Structures and Mechanisms

Learning Objectives

- understand role of structure subsystem in context of spacecraft as a whole and between other subsystems
- familiarize with typical requirements and configurations
- differentiate phases of structures development for verification and validation
- conduct structural analysis (load and vibe)

Suggested Reading

Technology for Small Spacecraft, Chapter 5: Spacecraft Structures and Materials

4.1 Definition

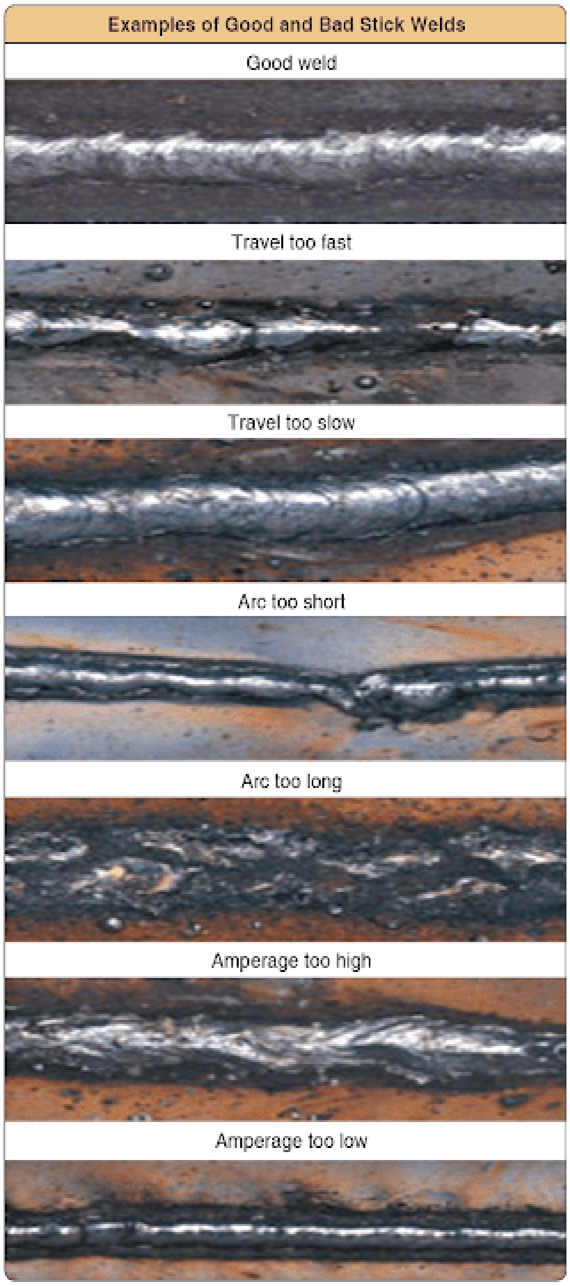
The structures and mechanisms subsystem is responsible for all the mechanical interfaces of the spacecraft throughout its lifetime:

- Adhere to mass constraints set out by form factor and launch provider (during design)
- Mechanically support all other spacecraft subsystems (during integration)
- Attaches the spacecraft to the launch vehicle (during launch)
- Provides for ordnance-activated separation (during orbit insertion)
- Provides deployment and other moving mechanisms (during mission)
- Shielding from space environment, like radiation, atmosphere, pyrotechnic shocks (during mission)

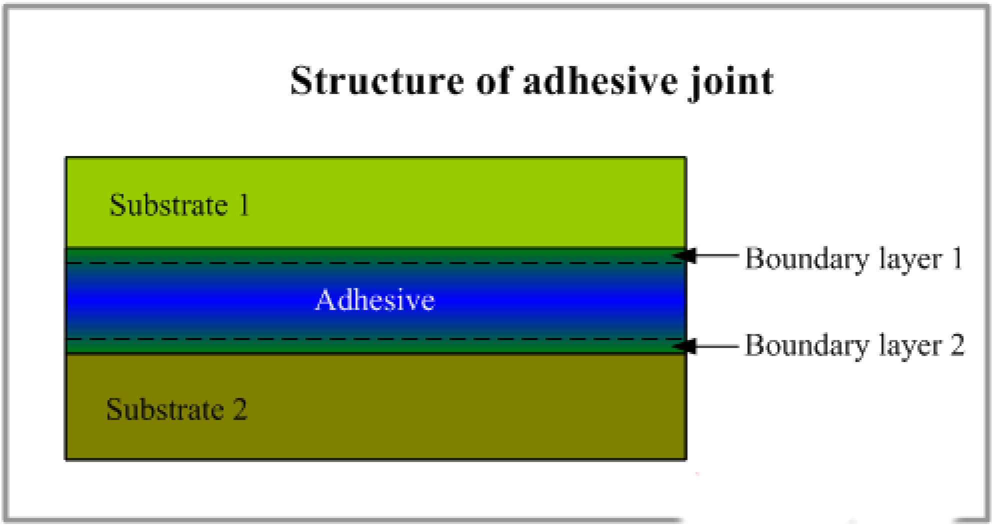
The mechanical design must satisfy all strength and stiffness requirements of the spacecraft and launch vehicle, such as deflection during thermal expansion in space and vibration tolerance during launch. The structures and mechanisms subsystem may be separated into a primary structure and a secondary structure: the primary structure carries the spacecraft's major loads and the secondary structure supports components and provides non-load bearing covers, etc.

Structural components include skin panel assemblies, trusses, ring frames, pressure vessels, fittings, brackets, equipment boxes, and much more. Structural interfaces include attachments, joining options, and fittings. Attachment or joining options in adhesive bonds, welds, and mechanical fasteners. Adhesive bond strength depends on the process and workmanship and requires strict process control and testing. Welds are possible for most aluminum alloys but heat from welds can lower material strength near welds by more than 50%.

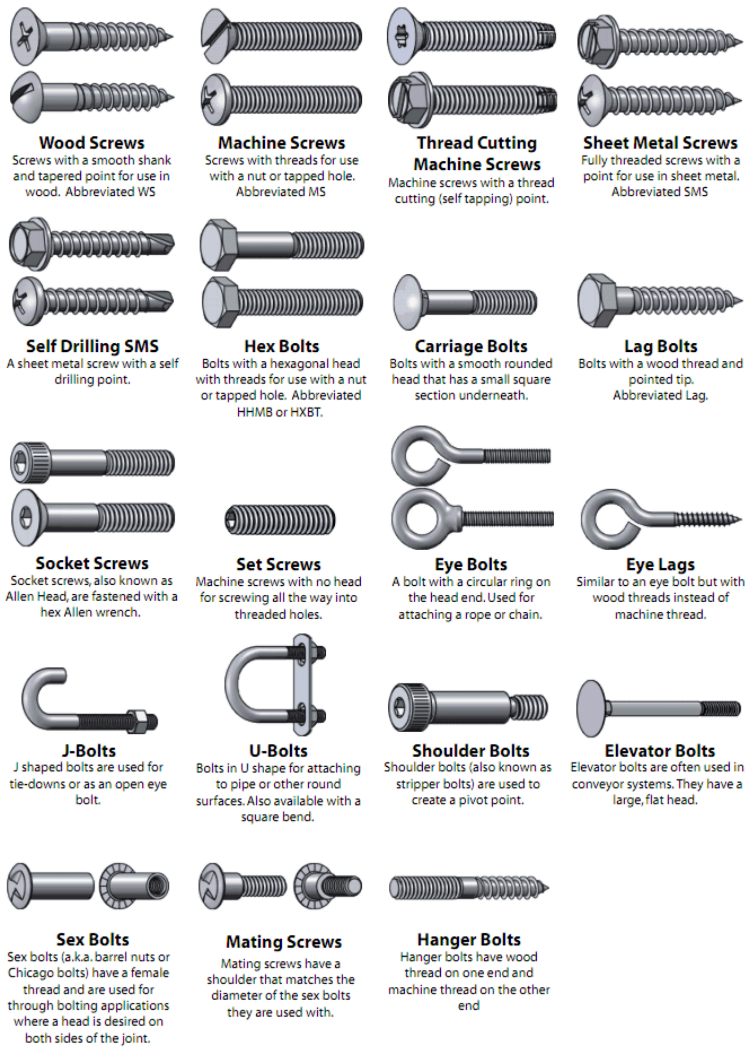
Welds require strict process control and testing. Mechanical fasteners, such as bolts and rivets, can experience fatigue. The Artemis CubeSat kit acknowledges the extensive testing the kit may undergo during your development process so replaceable inserts are embedded within the structure. Most connections within the kit are fastened by bolts. Most composite material structures have metal end fittings or edge members attached by bonding. Bolts can be used but local stress concentrations around fasteners can cause premature failure at lower load levels.



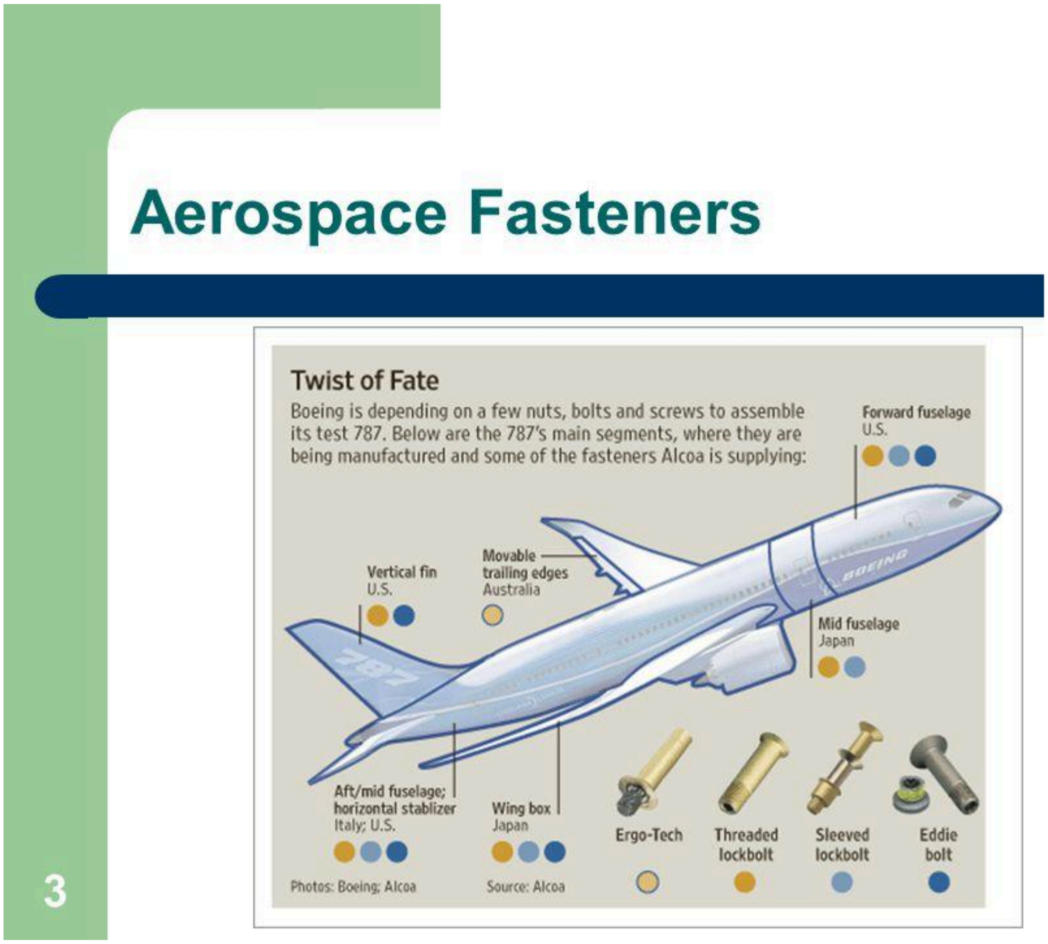
[Arc Welding Procedures, Techniques, and Welding Safety Equipment - Aircraft Welding]



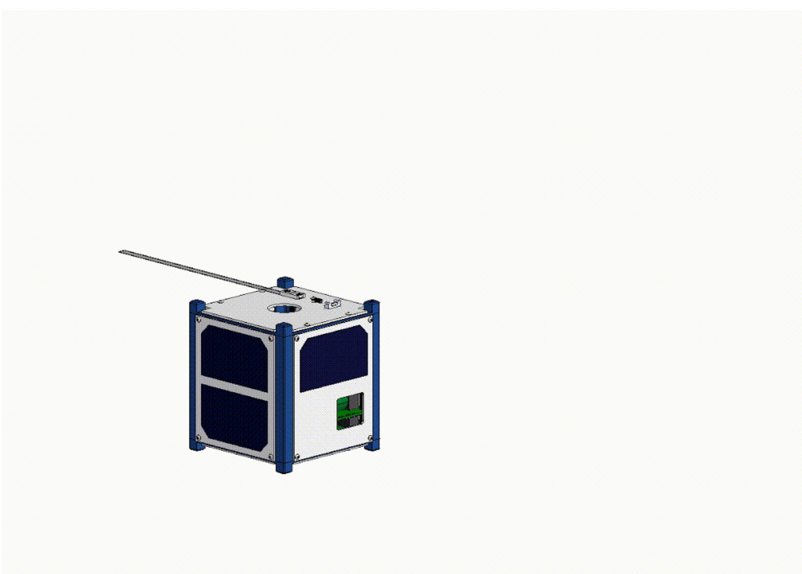
[Fundamentals of adhesive bonding by Dr. Dmitri Kopeliovich]



[Types of Screws. A variety of fasteners for different jobs.]

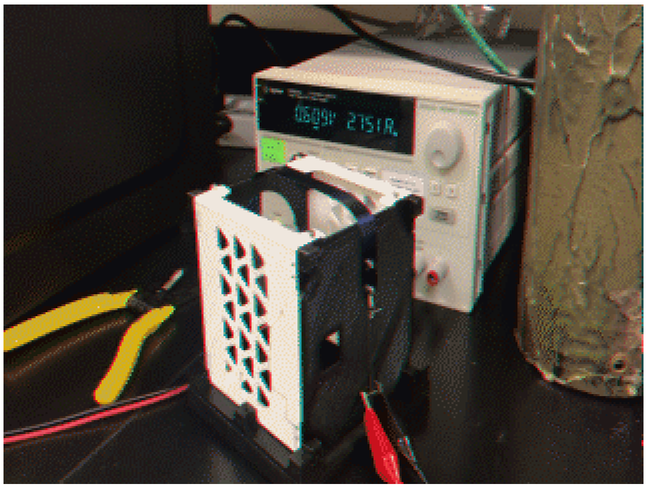


[Aerospace Fasteners Application: An Aerospace Manufacturing Perspective.]



3D Exploded Image of cubesat Created by Kevin Williamswith HFSL

In the Artemis CubeSat kit, the structures and mechanisms subsystem consists of the 1U cubesat frame, an antenna deployment mechanism, a threaded rod, spacers, and various fasteners. The structural frame is made out of aluminum 6061 T6 and is hard anodized. When stowed, the antenna deployment mechanism relies on a spring loaded antenna, fastened by a fishing line. Upon deployment, the fishing line is burned through by a nichrome wire, releasing the antenna. The threaded rod is made of super-corrosion-resistant 316 stainless steel. The various length spacers are made of aluminum. The inserts are made of stainless steel. The stainless steel fasteners include slotted screws, hex nuts, and washers.



Video of Antenna Deployment by Kevin Williams with HSFL

The custom structure's CAD can be found in the Artemis repository but the list of parts for the remaining structural components are as follows:

Manufacturer	Part name	Part Description	Part Number	Pack of	Qty	Unit Cost	Subtotal	General Notes
3D Hubs	Top Frame	Al 6061 T6, Hard Anodized	--	1	1	$44.70	$44.70	Price is subject to change
3D Hubs	Bottom Frame	Al 6061 T6, Hard Anodized	--	1	1	$44.50	$44.50	Price is subject to change
3D Hubs	Side Frame	Al 6061 T6, Hard Anodized	--	1	2	$29.90	$59.80	Price is subject to change
McMaster-Carr	316 Stainless Steel Hex Drive Flat Head Screw	90 Degree Countersink, M2 x 0.40mm Thread, 5mm Long	93395A137	25	1	$8.13	$8.13	Structure/frame screws
McMaster-Carr	Stainless Steel Helical Insert	M2 x 0.4 Thread Size, 4 mm Long	91732A183	10	2	$13.75	$27.50	Inserts for structure holes
McMaster-Carr	Super-Corrosion-Resistant 316 Stainless Steel Threaded Rod	M3 x 0.5 mm Thread Size, 1 M Long	94185A140	1	1	$10.68	$10.68	Uprights for the boards

McMaster-Carr	316 Stainless Steel Nylon-Insert Locknut	Super-Corrosion-Resistant, M3 x 0.5 mm Thread	94205A220	50	1	$4.75	$4.75	Fastner nut for the rods/uprights
McMaster-Carr	18-8 Stainless Steel Unthreaded Spacer	4.500 mm OD, 10 mm Long, for M3 Screw Size	92871A181	1	10	$1.52	$15.20	Spacers for boards
McMaster-Carr	" "	4.500 mm OD, 16 mm Long, for M3 Screw Size	92871A187	1	10	$1.88	$18.80	Spacers for boards
McMaster-Carr	" "	4.500 mm OD, 17 mm Long, for M3 Screw Size	92871A306	1	5	$1.49	$7.45	Spacers for boards
Digi-Key	D2MQ-4L-105-1	SWITCH SNAP ACTION SPDT 50MA 30V	D2MQ-4L-105-1-ND	1	4	$6.52	$26.08	Deployment switch
McMaster-Carr	18-8 Stainless Steel Slotted Screws	M1.4 x 0.3mm Thread, 6mm Long	91800A036	10	1	$9.99	$9.99	Deployment switch fastener
McMaster-Carr	18-8 Stainless Steel Washer	M1.4 Screw Size, 1.5 mm ID, 3.8 mm OD	93475A179	100	1	$3.79	$3.79	Deployment switch fastener
McMaster-Carr	18-8 Stainless Steel Hex Nut	M1.4 x 0.3 mm Thread	91828A005	1	8	$2.50	$20.00	Deployment switch fastener
McMaster-Carr	Wear-Resistant 1095 Spring Steel	0.0040" Thick, 1/2" Wide, 25 Feet Long	9075K4	1	1	$31.06	$31.06	Antenna material
McMaster-Carr	High-Strength High-Temperature PEEK Screw	Pan Head Phillips, 6-32 Thread, 3/8" Long	96367A451	1	2	$5.34	$10.68	Antenna material fastener
McMaster-Carr	High-Temperature PEEK Plastic Washer	Number 6 Screw Size, 0.15" ID, 0.38" OD	93785A300	10	1	$4.39	$4.39	Antenna material fastener
McMaster-Carr	High-Strength High-Temperature PEEK Hex Nut	6-32 Thread Size	98886A100	1	2	$4.26	$8.52	Antenna material fastener
Total						**$356.02**		

4.2 Subsystem Responsibilities

The structures and mechanisms system is responsible for:

- surviving loads through launch and the entire spacecraft mission.
- supporting geometric requirements or constraints in component placement.
- deploying structures successfully.
- ensuring mechanisms do not fail throughout the entire spacecraft mission.

