl chloride with bromine chloride complex additive (Li-BCX)
- Iron disulfide
- Lithium sulfur
- Mercuric oxide-zinc primary
- Nickel-cadmium secondary
- Nickel-metal hydride secondary
- Silver-zinc primary and secondary
- Zinc-air primary
- Sodium-sulfur secondary (external to habitable space)
- Thermal batteries

Note: Pressurized battery chemistries require coordination with SEOPS.

8.2.4 Required Battery Flight Acceptance Testing - Acceptance screening tests are required for all cells intended for flight to ensure the cells will perform in the required load and environment without leakage or failure. See Appendix A for the detailed battery testing plan.

8.2.5 Internal Short - Protection circuity and safety features shall be implemented at the cell level.

- Application of all cells shall be reviewed by SEOPS.
- Charger circuit and protection circuit schematics shall be reviewed and evaluated for required failure tolerance.

8.2.6 External Short Circuit –

- Circuit interrupters that are rated well below the battery's peak current source capability should be installed in the battery power circuit. Interrupters may be fuses, circuit breakers, thermal switches, PTCs, or other effective devices. Circuit interrupters other than fuses should be rated at a value that is equal to or lower than the maximum current that the cell is capable of handling without causing venting, smoke, Explosion, fire, or thermal runaway.
- The battery case is usually grounded/bonded to the structure; the interrupters should be in the ground (negative) leg of a battery where the negative terminal is connected to ground. Where the circuit is "floating," as in plastic battery cases

used in those for portable electronic devices, the circuit interrupters can be placed in either leg. In either case, the circuit interrupters should be placed as close to the cell or battery terminals as the design will allow maximizing the zone of protection.

- All inner surfaces of metal battery enclosures shall be anodized and/or coated with a non-Electrically conductive electrolyte-resistant paint to prevent a subsequent short circuit hazard.
- The surfaces of battery terminals on the outside of the battery case shall be protected from accidental bridging.
- Battery terminals that pass through metal battery enclosures shall be insulated from the case by an insulating collar or other effective means.
- Wires inside the battery case shall be insulated, restrained from contact with cell terminals, protected against chafing, and physically constrained from movement due to vibration or shock.
- In battery designs greater than 50 Vdc, corona-induced short circuits (high-voltage induced gas breakdown) shall be prevented.

8.2.7 Battery Charging – Battery charging is not permitted on-orbit but can be performed on the ground prior to hardware turnover, including in the Deployer.

8.2.8 Battery Energy Density – For battery designs greater than 80-Wh energy employing high specific energy cells (greater than 80 watt-hours/kg, for example, lithium-ion chemistries) require additional assessment by SEOPS due to potential hazard in the event of single-cell, or cell-to-cell thermal runaway. The best design practice for batteries of this size is physical separation of or thermal barriers between the battery cells to prevent thermal runaway situations.

8.2.8 Lithium Polymer "Pouch" Cells – Lithium Polymer Cells i.e. "pouch cells" shall be restrained at all times to prevent inadvertent swelling during storage, cycling, and low pressure or vacuum environments with pressure restraints on the wide faces of the cells to prevent damage due to pouch expansion. Coordinate with SEOPS for guidance on specific implementation.

8.3.0 Radio Transmitter System. Satellite providers shall fill in the following table of RF transmitter parameters to evaluate any potential hazards or RF interferences with ISS or Cygnus.

Table Error! No text of specified style in document.**-1: RF Transmitter Details**

Transmitter Specification	
Manufacturer	
Model	Model #: S/N:
Maximum power output to antenna [W]	
Maximum transmitter field strength (volts/meter); assume 1 meter from the source and transmitter radiating with deployed antenna	
TX Manufacturer	
TX Model No	
TX Antenna Manufacturer:	
Antenna Gain: [dBi]	
Frequency Upper [MHz]	
Frequency lower [MHz]	
Circuit Loss: [dB]	
Antenna Type: Other, dipole, helix, horn, loop, monopole, patch, phased array, reflector, slot, spiral	
Antenna Polarization: [Other, Horizontal, Left Handed EllipticDI, Right Handed EllipticDI, VertICDI	
Antenna Axial Ratio: [dB]	
Antenna location (with respect to CubeSat body)	

8.3.1 – Electromagnetic Interference for On-Orbit - Satellites may be exposed to the following EMI environment on the ISS.