During the design process, the structures and mechanisms (STR) specialist:

- is the owner and primary developer of the spacecraft Computer-Aided Design (CAD). The Artemis CubeSat Kit is supported by Solidworks and OnShape.

- must design the spacecraft frame and ensure the primary structure, the spacecraft frame, survives the spacecraft's major loads, which includes the launch vehicle conditions while contained in the deployer and during the pod deployment.
- must also generate secondary structures that support components, like mounting brackets, baffles, and camera covers.
- must geometrically fit the various subsystem components into the spacecraft structure, which may be an iterative, time-consuming process.
- is responsible for adhering to the structural requirements throughout the entire spacecraft lifecycle through analysis and testing.

4.3 Typical Requirements and Design Considerations

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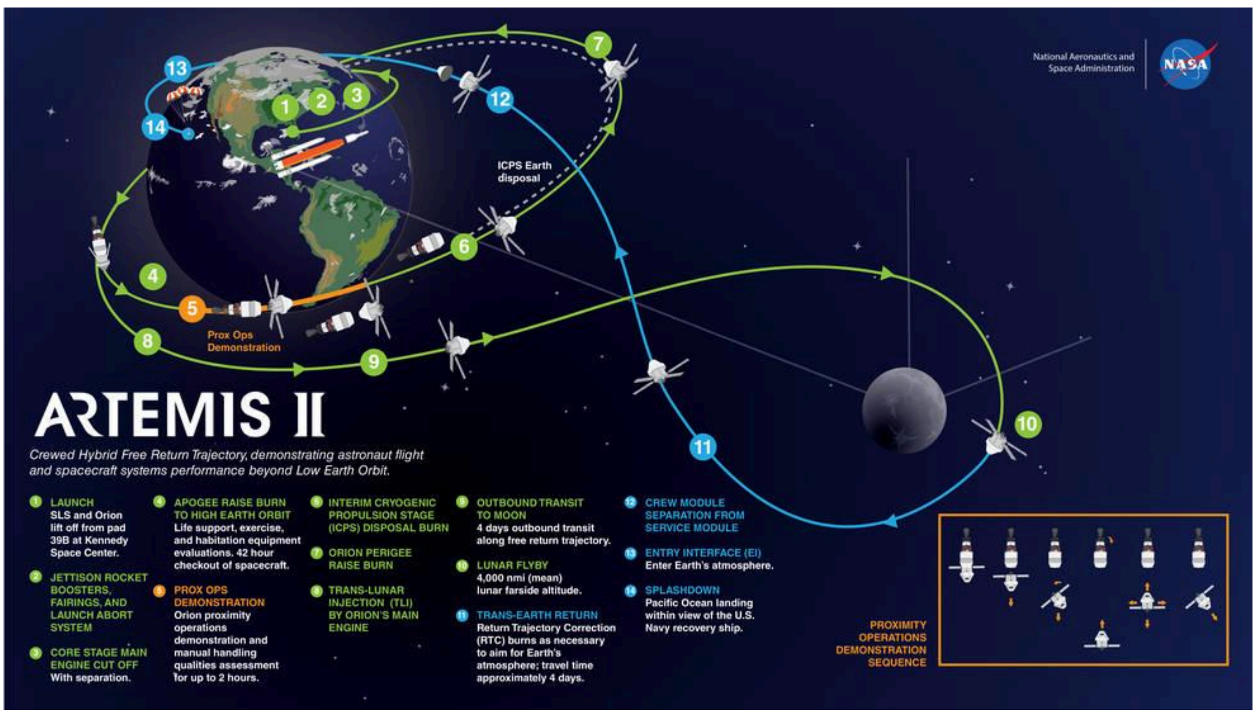
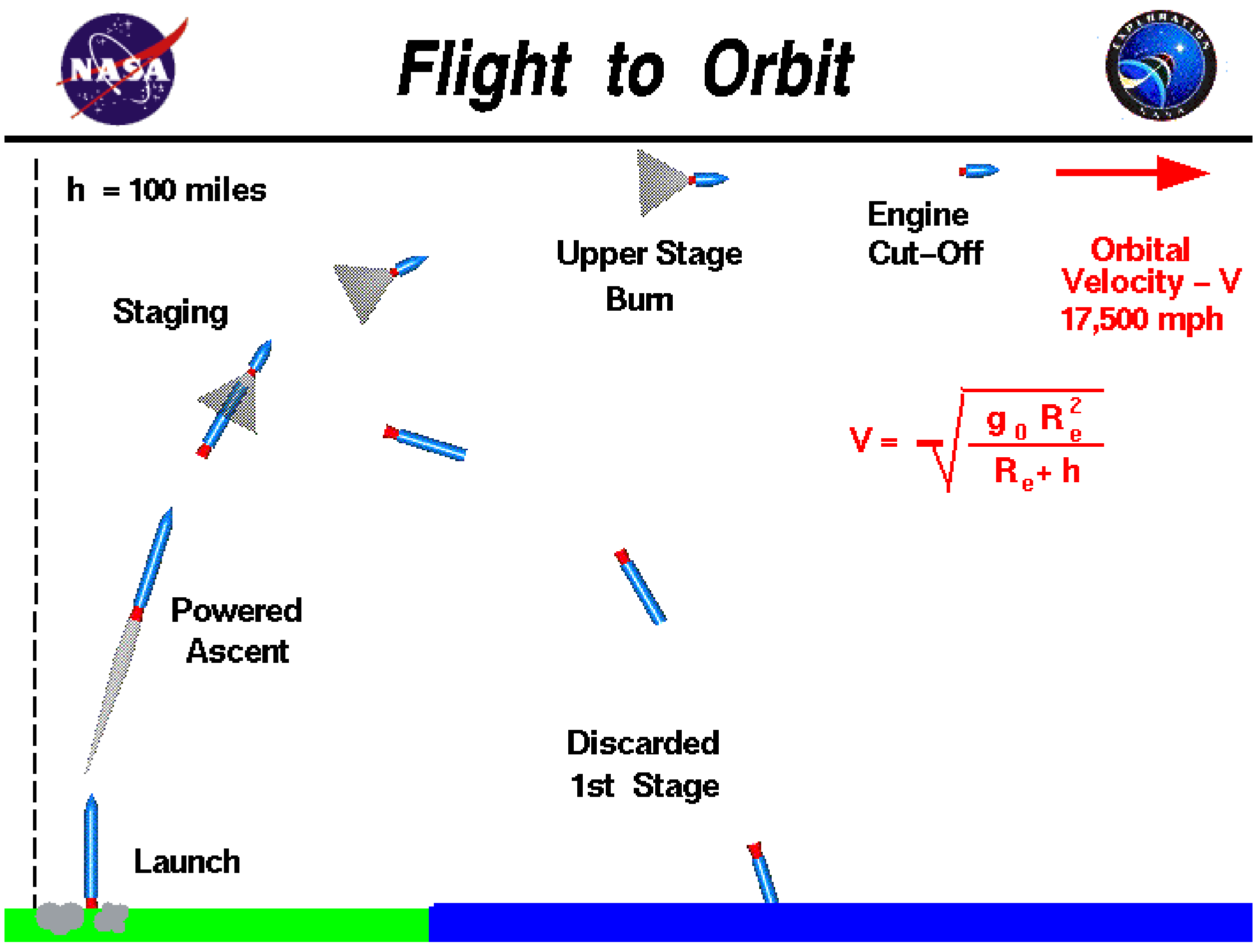


[The Artemis II mission map shows the planned flight path and test objectives for the flight. Image courtesy of NASA.]



Flight to orbit. The life cycle of a rocket. Image courtesy of NASA.

Sources for requirements may be generated from external constraints or internally generated needs from all parts of the spacecraft lifecycle, from manufacturing to spaceflight operations. The basic requirements or design drivers for any subsystem are the allocated size, weight, and power, which applies for structures and mechanisms. Requirements specific to the structures and mechanisms system related to the spacecraft mission include:

- components requiring a certain orientation within the spacecraft frame (facing away from the spacecraft center)
- components requiring a certain placement within the spacecraft frame (radiation shielding or protection)
- observing payloads that need an unobstructed view into the space environment (most optics)
- accommodation of specific size within spacecraft volume and weight distribution affecting moment of inertia
- required active mechanisms or deployables (like a extendable boom) to achieve mission objectives
- total mass and size of the spacecraft
- mechanical interfaces between spacecraft components and primary structure

Sources of internal requirements during manufacturing and assembly could include handling fixtures, container interfaces, or stresses induced by manufacturing processes. The Artemis CubeSat Kit does not have handling fixtures, has rails to interface with the P-POD deployer, and inserts to deal with manufacturing fatigue from assembly and disassembly.

During transport and handling, requirements may include crane or dolly interfacing (especially for large spacecraft) and considerations for land/sea/air transport environments (like shipping containers for freight boats or trucks). The Artemis CubeSat Kit does not have any handling interfaces but will arrive fitted snugly in fitted foam within a Pelican case, which offers protection during shipment.

During testing, external requirements from the launch provider commonly include environmental testing from vibration or acoustic profiles. These tests may require a fixture to the testbed that also must withstand vibration loads. During pre-launch, requirements could include handling during stacking sequence and pre-flight checks. Commonly, 1U cubesats are stacked into a deployer with other units of spacecraft (for Nanoracks six units in a silo), so our structural design offsets the mating face to avoid any physical contact and potential damage.

During launch and ascent, the structure must withstand steady-state booster accelerations, vibro-acoustic noise during launch and transonic phase, propulsion system engine vibrations, pyrotechnic shock from separation events, transient loads during stage separations, etc. Generally, spacecraft are designed to launch loads as these loads are the most intense out of any phase. The Artemis CubeSat Kit has been tested on a vibration table to withstand these loads per the Launch Services Program Level Dispenser and CubeSat Requirements Document [NASA LSP-REQ-317.01] and NanoRacks External CubeSat Deployer (NRCSD-E) Interface Definition Document (IDD) [NR-NRCSD-S0004]. The CubeSat may be soft-stowed on a resupply mission to the ISS or hard-stowed as a secondary payload. We've tested both profiles.

During mission operations, the spacecraft structure must withstand thruster acceleration, transient loads from pointing maneuvers, docking events, pyrotechnic shock from separation or deployment, and thermal expansion. The Artemis CubeSat Kit experiences antenna deployment and thermal expansion for which both the thermal vacuum chamber and antenna deployment tests verified survival.

In the final phase of reentry and landing, spacecraft may experience aerodynamic heating and transient winds or landing loads. These phenomena are particularly relevant for the astronaut return or for Mars entry, descent, and landing operations. The Artemis CubeSat Kit need not survive reentry as it is designed to burn up upon reentry.

Artemis Structures Requirements

From the CubeSat Design Specification Rev. 14, the CubeSat dimensions and features are outlined in the CubeSat Specification Drawings (Appendix B). Note: The CubeSat Inspection and Fit-check Procedure (CIFP) can be used to aid in verifying that the CubeSat meets the dimensional requirements specified in Appendix B. The CIFP can be found on cubesat.org. These requirements are applicable for all dispensers not utilizing the tab constraint method. CubeSats designed with tabs can find those specific requirements at the PSC website. Within the CubeSat Design Specification Rev. 14, the structural requirements fall under section 2.2 CubeSat Mechanical Specifications, which start on page 10. The most stringent requirements imposed on the Artemis CubeSat kit are from the NanoRacks External CubeSat Deployer (NRCSD-E) Interface Definition Document (IDD) [NR-NRCSD-S0004]. Please refer to section 4.1 Structural and Mechanical Systems Interface Requirements. The following table lists the Artemis CubeSat Kit's structural requirements, drawn from both the CalPoly and Nanoracks documents.

3.6	The CubeSat structure shall be contained within 1U and offer flexibility in mounting components internally	
	3.6.1	The CubeSat kit structure shall remain inside a 10 x 10 x 11.35 cm +/- 0.1mm volume while undeployed
	3.6.2	All four protruding corners on the top and bottom of the main body of the CubeSat shall not exceed a height of 6.75mm, shall have a minimum length and width of 6mm, and shall have a surface area of 6.5mm x 6.5mm, per NASA CLSI requirements
	3.6.3	There shall be a minimum of 20mm from the CubeSat surface to the top of the corners along the Z direction per NASA CSLI Requirements
	3.6.4	The four edges of the CubeSat along the Z direction shall have a hardness greater than or equal to Rockwell C 65-70 per NASA CSLI Requirements
	3.6.5	The overall structure shall withstand 1200N between two XY planes applied in the Z direction, per NASA CSLI Requirements
	3.6.6	The maximum mass of the entire CubeSat Kit shall not exceed 1.33 kg per NASA CSLI Requirements
	3.6.7	The center of gravity shall be within 2cm of its geometric center relative to the Z direction, per NASA CSLI Requirements
	3.6.3	The CubeSat kit shall be easy to assemble with the provided instructions

Suggested Activity

What kind of structural requirements must you impose on your system to fulfill your science mission?

Requirements Compliance Matrix

In a NASA Technical Standard to establish NASA structural design and test factors [NASA-STD-5001], this document's appendix provides a listing of requirements for selection and verification of requirements by programs and projects. You may use the entire appendix table to decide which requirements apply to your program and by entering "Yes" to describe the requirement's applicability to the program or project; or entering "No" if the intent is to tailor, and enter how tailoring is to be applied in the "Rationale" column. For all the requirements that you've deemed applicable, you should read the corresponding sections in NASA-STD-5001. The figure below is just a snapshot of the 8 pages of potential requirements.

NASA-STD-5001B w/CHANGE 1				
Section	**Description**	**Requirement in this NASA Technical Standard**	**Applicable (Yes or No)**	**If No, Enter Rationale**
	Approaches			
4.1.2.1a	Test Methods	[FSR 11] The strength verification program shall be approved by the responsible Technical Authority.		
4.1.2.1b	Test Methods	[FSR 12] The magnitude of the static test loads shall be equivalent to limit loads multiplied by the qualification, acceptance, or proof test factor.		
4.1.2.1c	Test Methods	[FSR 13] Strength model verification, if required, shall be accomplished over the entire load range.		
4.1.2.1d	Test Methods	[FSR 14] The test article shall be instrumented to provide sufficient test data for correlation with the strength model.		
4.1.2.1e	Test Methods	[FSR 15] Each habitable module, propellant tank, and SRM case shall be proof pressure tested.		
4.1.2.1f	Test Methods	[FSR 16] Departures from test plans and procedures, including failures that occur during testing or are uncovered as part of post-test inspection, shall be documented by a non-conformance report per the approved quality assurance plan.		
4.1.2.2a	Test versus Design Factors of Safety	[FSR 17] When using the prototype structural verification approach, metallic structures shall be verified to have no detrimental yielding at yield design load before testing to full qualification load levels.		
4.1.2.2b	Test versus Design Factors of Safety	[FSR 18] When using the protoflight structural verification approach, design factors shall be specified to prevent detrimental yielding of the metallic structure or damage to the composite/bonded flight structure during test.		
4.1.2.3a	Test versus No-Test Options	[FSR 19] Analysis shall be provided with an acceptable engineering rationale for the “no-test” option.		
4.1.2.3b	Test versus No-Test Options	[FSR 20] To use the “no-test” approach, project-specific criteria and rationale shall be developed for review and approval by the responsible Technical Authority.		
4.1.3	Probabilistic Methods	[FSR 21] Any proposed use of probabilistic criteria to supplement or as an alternative to deterministic factors of safety shall be approved by the responsible Technical Authority on an individual-case basis.		
4.2a	Design and Test Factors of Safety	[FSR 22] The design factors of safety and test factors of this NASA Technical Standard are the minimum required values for NASA spaceflight structures and shall be applied to the limit stress condition, including additive thermal or pressure stresses.		
4.2b	Design and Test Factors of Safety	[FSR 23] If pressure or temperature has a relieving or stabilizing effect on the mode of failure, then for analysis or test of that failure mode, the unfactored stresses induced by temperature or the minimum expected pressure shall be used in conjunction with the factored stresses from all other loads.		
4.2c	Design and Test Factors of Safety	[FSR 24] Material selection and derivation of material design allowables shall follow the requirements defined in NASA-STD-6016Standard Materials and Processes Requirements for Spacecraft.		
4.2d	Design and Test Factors of Safety	[FSR 25] The factored stresses shall not exceed material allowable stresses (yield and ultimate) under the expected temperature, pressure, and other operating conditions.		
4.2e	Design and Test Factors of Safety	[FSR 26] The hardware shall be designed to preclude any detrimental yielding under limit loads and, where applicable, under protoflight or proof test loads.		
4.2f	Design and Test	[FSR 27] Applications of design and test factors to the development and verification of a structure shall be		

4.4 General Arrangement and Design Drivers

This section will review the design drivers that affect the structural design of the spacecraft: geometry, mass, structural loads, materials, and processes. These design drivers affect each other, sometimes beneficially, but also in ways that oppose each other and cause problems. The design process is always iterative and must consider all of these design drivers

“The appropriate design and test factors for a given mechanical or structural flight hardware element depend on several parameters, such as the materials used, attachment methods (e.g., bonding), and the verification approach (prototype or protoflight). In addition to the minimum factors of safety specified in this NASA Technical Standard, some structural and mechanical members may be required to meet other more stringent and restrictive performance requirements, such as dimensional stability, pointing accuracy, stiffness/frequency constraints, or safety requirements (e.g., fracture control)” [NASA-STD-5001B].

Geometry

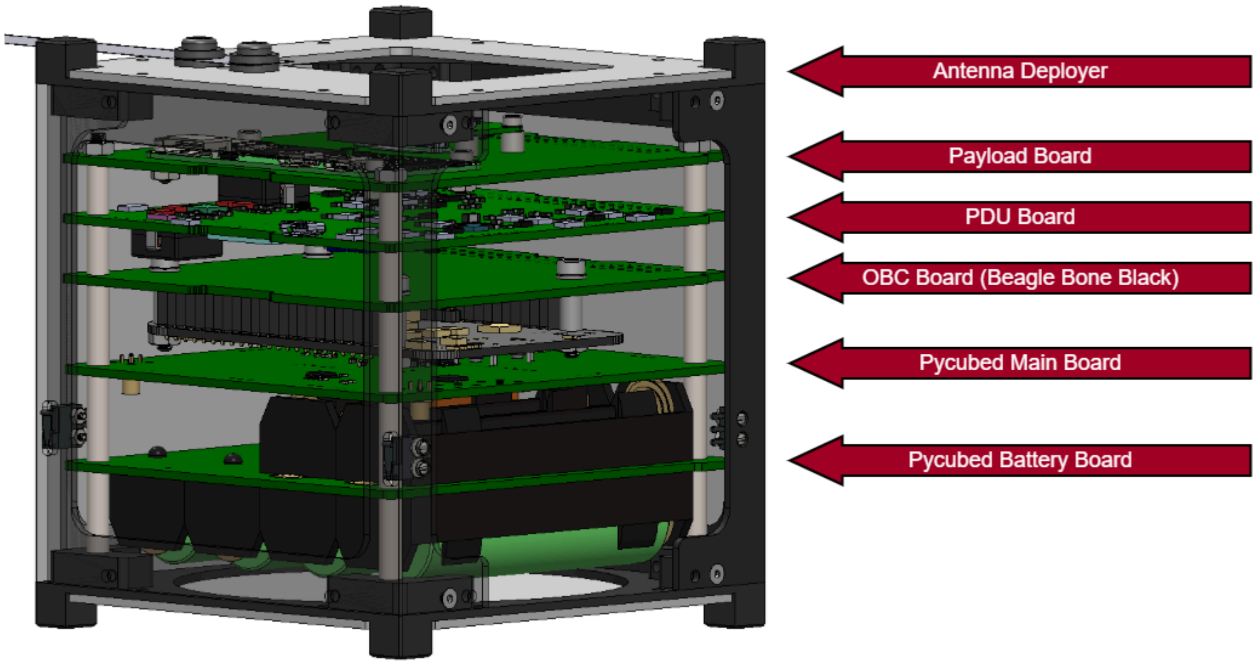
For the primary structure, the design drivers are the first and foremost the requirements, derived from external constraints and internal needs. The primary structure is constrained to the launch vehicle fairing or deployer enclosure and must enclose all the spacecraft bus components. Geometric considerations that affect every subsystem could include:

Subsystem	Consideration
Payload	• Unoccluded field of view for payloads • Thermal deformation in critical payload components • Electromagnetic Interference/Capability
Structure and Mechanisms	• Spatial organization to ensure components do not intersect • Vibration isolation
Thermal Control	• Facilitate thermal management through conduction or radiation • Regulating thermally sensitive components through placement • Enabling thermal isolation
Power (including harness)	• Securing harnessing in empty spaces • Insulating battery (which is typically thermally sensitive)
Telemetry and Control	• Unoccluded field of view for antenna
Command and Data Handling	• Radiation shielding by placing computer behind other components
Attitude Determination and Control	• Sensor and actuator mounting in defined orientations • Regulating moments of inertia (emphasize one principal axis or uniform across axes) • Minimize off diagonal moments of inertia
Propulsion	• Direction of thrusters • Placement of propellant exhaust exit with respect to payload optics

Artemis CubeSat Kit Arrangement

The Artemis CubeSat kit is straightforward in its geometric arrangement as the subsystem components are rather homogenous in their geometry; all the subsystems are generally mounted on PCB boards and stacked on a threaded rod as seen in Fig. __. The antenna and deployer on the exterior of the satellite to guarantee an unoccluded field of view. The exterior skin has a cut out to allow a camera to peer through; the first board in the stack has the payload and payload supporting electronics. The power distribution board follows. The onboard computer sits nearly in the center of the cubesat, providing radiation shielding.

Between boards starting from payload to onboard computer, 104-pin Cubesat kit bus headers are used to reduce clutter. Pycubed boards were not designed to be compatible with the cubesat kit bus header, and thus integrated through external connections. The low level control computer, called the PyCubed, is stacked underneath. Finally, we have the PyCubed battery board, which runs warm and needs exposure to the space environment to radiate its heat.



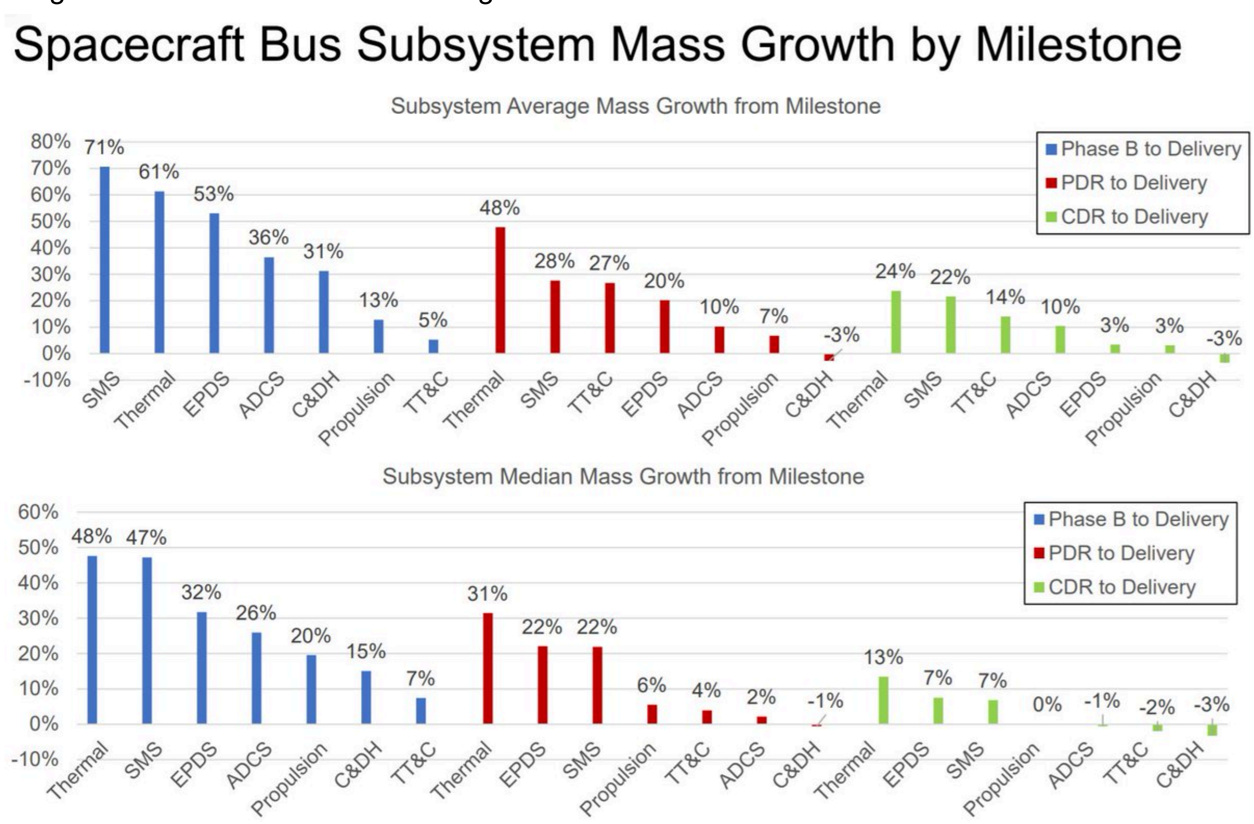
Mass

The structure and mechanisms subsystem specialist is an important player in generating and managing the mass budget, with the assistance of the systems engineer and the other subsystem specialists. For a 1U cubesat, the total mass of the spacecraft must not exceed 2 kg [CubeSat Design Specification Rev. 14]. Any additional mass may be negotiated with the launch provider with an immense amount of paperwork and persistence but it's not impossible. Typically, 1U cubesats are between 1 kg and 1.33 kg. To reiterate, a suggested mass budget and specific 1U cubesat project mass budgets are as follows:

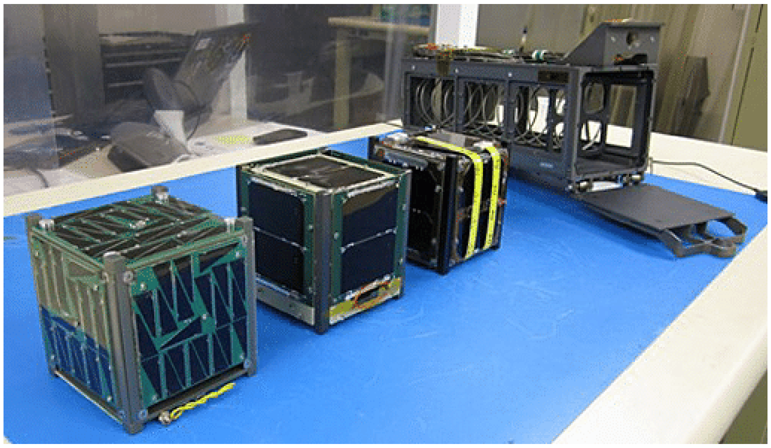
Subsystem (% of Dry Mass)	SMAD suggestion	Hermes CubeSat	OreSat	Artemis CubeSat	Ke Ao (variant of Artemis)
Payload	41%	Allocated in T&C			
Structure and Mechanisms	20%	32.3%			
Thermal Control	2%	0%			

Power (including harness)	19%	13.5%			
Telemetry and Control	2%	22.5%			
Command and Data Handling	5%	3.6%			
Attitude Determination and Control	8%	2.4%			
Other (balance + launch)	3%	25.7%			
Total	100%	100%			

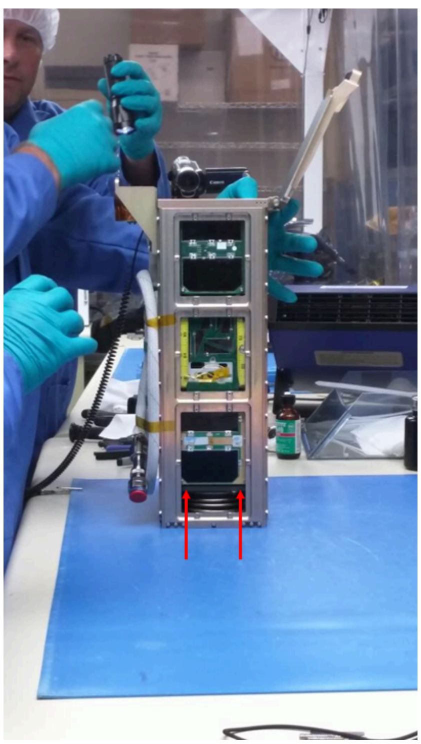
The mass budget typically carries margin at the preliminary design phase. The design margin decreases over time as the design converges to the final assembly. Refining the design toward spaceflight reveals additional interfacing and detailing that inevitably adds mass to the system [Hayhurst et al.]. Each subsystem's mass growth by design gate is shown in Fig. __. Note, the spacecraft studied are traditional in size and mass, which means that the study was not geared toward cube satellite design.



Structural Loads



Three 1U CubeSats beside a 3U (Poly Picosatellite Orbital Deployer (PPOD) developed at CalPoly. The spring mechanism used by P-PODs to deploy CubeSats can be seen within the main housing, prior to loading. Image Credit: California Polytechnic State University
Source publication



The Fox-1A CubeSat satellite has been integrated into the Poly-PicoSatellite Orbital Deployer rig (P-POD) with two other CubeSats. The red arrows show the static load on the cubesats generated from the compressed spring's force.

This section will provide an overview of typical structural loads and how they drive the spacecraft structural design. [Loads](#)are generated by forces, deformations, or accelerations which cause stresses, deformations, and displacements in structures. There are two types of structural loads: static and dynamic. Static loads are steady-state loadings, like loads imparted on spring-loaded deployers, launch acceleration, or pressurized vessels. Think of these loads as built-up loads that are ready to burst or buckle. Dynamic loads are loads from vibrations generated by natural frequencies, like launch vehicles, pyrotechnic separation, or deployment events. Think of these loads as shocking events. Structural engineers are concerned with mitigating the effects of the critical load: the load that the spacecraft most intensely feels and is most likely to break the spacecraft. Critical loads could be launch loads for an assembled spacecraft, pressurization loads for a rocket casing, thermal loads for a propulsion system, centrifugal forces from rapid rotation, or on-orbit collisions.

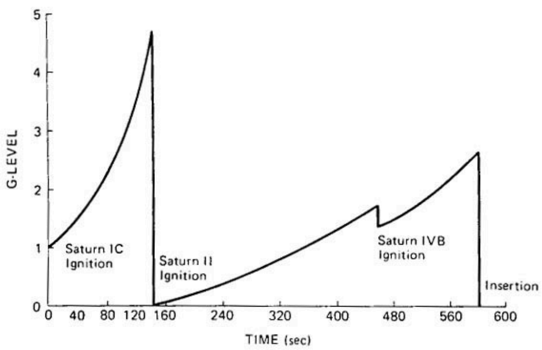
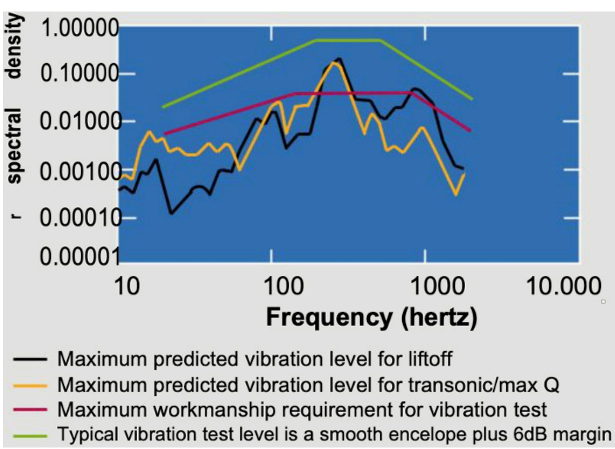


Figure 2. Typical Apollo launch profile – Saturn V launch vehicle.

Apollo 15 Launch and Reaching Earth Orbit. Courtesy of NASA.



Typical vibration spectrum of a launch.How to test satellites and not destroy them by Ben Sampson Courtesy of Aerospace Testing International.

Just as we reviewed every phase of the spacecraft lifecycle in the Typical Requirements section, we will revisit these phases to identify all loads and estimate the load. Load quantification may be obtained through measurements, tests, references, and asking the relevant engineers. Critical load estimation is not always straightforward and may need to be indirectly quantified or estimated.

- during manufacturing and assembly, stresses could include welding, joint stressing due to [tightening bolts]
- during transport and handling, requirements may include loads from transferring the spacecraft to a shipping container (especially for large spacecraft) and shock during transportation through land/sea/air transport environments (like on freight boats or trucks). The Artemis CubeSat Kit expects to be handled delicately by human hands, which yield gentle loads. The kit is not expected to survive being dropped. The kit will arrive fitted snugly in fitted foam within a Pelican case, which mitigates the shock loads during transportation. These loads will not be the critical loads.
- during thermal and vibration testing, critical loads could be the stress from misaligned thermal expansion, launch shock, acceleration loads, and random vibration environments. These tests replicate launch loads and thermal stressing from the mission operations environment. The difference between thermal and vibe testing in this phase with respect to the real environments could be the disparity in replicating the same loads in which case the spacecraft must survive two different loading profiles. Ideally, the test matches the launch and space environment conditions so we will discuss critical loads in those phases. A summary of tests is seen below:

Tests	Qualification by Test	Protoflight Test	Acceptance Test
Random vibration[6] (CubeSat and Dispenser) Ref Mil-Std 1540C	MPE + 6 dB for (3) minutes, each of (3) axes [1]	MPE+3 dB for (2) minutes, each of (3) axes [1]	MPE for (1) minute, each of (3) axes [1]
Sinusoidal Vibration[6] (CubeSat and Dispenser) Ref Mil-Std 1540C	MPE + 6 dB. Testing shall be performed for content that is not covered by random vibration testing	1.25 x MPE. Testing shall be performed for content that is not covered by random vibration testing	MPE. Testing shall be performed for content that is not covered by random vibration testing [1]
Shock[6] (CubeSat and Dispenser) Ref Mil-Std 1540C	MPE + 6 dB, 3 times in both directions of 3 axes [1,3]	MPE + 3 dB, 1 times in both directions of 3 axes [1,3]	N/A
Thermal Vacuum Cycle (Dispenser Only) Ref.: MIL-STD 1540 B, GSFC-STD-7000	MPE[2] +/- 10° C Minimum Range = -14 -3/+0°C to +71 -0/+3°C Cycles = 8 Dwell Time = 1 hour min. @ extreme Temp. after thermal stabilization Transition = < 5° C/minute Vacuum = $1x10^{-4}$ Torr	MPE[2] +/- 10° C Minimum Range = -14 -3/+0°C to +71 -0/+3°C Cycles = 4 Dwell Time = 1 hour min. @ extreme Temp. after thermal stabilization Transition = < 5° C/minute Vacuum = $1x10^{-4}$ Torr	MPE[2] +/- 5° C Minimum Range = -9 -3/+0°C to +66-0/+3°C Cycles = 2 Dwell Time = 1 hour min. @ extreme Temp. after thermal stabilization Transition = < 5° C/minute Vacuum = $1x10^{-4}$ Torr
Thermal Vacuum Bake out (Dispenser Only) Ref.: MIL-STD 1540 B, GSFC-STD-7000	N/A	Min. Temp 70°C [4,7] Cycles = 1 Dwell Time = Min. 3 hour after thermal stabilization Transition = N/A Vacuum = $1x10^{-4}$ Torr	Min. Temp 70°C [4,7] Cycles = 1 Dwell Time = Min. 3 hour after thermal stabilization Transition = N/A Vacuum = $1x10^{-4}$ Torr
Thermal Vac Bake out (CubeSat Only) Ref.: MIL-STD 1540 B, GSFC-STD-7000	N/A	Min. Temp 70°C [5,8] Cycles = 1 Dwell Time = Min. 3 hour after thermal stabilization Transition = < 5° C/minute Vacuum = $1x10^{-4}$ Torr	Min. Temp 70°C [5,8] Cycles = 1 Dwell Time = Min. 3 hour after thermal stabilization Transiton = < 5° C/minute Vacuum = $1x10^{-4}$ Torr
Hardware Configuration	Dispenser – Flight identical unit (includes NEA, cable and connector) **CubeSat** – Flight Identical unit	Dispenser – Flight unit (includes flight NEA, cable and connector) **CubeSat** – Flight unit	Dispenser – Flight unit (includes flight NEA, cable and connector) **CubeSat** – Flight unit

Launch Services Program. Courtesy of NASA.