FREQUENCY	RS03PL LIMIT (V/m)
14 kHz - 400 MHz	5
400 MHz - 450 MHz	30
450 MHz - 1 GHz	5
1 GHz - 5 GHz	25
5 GHz - 6 GHz	60
6 GHz - 10 GHz	20
13.7 GHz - 15.2 GHz	25

COMMENT: The less stringent RS03PL limit was developed to envelope the electric fields generated by ISS transmitters and ground–based radars tasked to perform space surveillance and tracking. Ground–based radars that are not tasked to track the ISS and search radars that could momentarily sweep over the ISS are not enveloped by the relaxed RS03PL. For most end items, the minimal increase of EMI risk for the reduced limits is acceptable.

The RS03PL limit does not account for module electric field shielding effectiveness that could theoretically reduce the limits even more.

9. Propulsion Systems

9.0.1 – Propulsion System - The propulsion system will need to be assessed for hazard potential. SEOPS will assist in the identification of hazards. Mechanical hazards may be related to pressure containment, flow containment, leakage, etc. Systems may also have hazard potential if inadvertent operation of the propulsion system in or around ISS could cause a catastrophic or Critical hazard. This particular hazard can be mitigated by simply requiring activation of two or more thrusters to provide lateral motion (i.e. if one thruster were inadvertently actuated it would only impose rotational motion to the spacecraft). Also, low thrust (e.g. ion thrusters) limit hazard potential by providing long response times to mitigate any collision hazards that may be present. Depending on hazard potential, both mechanical and electrical fault tolerance may be required.

Systems with toxic propellant are not be allowed onboard ISS. Propellants with explosive potential may not be approvable. Acceptable propellant type must be coordinated with SEOPS.

9.0.2 – Pressure Vessels - Pressure vessels (i.e. any vessel able to hold more than 2 atm of pressure at any time in its operation) may be made acceptable for Flight Safety with proper controls for any hazard potential both for inside ISS and outside ISS. Payloads should expect to provide documentation with respect to the materials used, tank history (including cycles and life time assessment) and control measure to assure tank integrity (damage control plan), testing performed, fracture control measures planned, inspection process and methods, etc. wherever hazard potential is present. All pressure vessels shall be DOT certified or have a DOT issued waiver for transportation across the US. Use of non-DOT certified pressure vessels generally is not be permitted. Exceptions must be coordinated with SEOPS. Systems will have to demonstrate via test that required factors of safety are present for tanks, lines and fittings that can be exposed to maximum design pressure. Pressure vessels and components procured from third party vendors must have proper certification records or the customer must develop the appropriate records to assure that the systems are safe for satellite use.

10. Environments

10.0.1 Ground Handling and Transportation Loads - Payload safety-critical structures shall (and other payload structures *should*) provide positive margins of safety when exposed to these accelerations.

TABLE 10-1. GROUND HANDLING AND TRANSPORTATION LOAD FACTORS

	Nx (g)	Ny (g)	Nz (g)	Rx (rad/sec^2)	Ry (rad/sec^2)	Rz (rad/sec^2)
I (1,2)	+/- 5.0	+/- 3.5	+2.0/ -3.5	N/A	N/A	N/A
S (1,2)	+/-2.0	+/- 2.0	+2.0/ -3.5	N/A	N/A	N/A

Notes:

1) The reference frame for the ground handling and transportation load factors with respect to the directions of motion is as follows:

   X: Longitudinal direction along axis of motion.

   Y: Y axis is perpendicular to the x and z axes and completes the right handed coordinate system.