- during pre-launch, the spacecraft must be handled and packaged into the launch vehicle. For large spacecraft, handling loads could include static loads at hoisting interfaces. Cubesats experience the compression of the P-POD deployer. Drawing from the Nanoracks External CubeSat Deployer Document, "The CubeSat shall be capable of withstanding a force 1320N across all load points equally in the Z direction". The Artemis CubeSat kit was analyzed to withstand this integrated load. This could be a critical load for which we should do structural analysis.
- during launch and ascent, the structure must withstand steady-state booster accelerations, vibro-acoustic noise during launch and transonic phase, propulsion system engine vibrations, pyrotechnic shock from separation events, transient loads during stage separations, etc. Generally, the critical loads to launch loads as these loads are the most intense out of any phase. The Artemis CubeSat Kit has been tested on a vibration table to withstand these loads per the Launch Services Program Level Dispenser and CubeSat Requirements Document [NASA LSP-REQ-317.01] and NanoRacks External CubeSat Deployer (NRCSD-E) Interface Definition Document (IDD) [NR-NRCSD-S0004]. The CubeSat may be soft-stowed on a resupply mission to the ISS or hard-stowed as a secondary payload. We've tested both profiles. For your

convenience, we have listed all relevant loads taken word for word from the Nanoracks document.

- 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 4.3.1-1 at the CG of the item, with all six degrees of freedom acting simultaneously.

Table 4.3.1-1: Launch/Landing Load Factors Envelope

	Nx (g)	Ny (g)	Nz (g)	Rx (rad/sec^2)	Ry (rad/sec^2)	Rz (rad/sec^2)
Launch	+/- 7.0	+/- 4.0	+/- 4.0	+/- 13.5	+/- 13.5	+/- 13.5

- Random Vibration Environment: The CubeSat shall be capable of withstanding the dynamic flight environment for the mission applicable launch vehicle (shown in Table 4.3.2.1-1 through Table 4.3.2.1-4). Nominally, NRCSDE missions are launched on the Antares rocket; however, Atlas V rockets have been utilized in the past.

Table 4.3.2.1-1: In-Plane Random Vibration Test Levels and Duration

Frequency (Hz)	ASD (g2/Hz))
20	0.016
30	0.025
800	0.025
2000	0.016
grms	6.55
Duration	60

Table 4.3.2.1-2: Out-of-Plane Random Vibration Test Levels and Duration

Frequency (Hz)	ASD (g2/Hz)
20	0.016
50	0.05
800	0.05
2000	0.016
grms	8.45
Duration	60

Table 4.3.2.1-3: In-Plane Sine Vibration Test Profiles

Frequency (Hz)	Levels (g's)
5	0.5 inch Double Amplitude (DA)
24	13.8
25	13.8
26	10.8
35	10.8
40	2.4
100	2.4
Sweep Rate	4 octave/minute

Table 4.3.2.1-4: Out-of-Plane Sine Vibration Profiles

Frequency (Hz)	ASD (g^2/Hz)
5	0.5 inch Double Amplitude (DA)
24	13.8
25	13.8
26	10.8
35	10.8
36	6.6
50	6.6
55	2.4
100	2.4
Sweep Rate	4 octave/minute

- Launch shock environment: The CubeSat shall be capable of withstanding the shock environment shown in Table 4.3.3-1. Any mechanical or electrical components on the spacecraft that are highly sensitive to shock should be identified and assessed on a case-by-case basis as defined in the unique payload ICA.

Table 4.3.3-1: Cygnus Shock Spectrum

CubeSat Deployer Shock Spectrum	
Frequency (Hz)	**Protoflight Level (g)**
100	40
500	494
1000	989
10000	989

- during mission operations, loads include thruster acceleration, transient loads from pointing maneuvers, docking events, pyrotechnic shock from separation or deployment, and loads from thermal expansion. The Artemis CubeSat Kit experiences antenna deployment and thermal expansion for which both the thermal vacuum chamber and antenna deployment tests verified survival. For your convenience, we have reiterated the thermal environment taken word for word from the Nanoracks document.
    - The CubeSat shall be capable of withstanding the expected thermal environments for all mission phases, which are enveloped by the on-orbit EVR phase prior to deployment. The expected thermal environments for all phases of the mission leading up to deployment are below in Table 4.3.5-1.

Table 4.3.5-1: Expected Thermal Environments

Mission Phase	Temperature Extremes
Ground Transport (Customer facility to NanoRacks)	Determined for each payload
Ground Processing NanoRacks	Determined for each payload
Ground Processing NASA Envelope	10°C to 35°C
Pre-Deployment **extreme temperatures seen by the deployer**	-14°C to 44°C

Ref SSP 50835, Table E.2.10-1

- in the final phase of reentry and landing, spacecraft may experience aerodynamic heating and pressure, transient winds or landing loads. The Artemis CubeSat Kit need not survive reentry as it is designed to burn up upon reentry.

Upon identifying the various loads, we may conduct structural analysis to determine which of these loads is the critical load. From experience and intuition, a good guess is to design the structural components of the spacecraft to the launch conditions if your spacecraft will remain in orbit. If your spacecraft will re-enter Mars atmosphere, for example, the entry, descent, and landing phase may be more critical. The structural analysis will be described in the last section of this chapter but the consequence of the structural analysis is the iterative design and redesign of the structural components to fulfill sufficient margins of safety.

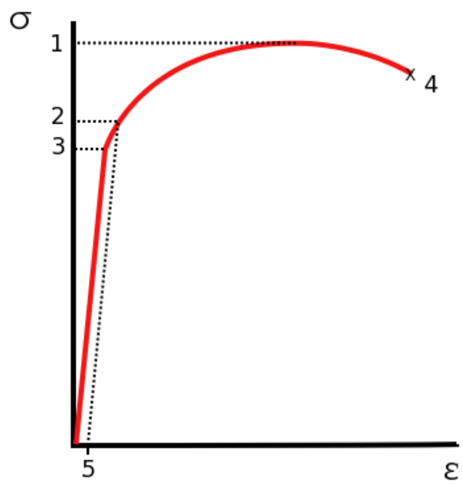
Structural analysis plays into the initial design of structures by conducting back-of-the-envelope (simplified) calculations as to the sizing or thickness of a structural component, like the primary structure wall or supporting bracket. The structural analysis enters at the redesign phase by showing that some structural components fail at the critical load and need reinforcement to achieve mission success. Structural analysis may also show some structural components more than sufficiently carry that piece's critical load and could be trimmed in mass to allocate elsewhere. Finally, structural analysis in the way of finite element analysis is a critical method of verifying that structural designs will survive tests or survive conditions that would otherwise be infeasible to test.

Materials

The selection of the structural material affects the survivable structural load, mass, geometry, and concerns around outgassing. Material properties include density, stiffness, strength, weight, ductility, coefficient of thermal expansion, fatigue, and outgassing:

- Density is the mass per unit volume of a material. As space missions are proportional in cost to the mass launched into space, lower density materials are preferred.
- The precise term for material stiffness is Young's modulus, which "defines the relationship between stress (force per unit area) and strain (proportional deformation) in a material in the linear elasticity regime of a uniaxial deformation. Young's modulus enables the calculation of the change in the dimension of a bar made of an isotropic elastic material under tensile or compressive loads" [Wikipedia]. This value is commonly represented by the letter *E* or *Y*.
    - $E = \frac{\sigma}{\varepsilon}$, where[2]
    - E is Young's modulus
    - \sigma is the uniaxial stress, or uniaxial force per unit surface
    - \varepsilon is the strain, or proportional deformation (change in length divided by original length); it is dimensionless
    - Both *E* and *sigma* have units of pressure, while {\displaystyle \varepsilon }\varepsilon is dimensionless. Young's moduli are typically so large that they are expressed not in pascals but in megapascals (MPa or N/mm2) or gigapascals (GPa or kN/mm2).
- There are two types of material strengths that we care about: yield strength and ultimate strength.
    - Materials, when stressed below the yield point, return to their original form. Imagine a rubber band stretched gently. When you stop stretching the rubber band, the rubber band returns to its original form; this is elastic deformation.

- ○ Yield strength is when a material is stressed to the point where the material does not return to its original shape; this is plastic deformation. Imagine that the rubber band was stretched more intensely and when the stretching is eased, the rubber band looks a bit longer than it started out.
- ○ Ultimate strength is the maximum stress that a material can withstand while being stretched or pulled before breaking. Imagine that this is when the rubber band snaps.

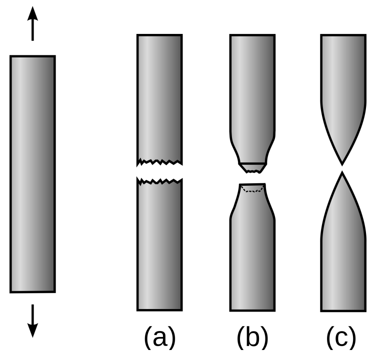


["Engineering" stress–strain (σ–ε) curve typical of aluminum](#)
[1. Ultimate strength](#)
[2. Yield strength](#)
[3. Proportional limit stress](#)
[4. Fracture](#)
[5. Offset strain (typically 0.2%)](#)
[Image courtesy od Wikipedia](#)

- Ductility "is a measure of a material's ability to undergo significant plastic deformation before rupture or breaking, which may be expressed as percent elongation or percent area reduction from a tensile test" [Wikipedia].

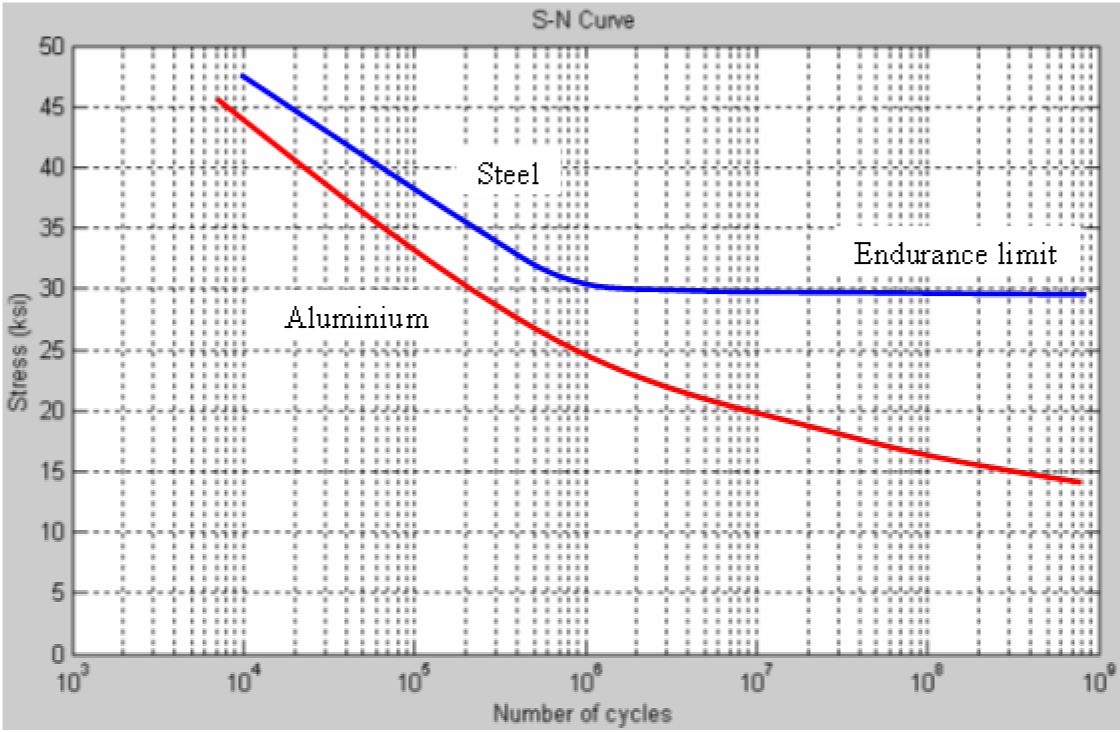
$$\%EL = \frac{final\ gage\ length - initial\ gage\ length}{initial\ gage\ length} = \frac{l_f - l_0}{l_0} * 100$$

Malleability is the compressive counterpart to ductility. Malleability "is a material's ability to deform under compressive stress".



- Coefficient of thermal expansion (CTE) “describes how the size of an object changes with a change in temperature. Specifically, it measures the fractional change in size per degree change in temperature at a constant pressure” [Wikipedia]. You’ll notice bridges or parking lot structures have expansion joints that fill gaps within the structure and act as a flexible, variable filler that help the structure adapt to temperature changes without distorting [Science Clarified].
- Fatigue occurs when a material is cyclically loaded and unloaded at a mean stress. Fatigue limit “is the stress level below which an infinite number of loading cycles can be applied to a material without causing fatigue failure” [Wikipedia]. Interestingly, aluminum seemingly has no fatigue limit. “Fatigue failures, both for high and low cycle, all follow the same basic steps process of crack initiation, stage I crack growth, stage II crack growth, and finally ultimate failure” [Wikipedia]. Characteristics of fatigue include randomness in location of failure, usual association with tensile stresses, inverse relationship between applied stress and life, and irreversible damage. “Fatigue life is influenced by a variety of factors, such as temperature, surface finish, metallurgical microstructure, presence of oxidizing or inert chemicals, residual stresses, scuffing contact (fretting), etc.”, which is why attention to manufacturing processes is important to preserve the structural integrity

of components likely to fatigue.



Representative curves of applied stress vs number of cycles for steel (showing an endurance limit) and aluminium (showing no such limit). Image by Andrew Dressel.

- Outgassing or off gassing is the “release of gas that was dissolved, trapped, frozen, or absorbed in some material” [Wikipedia]. Outgassing commonly occurs when the spacecraft is exposed to a high-vacuum environment. NASA keeps a database of outgassing data of materials intended for spacecraft use and promotes the use of materials with low-outgassing properties. “Outgassing products can condense onto optical elements, thermal radiators, or solar cells and obscure them. For most solid materials, the method of manufacture and preparation can reduce the level of outgassing significantly. Cleaning of surfaces, or heating of individual components or the entire assembly (a process called "bake-out") can drive off volatiles” [Wikipedia].

Common choices for spacecraft structures include aluminum, steel, titanium, and composites. Aluminum is incredibly common due to its high material strength with relative low density to save on mass and low cost. Steel is stronger and generally cheaper but heavier. Titanium is stronger and lighter but much more expensive. Composite materials are higher in strength and lower in density, also making them attractive candidates, but have less space heritage or historical use. For a more quantitative comparison, refer to the table below:

Material	Aluminum 6061-T6	Stainless Steel 316	Titanium Ti-6Al-4V	Carbon-carbon composite

Density	2.7 g/cc	8 g/cc	4.43 g/cc	1.6 g/cc
Young's Modulus	68.9 GPa	193 GPa	113.8 GPa	80 GPa
Tensile Yield Strength	276 MPa	290 MPa	880 MPa	260 MPa
Tensile Ultimate Strength	310 MPa	580 MPa	950 MPa	
Ductility	12 - 17 %	50 %	14 %	
CTE	23.6 - 25.2 µm/m-°C	16 - 17.5 µm/m-°C	8.6 - 9.7 µm/m-°C	0.2 - 5.7 µm/m-°C
Fatigue Strength	96.5 MPa	270 MPa-N/mm2	510 MPa	
Outgassing rate	33 x 10^-9 Pa-m^3/sec-m^2	5.1 × 10−9 Pa m^3/ sec-m2	10 x 10^−9 Pa m^3/sec- m^2	
Thermal Conductivity	167 W/m-K	16.3 W/m-K	6.7 W/m-K	4 - 27 W/m-K
Cost for ¼" x 1" x 1' bar	$3.46 USD	$23.31 USD	$98.46 USD	$20.50 USD

Process

Although the process may seem like an afterthought, we must consider the manufacturing, integration, assembly, and testing process. The design may be geometrically elegant or structurally strong but the design is not feasible if the structure components are impossible to manufacture or assemble. The most straightforward way to gauge if a design is possible to fabricate, assemble, and test is to attempt to fabricate, assemble, and test. Infeasible plans may be revealed through preliminary plans, like consulting with machinists on part drawings or generating an integration procedure. Best practice is to fabricate and test prototypes prior to the actual deadline to iron out any hiccups in the implementation progress.

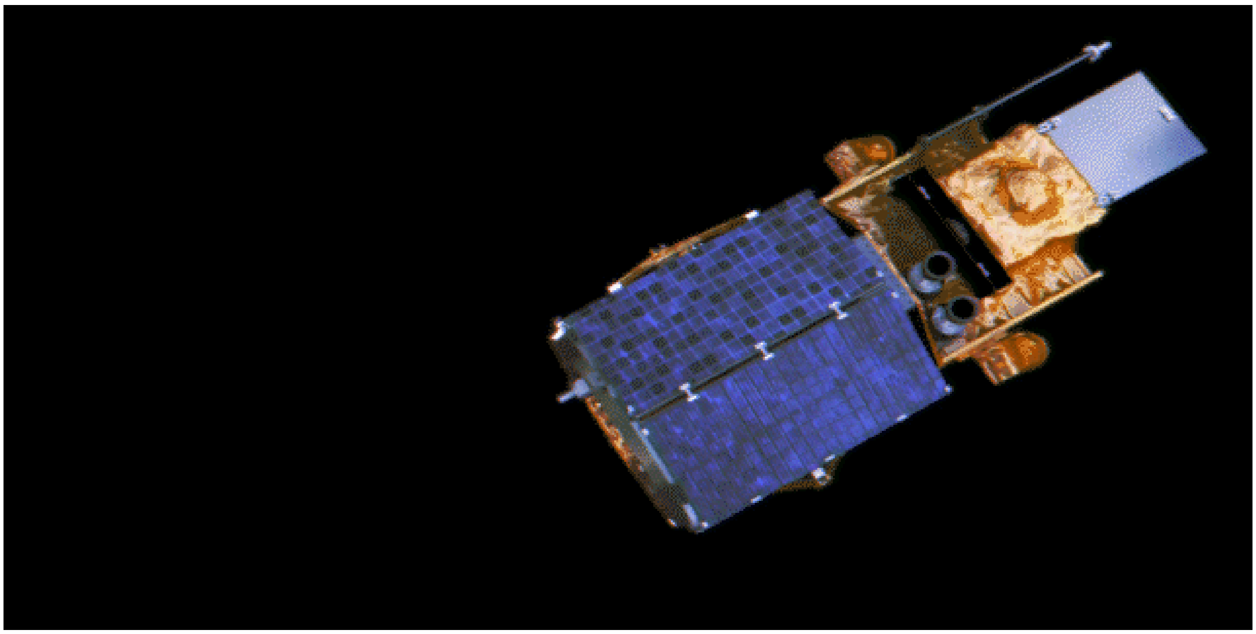
4.5 Mechanisms

For the beginner structures and mechanisms engineer, this section will provide a brief overview of the various components and what they're used for in spacecraft. These mechanisms are typically very risky as they have a significant rate of failure. As they are so risky, systems engineers prefer not to use active mechanisms unless necessary. Mechanisms may be critical for engineering solutions or science applications, which will be discussed below.