   Z: Z axis is perpendicular to the x and y axis. Positive direction is vertically upward. Gravity (g) is acting in the z axis in the negative direction.

2) (I) indicates that the loads occur independently in the three directions (except for gravity). (S) indicates that the loads occur simultaneously.

3) These levels envelope the maximum ground handling and transportation loads.

10.0.2 Acceleration Loads - Payload safety-critical structures shall (and other payload structures *should*) provide positive margins of safety when exposed to the accelerations documented in **Table 10-2** at the CG of the item, with all six degrees of freedom acting simultaneously. The acceleration values are applicable to both soft stowed and hard mounted hardware. (Per SSP 57000, Section D.3.1.1)

Table 10-2: Launch Load Factors Envelope

	Nx (g)	Ny (g)	Nz (g)	Rx (rad/sec^2)	Ry (rad/sec^2)	Rz (rad/sec^2)
Launch	+7.7/-9.0	+/- 11.6	+/- 11.6	+/- 70.8	+/- 70.8	+/- 70.8

All analysis and or testing shall be in accordance with the guidelines specified in SSP 52005 for payload hardware. SEOPS will provide guidance on what structures are safety critical and how to complete structural analysis.

10.0.3 Random Vibration Loads - Payload safety-critical structures packed in foam or bubble wrap and enclosed in hard containers such as lockers, boxes, or similar structures, and payload safety-critical structures packed in foam or bubble wrap and soft stowed in bags shall meet the specified performance requirements when exposed to the maximum flight random vibration environments defined in **Table 10-3.** The standard stowage configuration is the payload wrapped in bubble wrap.

TABLE 10-3. ATTENUATED RANDOM VIBRATION ENVIRONMENTS FOR END ITEMS SOFT-STOWED IN A SINGLE CTB, X/Y/Z AXIS

Frequency (Hz)	**20 lb ORU in Pyrell in a Single CTB**
20	0.1465 (g^2/Hz)
20-153	-9.76 (dB/oct)
153	0.0002 (g^2/Hz)
153-190	0 (dB/oct)
190	0.0002 (g^2/Hz)
190-250	0 (dB/oct)
250	0.0002 (g^2/Hz)
250-750	0 (dB/oct)
750	0.0002 (g^2/Hz)
750-2000	0 (dB/oct)
2000	0.0002 (g^2/Hz)
OA (grms)	**1.29**

10.0.4 Random Vibration Testing – SEOPS prefers all satellie providers to test to the following spectra using a SEOPS provided SlingShot Deployer to hold their satellites during the random vibration test.

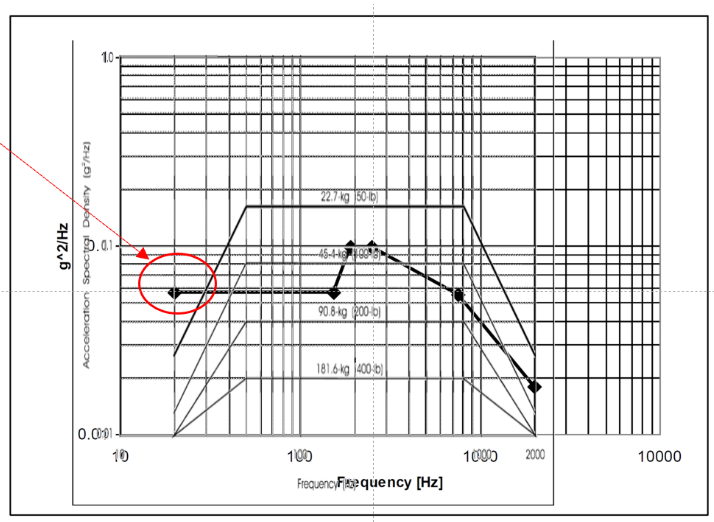
SlingShot Verification Random Vibration Testing Spectra

FREQUENCY	LEVEL
20 – 153 Hz	0.057 g^2/Hz
153 – 190 Hz	+7.67 dB/oct
190 – 250 Hz	0.099 g^2/Hz
250 – 750 Hz	-1.61 dB/oct
750 Hz	0.055 g^2/Hz
750 – 2000 Hz	-3.43 dB/oct
2000 Hz	0.018 g^2/Hz
Composite	9.47 g root mean square (rms)
Duration	60 seconds

This spectra will be applied to a hard mounted SlingShot Deployer with satellites under test installed in the SlingShot Deployer. This spectra will be applied to each axis (X,Y,Z).