Thus, deployers are rigorously tested on ground in gravity offloading testbeds, which simulate microgravity by placing wheels underneath a structure or stringing cables from the structure to the ceiling to compensate for gravity.

Deployers

Deployers or deployment mechanisms transform a packaged spacecraft into its operational form. The common need for all deployers is the desire to achieve a different geometry than is feasible with the rocket fairing volume constraint. Deployers can achieve great lengths (booms), large surface areas (solar panels or solar sails), or immense volumes (habitation modules).



Full video can be seen: Glory Solar Array Deployment. Video Courtesy of NASA via You Tube.

Booms are typically used to take advantage of length extension. This length extension could offer spatial isolation, like mitigating electromagnetic noise for a magnetometer on the tip of the boom. A boom could also offer geometric placement for optics, like a shade or occluder. Booms may also be used to manipulate spacecraft dynamics. A boom can modify the moment of inertia of a spacecraft to create spin stability [Pankow et al.] or mass distribution of a spacecraft to create a gravity differential to preference an orientation [Kowalski et al.].

Solar panels rely on surface area to generate power. Some spacecraft, like our Artemis Cubesat kit, have solar panels on most faces of the spacecraft structure. But some spacecraft have opted to extend solar arrays away from the primary structure to get as much surface area and thus as much power as possible. This level of power generation may be critical to fulfill mission requirements. These solar arrays can't fit in the rocket fairing as is so the solar panels must be folded close to the primary structure and deployed once in orbit. Solar panel hinges and motors deploy these solar panels to their full extent. Vipavetz and Kraft give great lessons learned as to the reasons solar panel arrays have historically failed grouped into mechanical loading, on-orbit space environment, tribology (mechanisms and lubricants), and bas systems engineering.

Full video can be seen at Raw video: LightSail solar sail deployment test. Video Courtesy of The Planetary Society.

Light sails or shades are deployed much like thin booms with the addition unfurling a thin sail. The careful folding, like origami, of the sail is ingenious. The sail is made of an incredibly thin mylar material that could risk tearing with poor fabrication or assembly. This surface area is necessary for a light sail to capture as much linear momentum from photons as possible, as the individual exchange from a single photon is not much, but the summation across a large surface area can propel a small spacecraft.

Full video can be seen at James Webb Space Telescope - Unfurls. Video courtesy of Northrop Grumman.

Certain antennas, like the radar antenna on RainCube, require a parabolic dish shape that is too large to be launched as is, thus they must be compacted and deployed after launch. A small business, named Freefall Aerospace, has created a lightweight, low-volume stowed spacecraft antenna that is inflatable, bypassing rigid deployment. MarCO-A and B are “our first and second interplanetary CubeSats”, enabled by a deployable high-gain, X-band antenna flat panel.

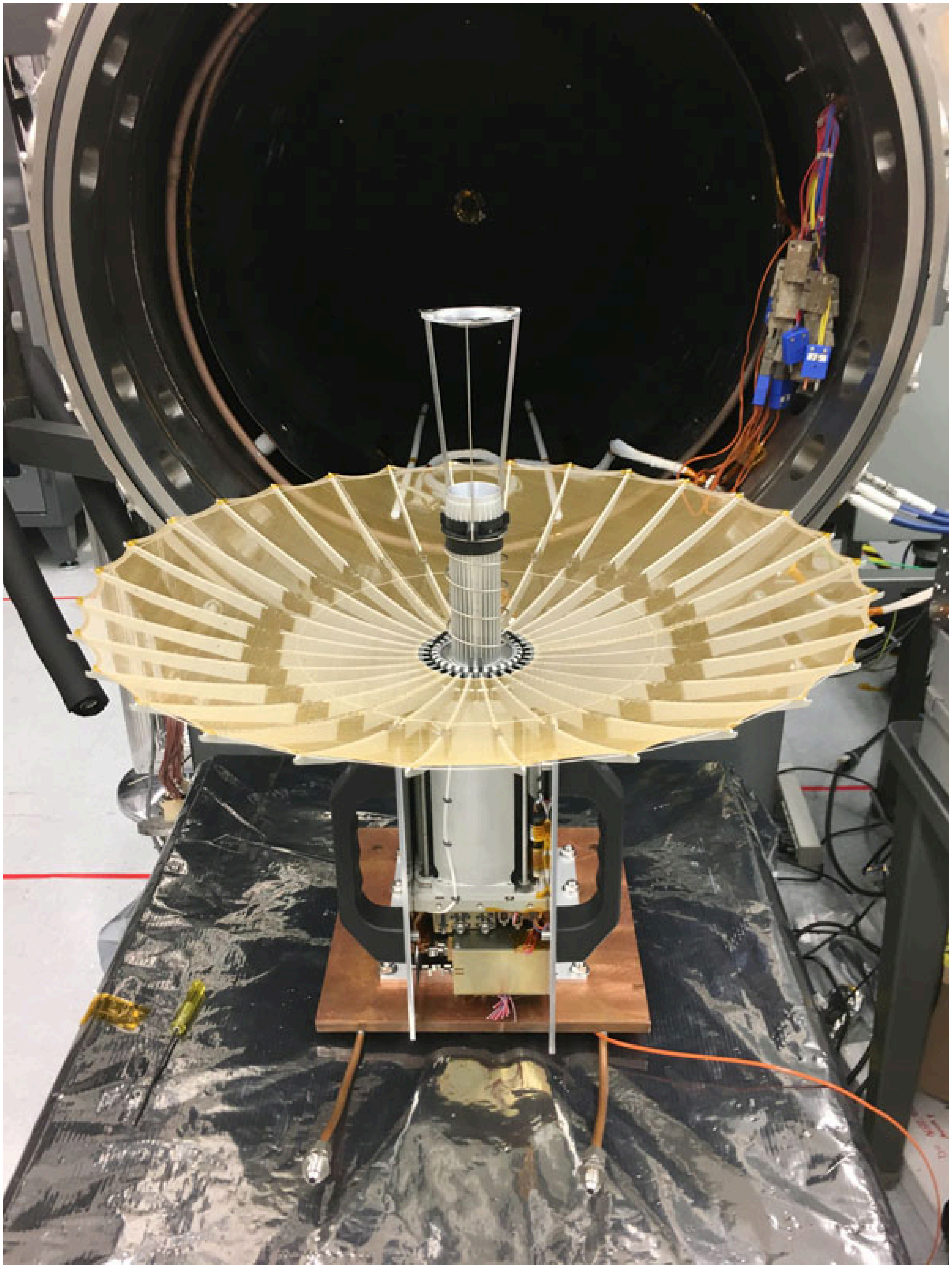
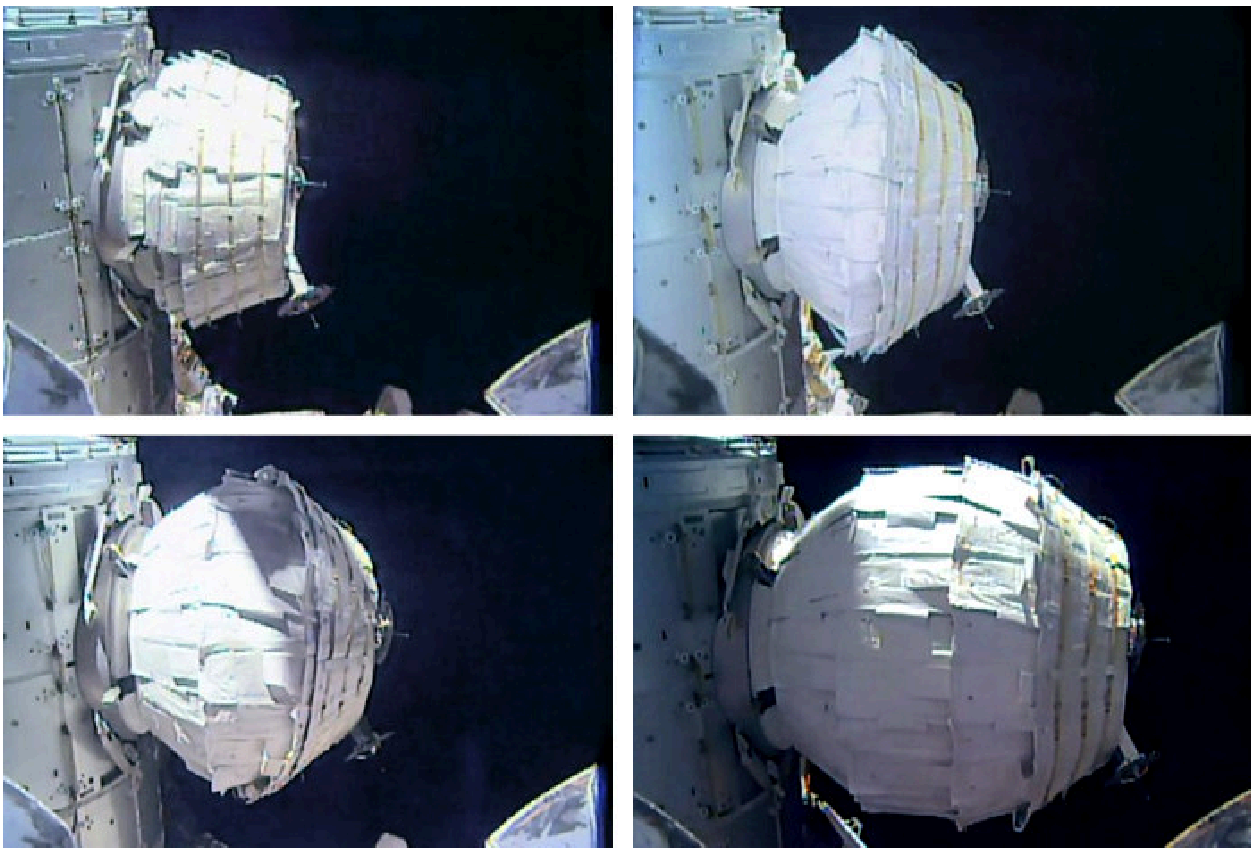


Photo: JPL/NASA

RainCube's Umbrella: The radar antenna for the tiny RainCube satellite folds up into a 10-by-10-by-15-centimeter canister. Upon deployment, its 30 ribs extend like an umbrella to form a parabolic dish that's still small enough to test in a thermal vacuum chamber.

Inflatable space habitats are a deployable module for crewed space. The ISS has an expandable habitat called the Bigelow Expandable Activity Module that has been operational since 2016. These habitats are pressurized structures and provide greater volume of living space [Wikipedia]. There are proposed uses of inflatable habitats on planetary surfaces but no instances yet, only uses in space.



Series of photos showing the expansion of the Bigelow Expandable Activity Module to its full size on May 28, 2016.

Restraints or Launch Locks

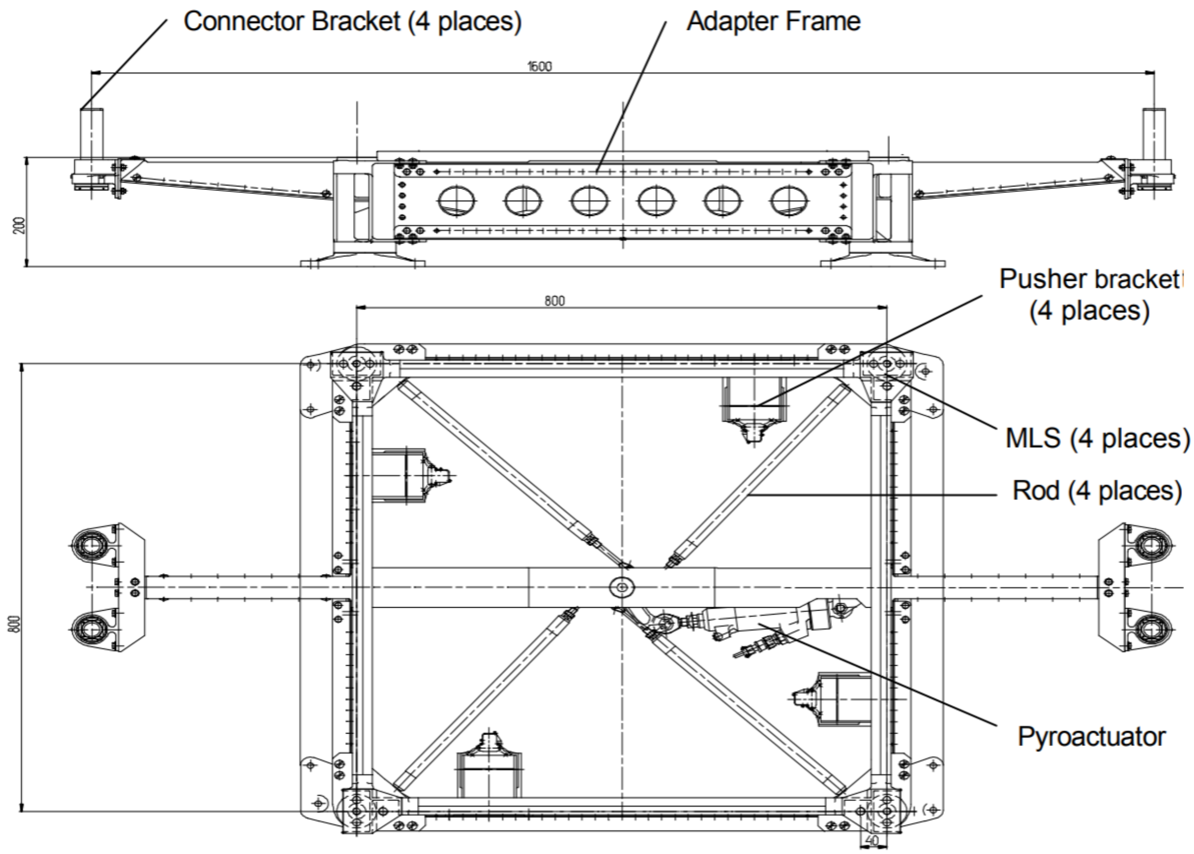
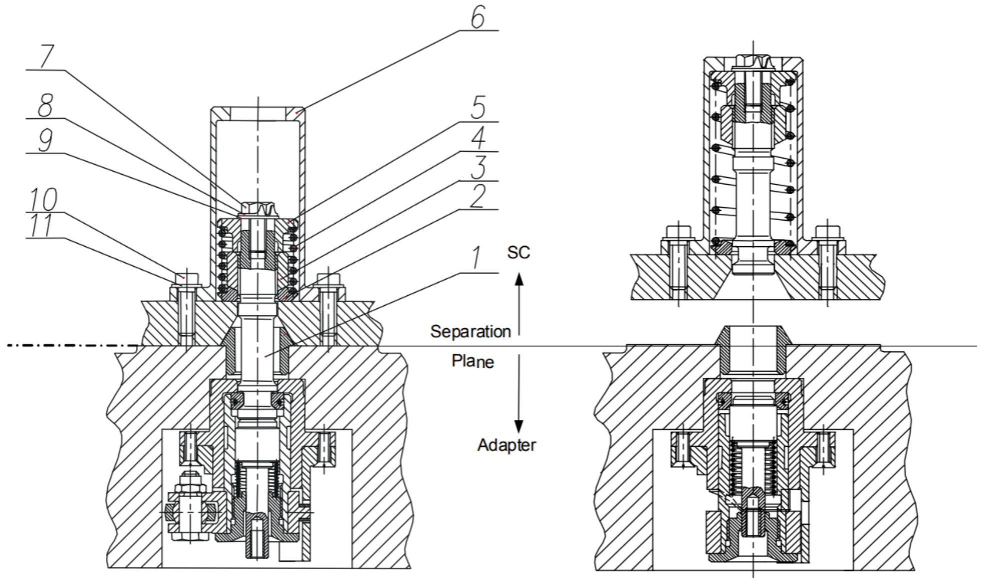


Figure 4-14 Example of an MLS adapter system for single satellite accommodation.

[Eurockot's MLS adapter system for a single satellite accommodation](#)
[Rockot User's Guide, EHB0003, Issue 5, Revision 0, August 2011](#)

Restraints or launch locks restrain the payload and isolator during spacecraft launch. The spacecraft interfaces with an adapter or dispenser system of the launch vehicle, commonly called the Mechanical Lock System [Eurockot]. The adapters and interfaces vary with the spacecraft and rocket, but there are some standards associated with size. For example, CubeSats can rely on the PPOD deployer for mechanical interfacing.



Note: The indicated parts remain on the spacecraft after separation: 1,7,10 = Bolt, 2,8,9,11 = Washer, 3 = Screw-nut, 4 = Spring, 5 =Support, 6 = Bolt Retainer.