In lieu of the above test, the GSFC-STD-7000A Generalized Random Vibration Test Levels test (see below) are acceptable, however, there is a small area (between 20 and 30 Hz) that SEOPS recommends to modify to envelope both test cases if GSFC-STD-7000A Generalized Random Vibration Test Levels are used.

Area not enveloped by GEVS



Comparison of GEVS to ISS VV Spectra

SEOPS recommends performing this test using a SEOPS provided SlingShot Deployer to hold their satellites during the random vibration test.

Table 2.4-3
Generalized Random Vibration Test Levels
Components (ELV)
22.7-kg (50-lb) or less

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

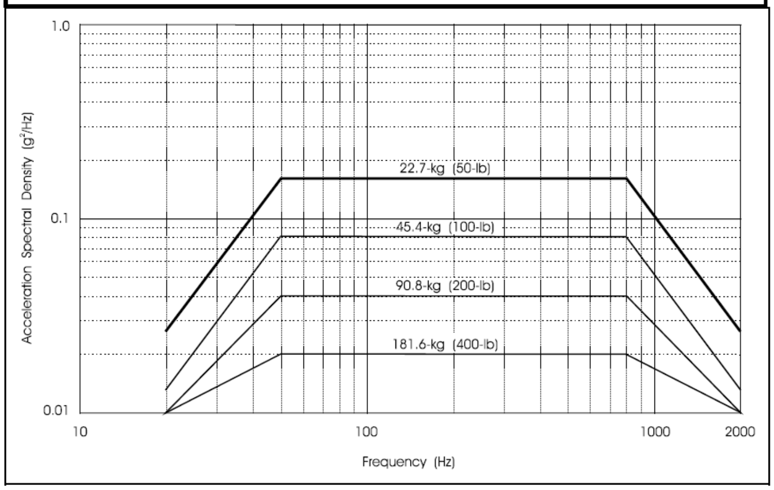
The acceleration spectral density level may be reduced for components weighing more than 22.7-kg (50 lb) according to:

	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).



GSFC-STD-7000A

10.0.5 Acoustic Environment – See Table 10-4 and Figure 10-1.

TABLE 10-4. ACOUSTIC ENVIRONMENT FOR LAUNCH

OCTAVE BAND CENTERED FREQUENCY [Hz]	SOUND PRESSURE LEVEL Ref. $2x10^{-5}$ N/m^2 (20 microPascals) [dB]
31.5	127.5
63	125.3
125	131.8
250	136.0
500	131.0
1000	129.0
2000	119.0
4000	100.0
Overall	139.3