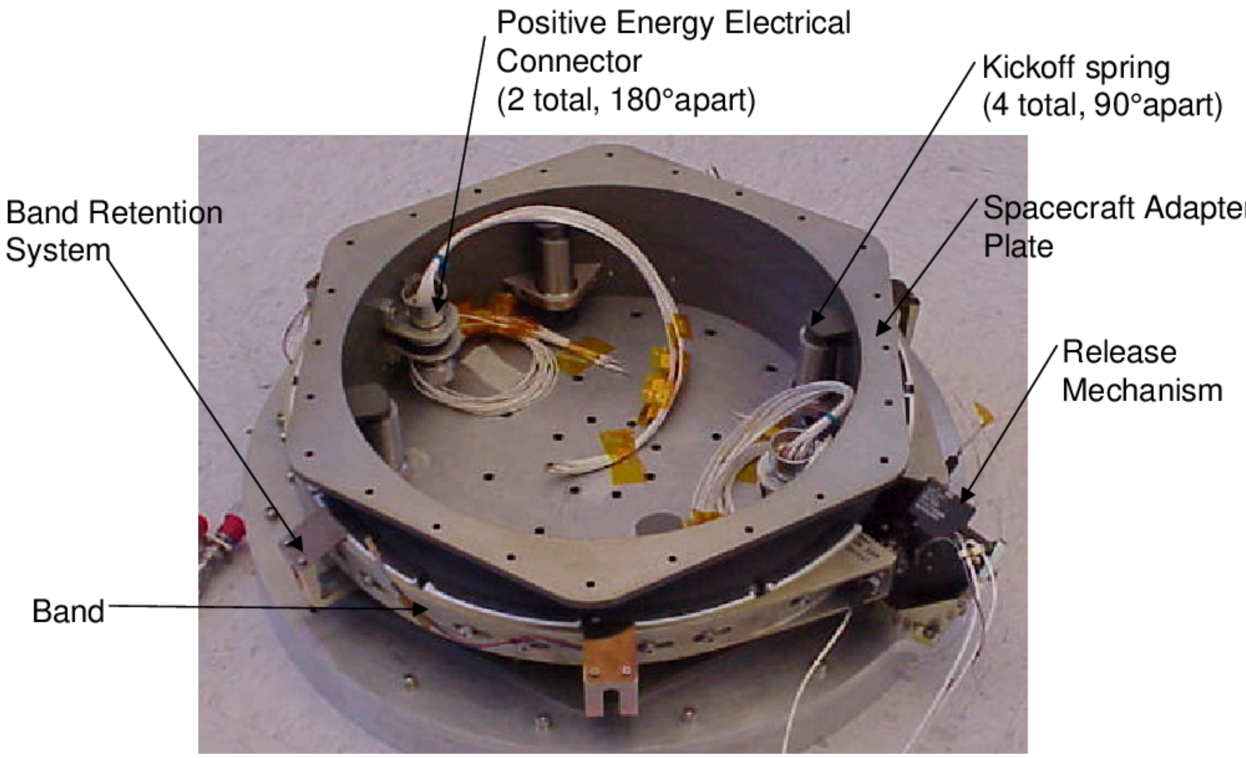
Figure 4-8 Cut-away detail of the Mechanical Lock System.

[Cut-away detail of the Eurockot's Mechanical Lock System](#)
[Rockot User's Guide, EHB0003, Issue 5, Revision 0, August 2011](#)

Separation Mechanisms

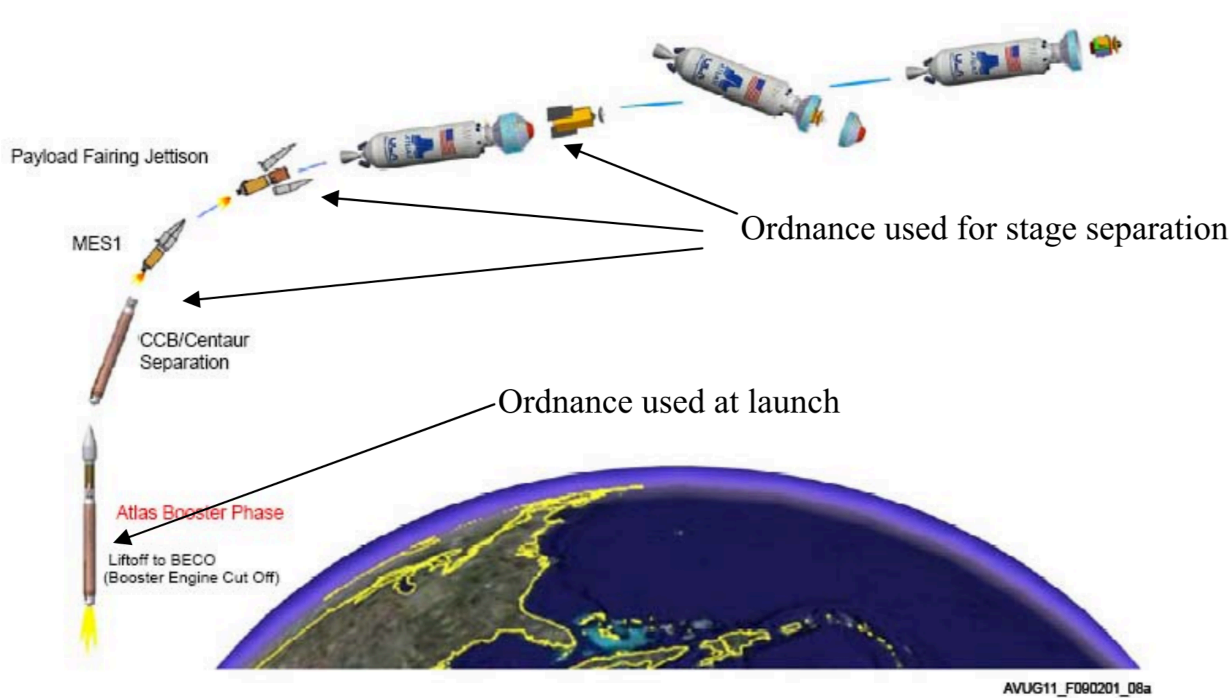
https://youtu.be/LE1e1Bg2Dj8

Separation mechanisms disconnect the spacecraft from the launch vehicle once in proper orbit. There are many options for separation mechanisms: clamp bands, motorized light bands, Marmon clamps, dispensers, and custom systems [Spaceflight]. The best technical solution depends on the size of the spacecraft, launch vehicle provider, allowable shock, and typical tip-off rate. Separation mechanism characteristics include the imparted velocity in the axial and lateral direction, spin rate, umbilical connectors to supply power or data, allowable volume or length dimensions, and any applied loads.



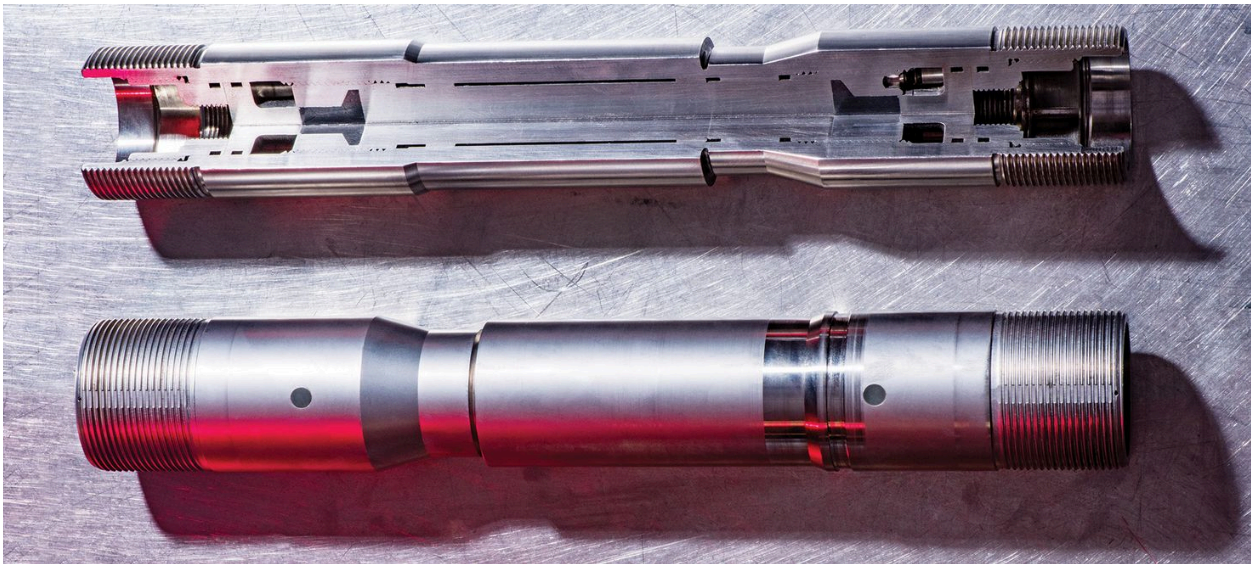
[Clamp band system with major components identified. System is shown in clamped, preloaded configuration.]

Ordinance Devices



[Typical launch with ordnance initiated events. Iyengar et al.]

Ordnance is an explosive device that enables the sudden release of spacecraft. Ordnance systems initiate important discrete events, like lift off, stage separations, spacecraft separation and flight termination [ULA launch]. Ordnance often incorporate the use of explosive bolts, or pyrotechnic fasteners, in separating different stages of the launch vehicle and spacecraft. An explosive charge separates the bolt at a specified break plane. The explosion can be the result of an explosive detonating material or a pyrotechnic pressure generating material [Pacsci EMC].



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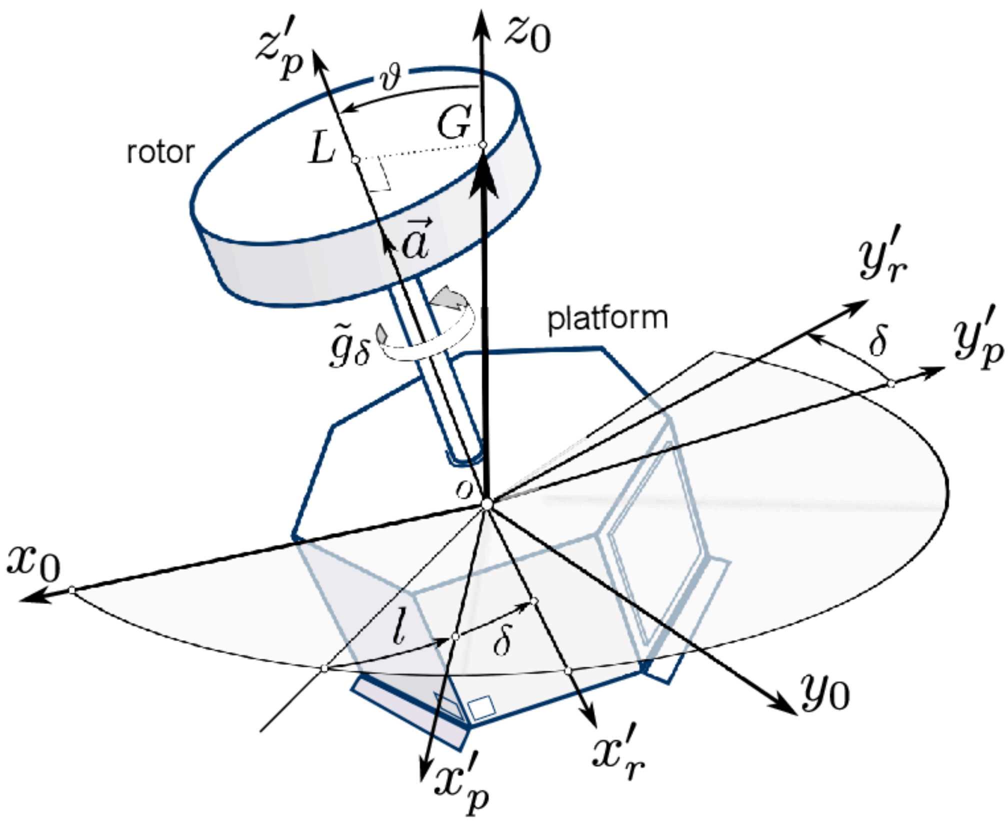
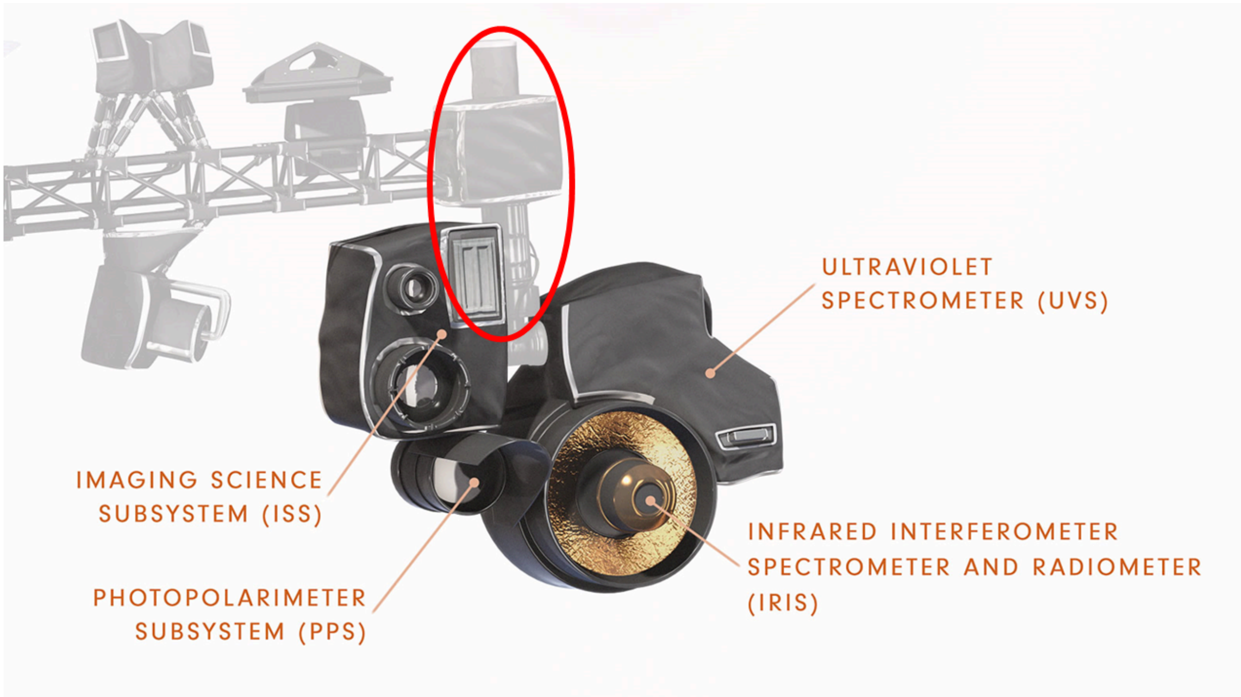
Spin Bearings



Dynamics and control of dual-spin gyrostat spacecraft with changing structure. V. Aslanov, V. Yudintsev.

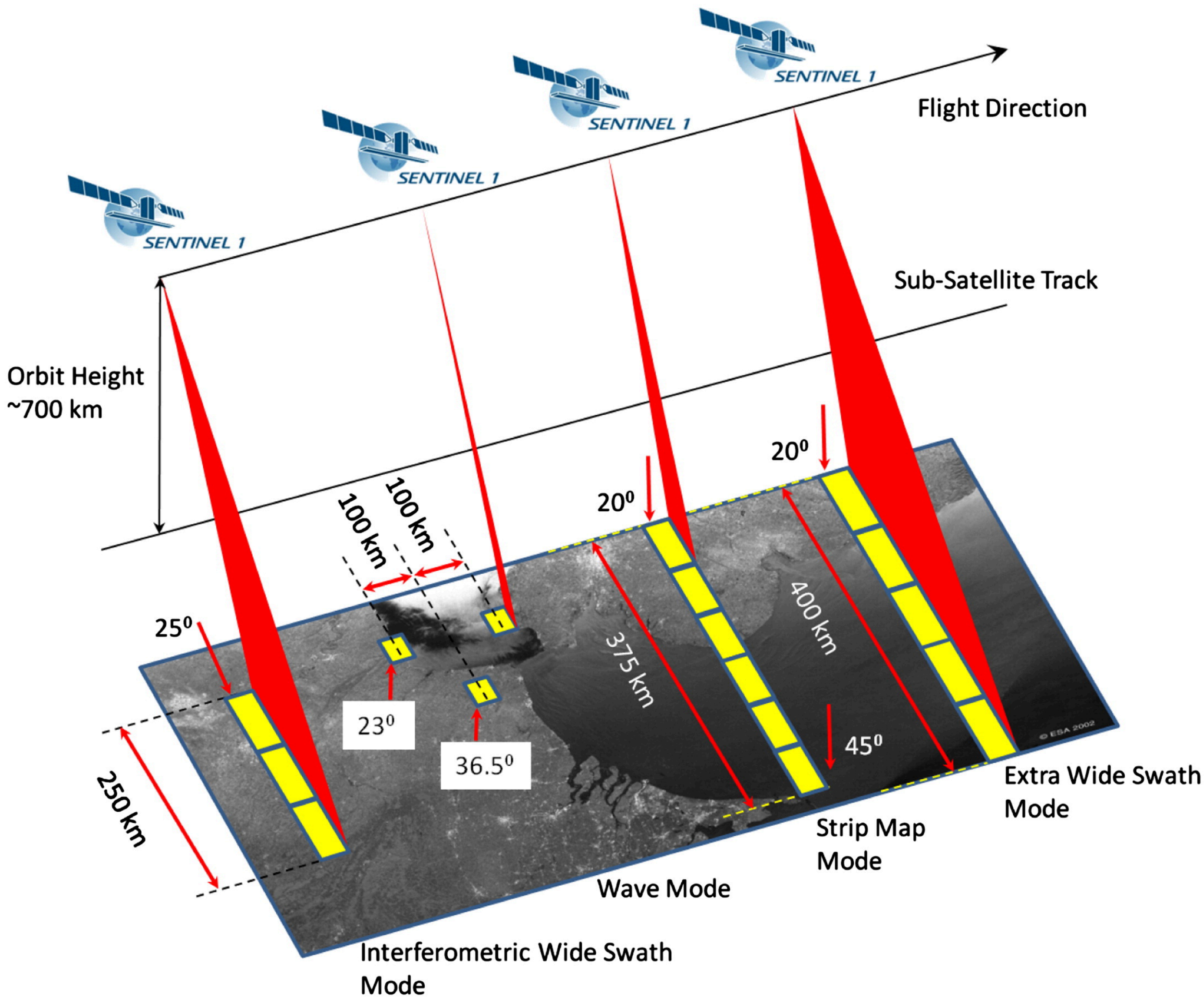
An example of spin bearings, or ball bearings, in space is a dual-spin spacecraft. Two rigid bodies are connected by a bearing that allows the two bodies to rotate at different rates. The spinning of one body stabilizes the other body so that the payload may track or point. Bearings notoriously fail in space, although specifically reaction wheel bearings and not passive mechanical joints. Bearings naturally wear over time, accumulating friction and potentially jamming up. Some bearings fail due to bearing damage caused by electrical arcing [AEGIS]. Rotating joints also increase complexity in design. “When electrical power or signals must be passed across a rotating interface, a slip ring (sliding electrical contact) or twist capsule (specialized flexible harness) is required” [Honeybee Robotics].

Scan Platforms



[At the end of the science boom is the moveable scan platform that houses 5 optical sensing instruments, including the two cameras. Image courtesy of PBS.]

Some payloads require sweeping across a field of view to collect swaths of information. To conduct a sweep, the payload may be mounted to a scanning platform. A famous example of a scan platform is on the Voyager spacecraft. The payloads look away from the rotation axis and collect visible, UV, and IR data along a horizontal plane. Another sample mission is SENTINEL-1, that carries a synthetic aperture radar instrument that scans quickly along elevation and azimuth.



[SENTINEL-1 Modes: The IW mode images three sub-swaths using Terrain Observation with Progressive Scans SAR (TOPSAR). Image courtesy of ESA.]

4.6 Structural Analysis

The goal of structural analysis is to aid the structural engineer in designing, evaluating, and verifying the structural integrity of the structures and mechanisms on the spacecraft. Typical requirements dictate margins of safety for critical structural components that must be proven through testing or finite element analysis. In aerospace engineering, safety is a critical consideration in the design process.

Safety Factors

$$Factor\ of\ Safety = \frac{Failure\ Load}{Design\ Load} = \frac{Failure\ Stress}{Design\ Stress}$$

$$Margin\ of\ Safety = Factor\ of\ Safety - 1.0$$

The idea of safety can be numerically characterized by the terms factor of safety and margin of safety. Factor of safety is the ratio of failure load to design load, or equivalently, failure stress to design stress as stress is load normalized to area. The design load is what you anticipate seeing on the structure in realistic conditions; in our analysis, the design load is the critical load. The failure load is how much the structure can withstand before failure, derived from a back-of-the-envelope calculation, finite element analysis, or testing. Structural components are not just designed to bear the critical or design load; they are designed to withstand much more than the intended critical load. For bridges, the factor of safety is 10, meaning that if the bridge anticipates 1 car's weight in a footprint, the bridge was designed and built to withstand 10 cars in that same footprint: a very conservative and safe design. For aircraft and other aerospace engineering applications, the factor of safety is very commonly 2. Factor of safety is a user-defined threshold that the structure design must meet, typically imposed by the end-user, customer, or structural engineer. This number is defined by how uncertain you are of the load or structure or how safe you want to be; more uncertainty and more safety both lead to higher factors of safety. When in doubt, crank that factor of safety up. The trade-off to imposing too high of a safety factor is that could lead to significant mass accumulation as stronger parts are usually achieved with more mass.

The NASA Structural Design and Test Factors of Safety for Spaceflight Hardware document specifies for various factors of safety that must be met for various materials. There are two different failure loads that are used in the definitions: 1) Ultimate Design Load: The product of the ultimate factor of safety and the limit load and 2) Yield Design Load: The product of the yield factor of safety and the limit load. These loads correlate with ultimate strength and yield strength of the material structure. "Structural designs generally should be verified by analysis and by either prototype or protoflight strength testing. For metallic structures only, it may be permissible to verify structural integrity by analysis alone without strength testing" [NASA STD].

Table 1—Minimum Design and Test Factors for Metallic Structures

Verification Approach	Ultimate Design Factor	Yield Design Factor	Qualification Test Factor	Proof Test Factor
Prototype	1.4	1.0*	1.4	N/A or 1.05**
Protoflight	1.4	1.25	1.2	N/A or 1.05**

* Structure has to be assessed to prevent detrimental yielding during its design service life, acceptance, or proof testing.
** Propellant tanks and SRM cases only.

Structural Design and Test Factors of Safety for Space Flight Hardware. Image Courtesy of: NASA.

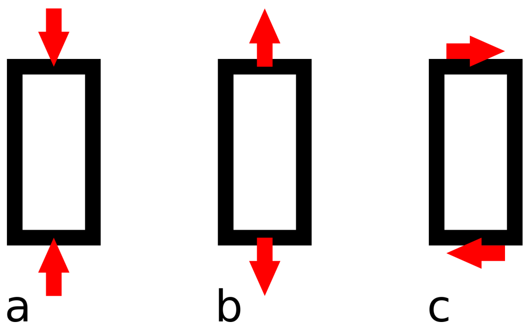
For the Artemis CubeSat Kit, the primary structure is the aluminum skeleton frame that immediately interfaces with the deployer and contains all the cubesat components. The critical load is the launch acceleration load coupled with the deployer compression at 1320 N. Through finite element analysis, the factor of safety was deemed to be __, or __ toilet paper rolls in the time of COVID. A finite element analysis tutorial is at the end of the chapter.

Load Equations

In the Structural Loads section of this chapter, we discussed the driving critical loads (design loads) but how do we relate these loads to factors of safety? In this section, we will cover some key structural formulas for back-of-the-envelope calculations, valid for simplified geometries/models. The following sections describe 1 degree-of-freedom problems but structures reside in 3 dimensions. Make sure to repeat calculations for all degrees of freedom or axes.

Ultimate and Yield Loads

We talked about estimating critical loads or design loads but to get a factor of safety, we need to also find the failure loads. The failure load comes in two flavors: ultimate load and yield load. To find ultimate load and yield load, we refer to the structure's material properties to extract ultimate strength in yield strength in units of Pascal or N/m^2.



A material being loaded in a) compression, b) tension, c) shear. Strength of Materials Courtesy of Wikipedia.

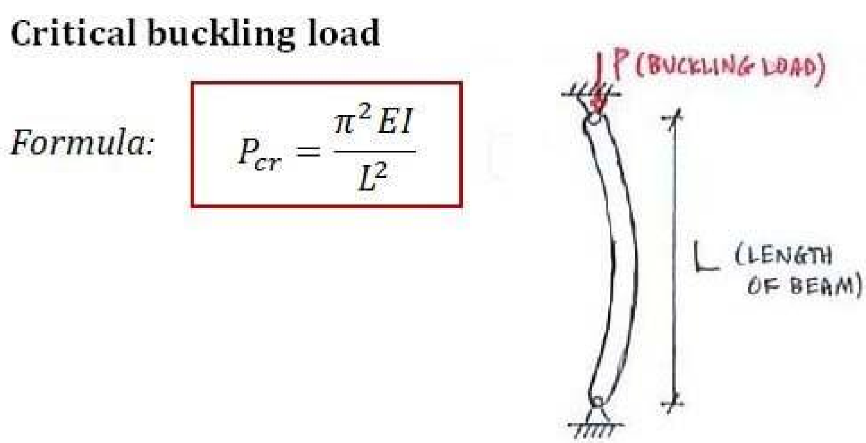
Stresses come from different directions of loading: compression, tension, and shear. Material sheets will typically specify the strength associated with each direction as the yield or ultimate strengths value are different. Direction of loading matters so make sure you use the correct strength number! The area that the load travels across also matters. The stress formula is:

$$\sigma = \frac{F}{A} \text{ or}$$

where F is the force and A is the cross-sectional area. If we plug in yield or ultimate strength in \sigma and we know the cross-sectional area of the piece we are analyzing, we have yield/ultimate force as our one unknown to solve for.

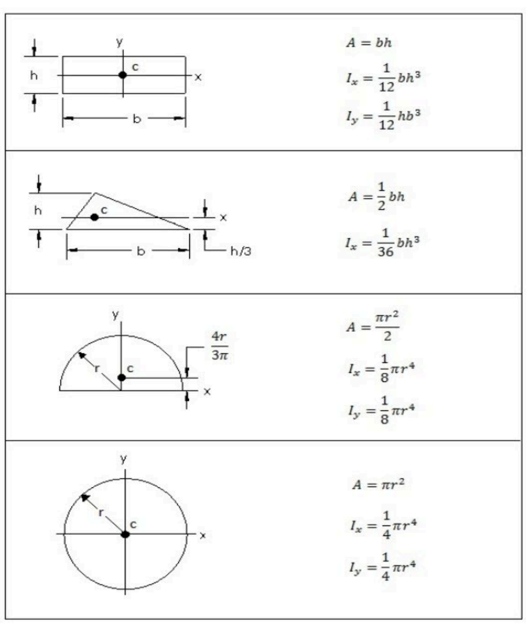
$$F = \sigma A$$

Buckling Load

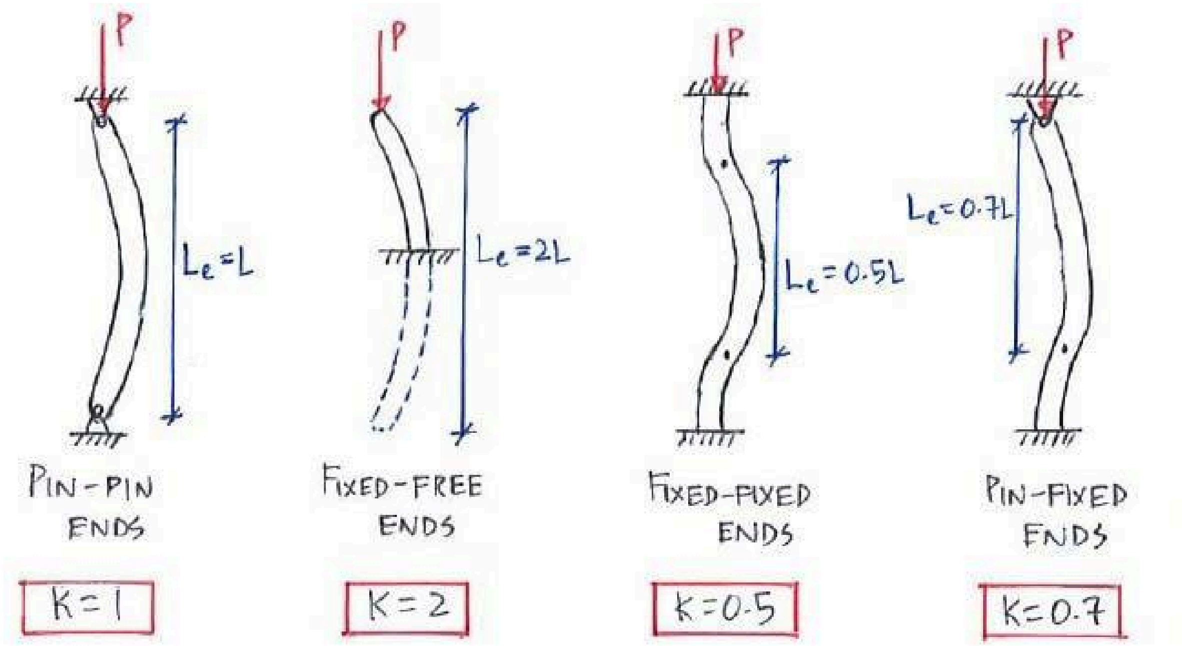


Euler's Buckling Formula. Courtesy of Engineering Course.

Buckling is a failure mode of compressive loading in which the two ends of a beam are constrained and the beam fails by bending, seen in Fig __. This loading scenario describes a slender member bolted at two ends experiencing a compressive load, like the Artemis CubeSat frame corner posts under launch acceleration. The equation to find the critical buckling load, P_cr, depends on the Young's modulus, E, the moment of inertia that resists the direction of buckling, I, and the length of the slender member. As the length of the slender member is likely constrained, the slender member's strength can be scaled by varying the moment of inertia in that direction. One of the edge lengths affects the moment of inertia to the cubic power, which could be taken advantage of to quickly reinforce a beam that is facing critical loading.



The Moment of Inertia Modul. Image Courtesy of VCCS Engineering Education via Course Bridge Modules.



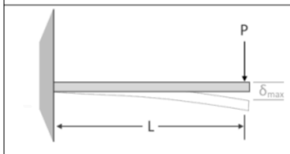
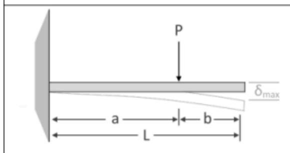
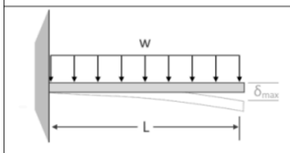
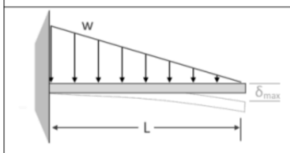
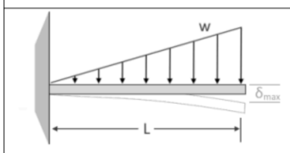
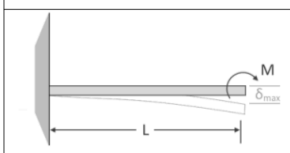
For different boundary conditions and pin constraints, the modified critical load equation is scaled by K^2, where K is defined by the effective length of the slender member:

$$P_c = \frac{\pi^2 EI}{K^2 L^2}$$

You'll notice that by minimizing K, the critical load will increase. Decreasing K involves constraining the beam or slender member along the length, "breaking up" the effective length. For spacecraft support members, "breaking up" the effective length could involve bolting a strut to the main member.

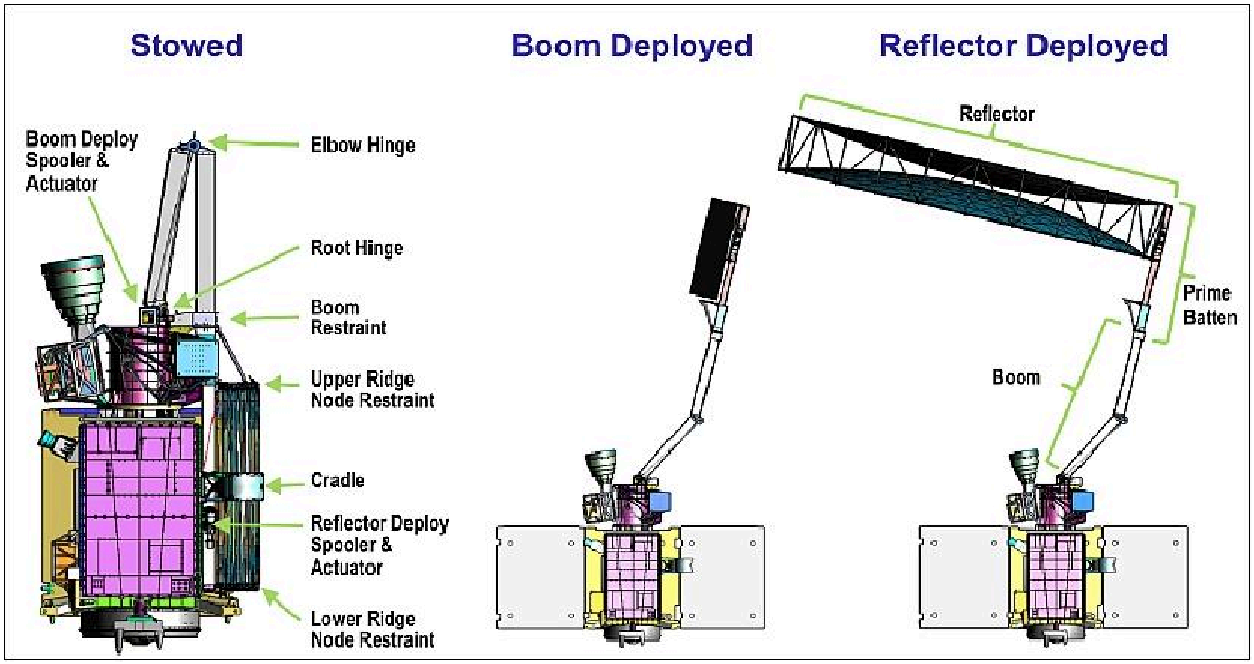
Beam Stiffness

A cantilever beam is a slender structure with a fixed constraint on one end and no constraints on the other end (free end), like a deployed solar array or boom. A cantilever beam deflects if a load is imparted along the length of the beam or the beam experiences a load if a deflection is forced. There are various loading cases seen in Fig. __ with corresponding formulas, where \delta_max is the maximum beam deflection, P is the load, L is the length of the beam, E is the Young's modulus, I is the moment of inertia in the loading direction, and M is a moment or torque.

Beam and load cases	Maximum Beam Deflection
	$$\delta_{max} = \frac{PL^3}{3EI}$$
	$$\delta_{max} = \frac{Pa^2(3L-a)}{6EI}$$
	$$\delta_{max} = \frac{wL^4}{8EI}$$
	$$\delta_{max} = \frac{wL^4}{30EI}$$
	$$\delta_{max} = \frac{11wL^4}{120EI}$$
	$$\delta_{max} = \frac{ML^2}{2EI}$$

[Beam Deflection Calculator. Image Courtesy of: Omni Calculators]

These formulas may be useful to convert between beam deflection and force. If we know the acceleration profile acting on the beam, we can calculate the deflection along the beam, important for missions like SMAP, which has a spinning large flexible reflector/structure that points toward the Earth. The spinning motion produces centrifugal force and the mass at the end of the beam accentuates the centrifugal loading.



Stages of Reflector Deployment. Image Courtesy of Earth Online

A cantilever beam's yield and ultimate load differs from the buckling load in that the buckling load is parallel with the length of the beam whereas the cantilever beam's load is perpendicular with the length. The failure load may be calculated by calculating the intermediate variable, bending moment. An example of the moment calculation for the simple point source force at the end of a cantilever beam is: M_max = - F L, where F is the magnitude of force and L is the length of the beam. The critical bending moment occurs at the fixed end of the beam, or the joint between the beam and primary structure. The stress in a bending beam can be expressed as σ = y M / I where σ is stress, y is distance to point from neutral axis, M is bending moment, and I = moment of Inertia. This stress value may then be used in the yield and ultimate factor of safety.

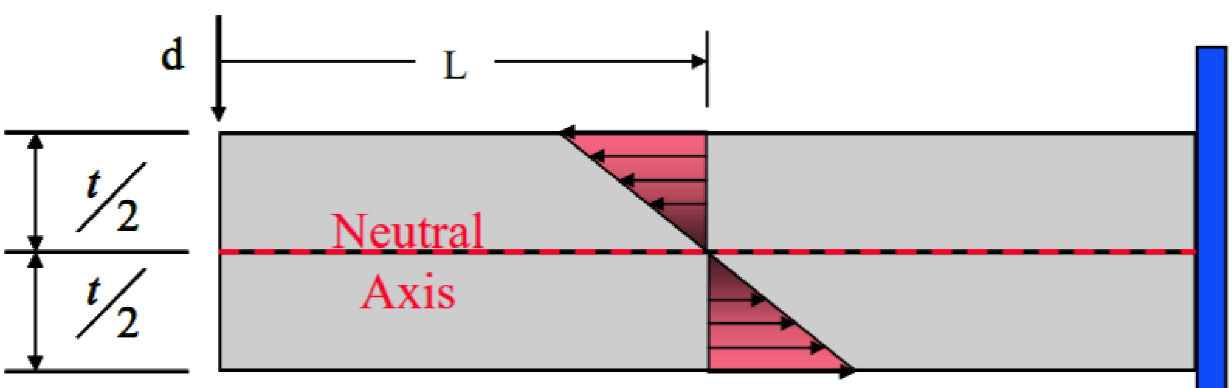


Figure 2. Cantilever Beam Stress Distribution

Cantilever Beam Stress Distribution. Technical Tidbits ©2010 Brush Wellman Inc.