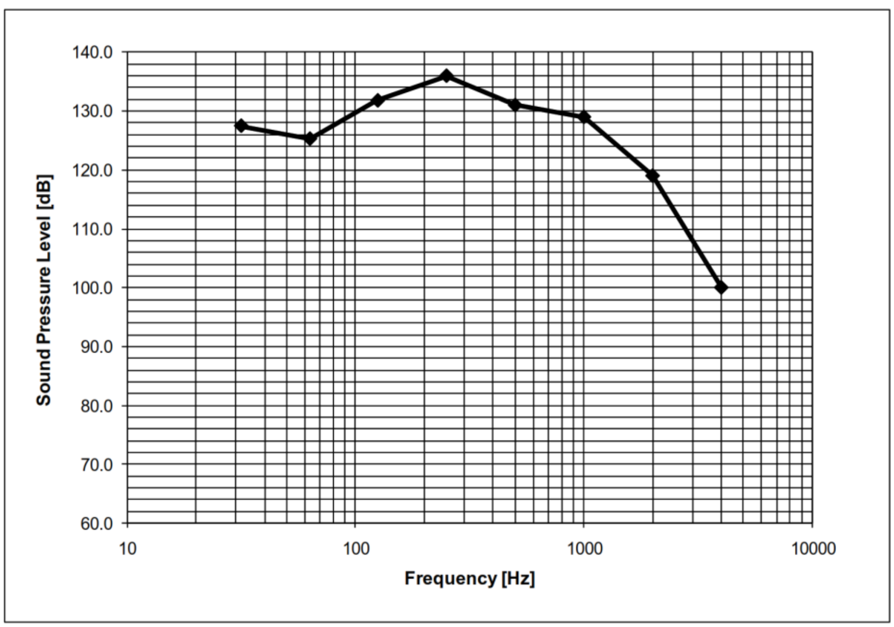


FIGURE 10-1 ACOUSTIC ENVIRONMENT FOR LAUNCH

10.0.6 Shock Environment – End items packed in the foam/bubble wrap materials do not experience significant mechanical shock.

10.0.7 General Atmosphere – See Table 10-5

TABLE 10-5. ISS ENVIRONMENTAL CONDITIONS SUMMARY

Environmental Condition	Value
Atmospheric Conditions on ISS	
Pressure Extremes	0 to 104.8 kPa (0 to 15.2 psia)
Normal operating pressure	14.7 psia
Oxygen partial pressure	3.1 psia
Nitrogen partial pressure	11.6 psia
Dewpoint	4.4 to 15.6°C (40 to 60°F)
Percent relative humidity	25 to 70%
Carbon dioxide partial pressure during normal	24–hr average exposure 5.3 mm Mercury (Hg)
Carbon dioxide partial pressure during crew changeout with 11 crew members plus animals	24–hr average exposure 7.6 mm Hg Peak exposure 10 mm Hg
Cabin air temperature in USL, JEM, and	16.7 to 28.3°C (62 to 83°F)
Cabin air temperature in Node 1	16.7 to 31.1°C (62 to 88°F)
Air velocity (Nominal)	0.051 to 0.203 m/s (10 to 40 ft/min)
Airborne microbes	Less than 1000 CFU/m3
Atmosphere particulate level	Average less than 100,000 particles/ft3 for particles less
Thermal Conditions	
USL module wall temperature	13°C to 43°C (55°F to 109°F)
JEM module wall temperature	13°C to 45°C (55°F to 113°F)
Columbus module wall temperature	13°C to 43°C (55°F to 109°F)
Other integrated racks	Front surface less than 37°C (98.6°F)
General Illumination	108 Lux (10 foot-candles) measured 30 inches from the

10.0.8 Pressure Loading – Integrated end items must maintain positive margins of safety when pressure effects (operational and pressurization/depressurization environments).

Integrated end items shall maintain positive margins of safety and not induce a hazard during or after exposure to a maximum pressure environment of 129.3 kilopascals (kPa) (18.8 pounds per square inch absolute (psia)) and a minimum pressure environment of 0.0 psia. These values are an envelope of the launch vehicle pressure design environments.

Integrated end items shall maintain positive margins of safety when exposed to a depressurization rate of 13.3 kPa/second (116 pounds per square inch (psi)/minute).

10.0.9 Temperature – Integrated end items shall meet all performance and safety requirements after being exposed to temperatures ranging from 0 to +50°C (32 to 122°F). This includes the SlingShot external environment when mounted in the Cygnus PCBM.

Integrated end items shall meet all performance requirements when exposed to the ISS atmosphere temperatures ranging from 5 C to 45°C (41 to 113°F).

10.0.10 Humidity - Integrated end items shall operate properly after being exposed to a ground processing and launch atmosphere ranging from -34°C (-29.2°F) dewpoint to 90% relative humidity (at 20°C (68°F)).

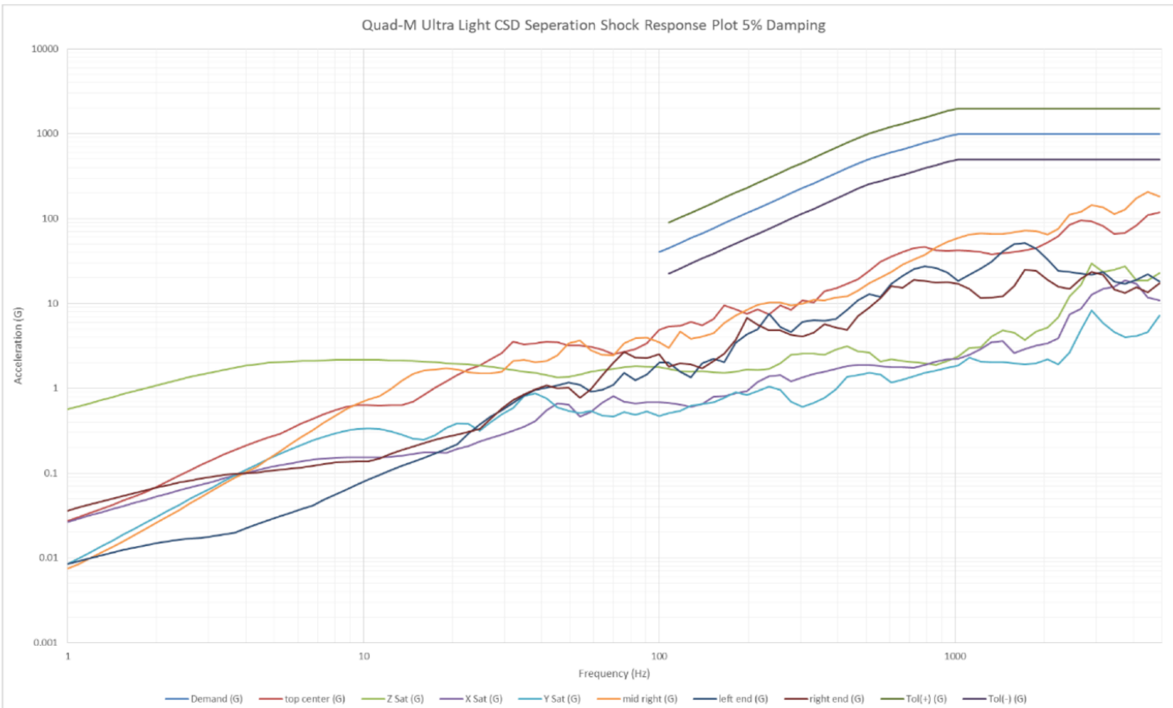
Integrated end items shall be designed to not cause condensation when exposed to the ISS atmosphere ranging in dewpoint from 4.4 to 15.6°C (40 to 60°F) and in relative humidity from 25 to 70% for air (21% oxygen, 79% nitrogen) at one atmosphere pressure (14.7 psia).

10.0.11 On-Orbit Loading Environments - Integrated end items must be designed to accommodate the expected on-orbit environments for the potential launch vehicles on which the end item may launch to the ISS, as well as the ISS on- orbit environment. The on-orbit environment begins with the final stage separation. The on-orbit environment ends with SlingShot deployment.

Integrated end items shall provide positive margins of safety for on–orbit loads of 0.2 g acting in any direction for nominal on-orbit operations.

For launch vehicle berthing to ISS integrated end items shall meet structural integrity requirements while experiencing a one-time peak transient acceleration of 0.4 g acting in any direction due to vehicle berthing to the ISS.

10.0.12 Deployment Shock - Satellites will be exposed to the following deployment shock spectra from the SlingShot Deployer.



SSCSD Maximum Imparted Shock Time Domain Measurement Data

11. Revision History:

Revision	Release Date	Created By	Approved By
-	10-18-18	TH	

Revision Matrix

Revision	Paragraph Number	Change
-	-	INITIAL RELEASE