Beam Natural Frequencies

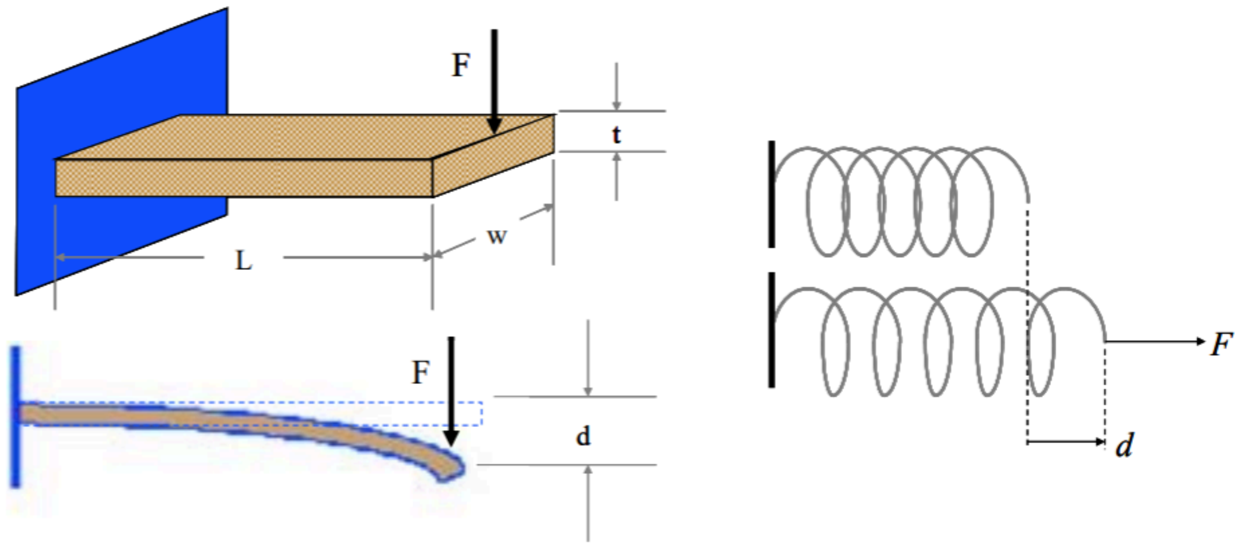


Figure 1. Cantilever Beams vs. Coil Springs

Cantilever Beam vs. Coil Spring Technical Tidbits ©2010 Brush Wellman Inc

Structures under dynamic loads (vibration or acoustic loads) can exhibit resonate at the natural frequencies and cause failure. The cantilever beam, like a diving board, can deflect and vibrate once the load suddenly disappears. A common scenario that this vibration describes is a sudden boom deployment, like the Artemis CubeSat antenna deployment event. The phenomenon that a cantilever beam experiences deflection and vibration may be captured in an analogy with a spring. The structure deflects or displaces with a load, like a spring, and upon release, the beam oscillates around the unloaded equilibrium, like a spring. The natural frequency of the beam in units of rad/s may then be calculated with the following formula

$$\omega_{nf} = \alpha_n^2 \sqrt{\frac{EI}{mL^4}}$$

[Meirovitch, 1967]:

where \omega_nf is the natural frequency, m is the mass of the beam, L is the length of the beam, E is the Young's modulus, I is the moment of inertia in the loading direction, and \alpha_n are coefficients describing the first, second, and third natural frequencies. The first natural frequency is the fundamental frequency.

$$\alpha_n = 1.875, 4.694, 7.885$$

The natural frequency, _n may be converted to units of Hz with a 2\pi conversion factor: f_n = \omega_nf / 2*pi.

Random Vibe and Acoustic Equivalent g's

A structure experiences a load due to random vibrations and can be approximated by a number of g's, an acceleration unit. Developed by John Miles in 1954, GRMS is Root Mean Square Acceleration in G's (sometimes given as ÿRMS) that relates natural frequency f_n, transmissibility Q at f_n (1/2\zeta where \zeta is the critical damping ratio), and input acceleration spectral density ASD_input at f_n [Simmons]:

$$G_{RMS} = \sqrt{\frac{\pi}{2} f_n Q \; [ASD_{input}]}$$

The expected stress as a result of G_rms from random vibration loads is then:

$\sigma_{RV} = \frac{mG_{RMS}}{I}$ where m is the mass of the beam and I is the moment of inertia in that vibration axis.

Thermal Load

Materials expand at different rates, dictated by their coefficient of thermal expansion (CTE) and their temperature difference. If these structures of differing CTE are bonded together and undergo a temperature difference, the structure will change in length and experience stress. The structural change could result in a deflection (load perpendicular to the length of the beam) or shrinkage/elongation (load parallel to the length of the beam). The change in length is calculated by the total length L, change in temperature \delta T and CTE \alpha.

$$\Delta L = \alpha \Delta T L$$

The resultant stress from deflection may be calculated with a previous Beam Stiffness section and the axial stress, \sigma, can be found with the following formula:

$\sigma = E \, \Delta L$

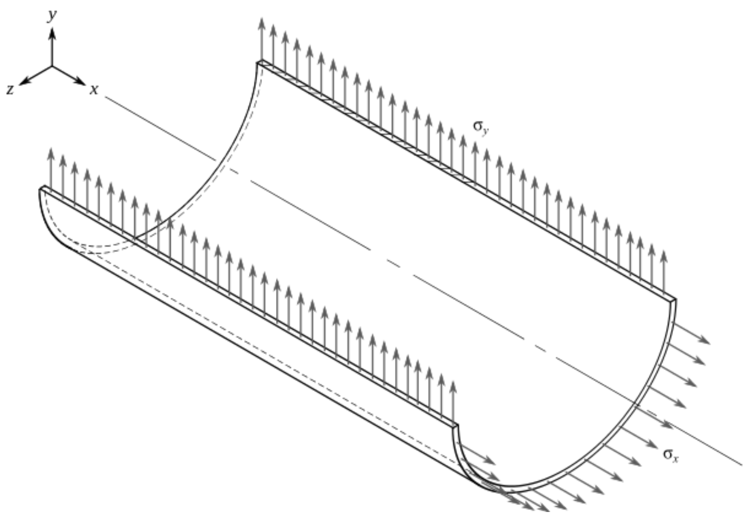
where E is Young's modulus and \delta L is the length change calculated previously.

Simple Pressurized Shell

The large hydrazine propellant tank prior to integration with the core structure of the MAVEN spacecraft at a Lockheed Martin clean room near Denver. Image Courtesy of NASA.

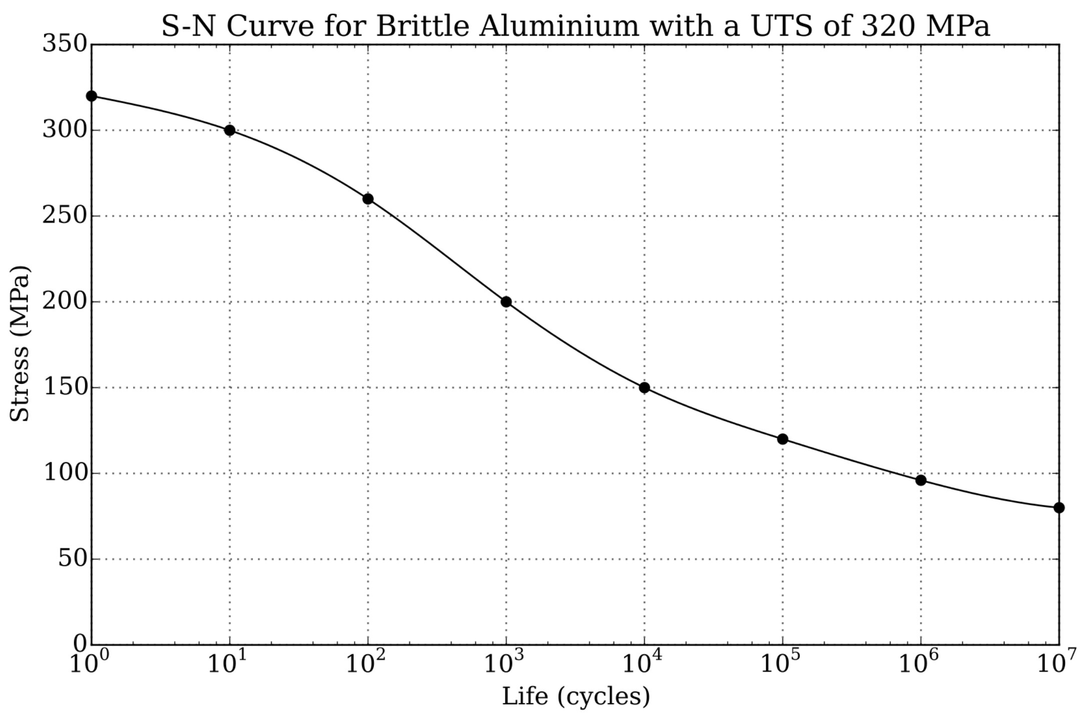
Pressurized vessels are rounded sheets that experience tension if the vessel is in positive pressure or compression if the vessel is in negative pressure. Think of blowing up a balloon as positive pressure and sinking a submarine for negative pressure. We care about this stress when dealing with propulsion systems that store propellant. These walls can burst if the cylinder walls experience more stress than the tensile ultimate strength. The thin walls of the vessel experience hoop stress and longitudinal stress, related to pressure p, radius R, and thickness t:

$$\sigma_{hoop} = p\,R\,/\,t \;>\; \sigma_{long} = p\,R\,/\,2\,t$$



Stress in the cylinder body of a pressure vessel. Image Courtesy of Wikimedia

Fracture and Fatigue Analysis



S-N curve for a brittle aluminium with an ultimate tensile strength of 320 MPa. Image Courtesy of Wikipedia.

Load cycle modeling examines the periodic nature of loads throughout a spacecraft's lifetime. These loads may incite fatigue and failure in components; think of it like wear and tear in structural components. Load cycles can occur from thermal cycling due to periodic exposure to the sun or dynamic maneuvers (like SMAP's constant rotation deflecting a slender member or Curiosity's wheels running over sharp rocks).

Briefly mentioned in the materials section, fatigue limit is correlated with the material's endurance limit, the component's mean load, and ultimate tensile strength. It's hard to know when a component will break under fatigue due to the stochastic nature of the cracks and failures. Fatigue S-N curves, fatigue prediction models, and tests can offer validation that a structural component will not fail under fatigue. For metallic structures, if the number of cycles stays below 10^4 with loading reasonably far away from yield or ultimate loads, the component will likely survive the mission lifetime. Components undergoing higher cycles should be more carefully scrutinized or replicated for fatigue testing.

Finite Element Analysis

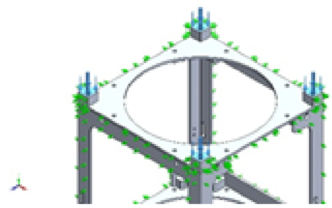
Finite element analysis is the use of computer generated geometries, numerical methods, and first principles of loads described above. The finite element model breaks down computer geometries into smaller elements and approximates the transfer of loads, cumulative deflection, and distribution of stress for static analysis. Finite element analysis software is complementary and commonly built into the CAD software, like Autodesk Inventor, Solidworks, and OnShape.

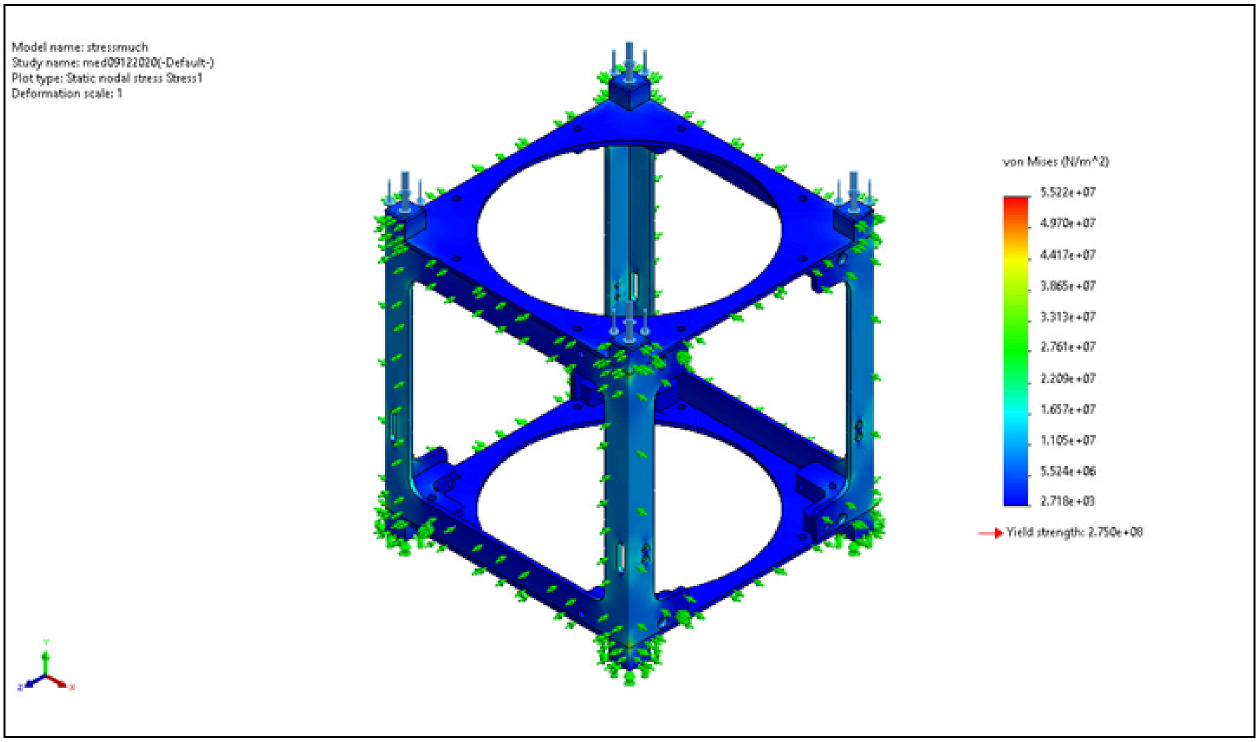
The entire spacecraft model may be designed and analyzed within one of these software packages.

1. By defining material properties for each component in the spacecraft model, ultimate strength, yield strength, CTE, Young's modulus, and density are all embedded in the model.
2. The primary structure, secondary structure, and all supporting component interfaces must be defined and constrained in motion.
3. Finite element analysis assists in identifying critical loads on each part in the spacecraft model by applying load conditions on the primary structure and probing the resultant stress on the rest of the spacecraft structure.
4. The stress can then be converted to margins or factors of safety. The software will identify the location of the minimum safety factor. The load acting on this component at this location is the critical load.
5. If this minimum safety factor does not satisfy the requirements, there must be a redesign of the critical component so that the minimum safety factor is achieved. This process must be iterated from step 3 until all components meet the minimum safety factors for all potentially critical loading scenarios.

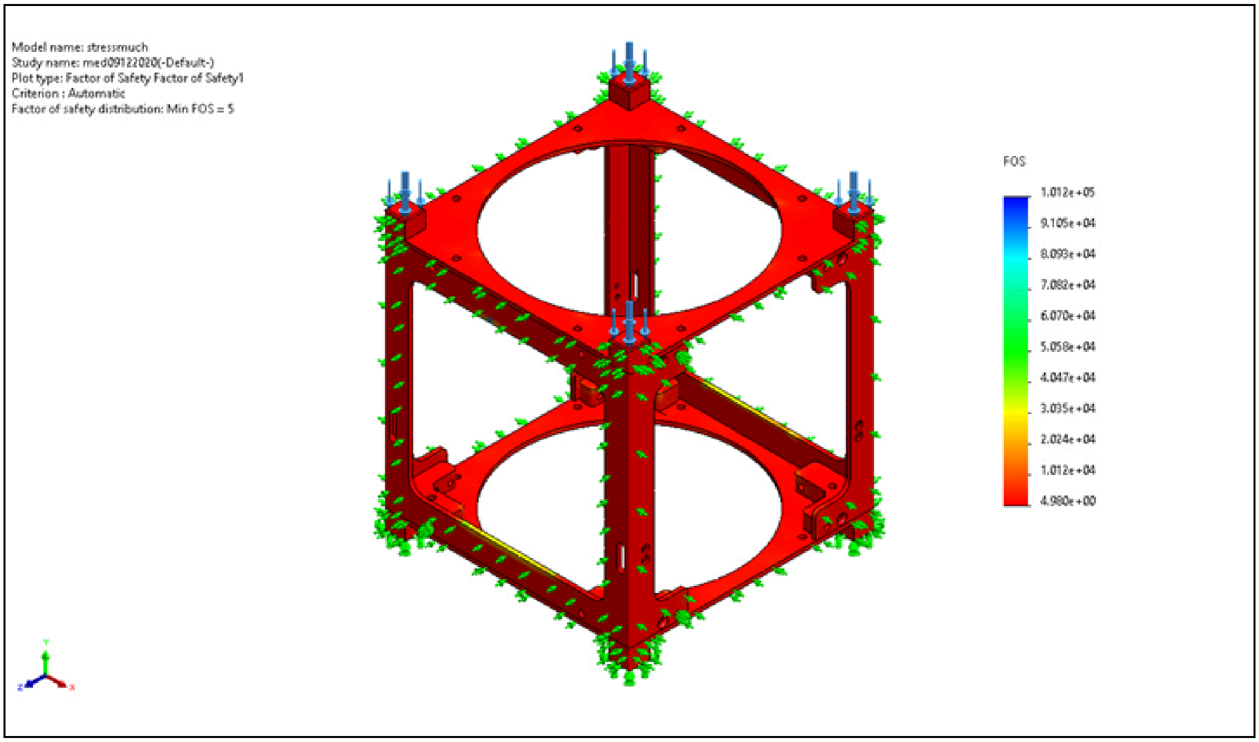
Artemis Finite Element Analysis Results

In order to meet requirements set by launch service providers, the artemis project completed a finite element analysis for static loads using Solidworks. Graphs are shown below to detail the load conditions, von mises stress, and factor of safety. By taking the calculated minimum factor of safety and design load (1,200 N), the max failure load was found to be 6 kN (approximately 2,700 toilet paper rolls).

Load Name	Load Image	Load Details
Force 1		Entities: 4 face(s) Type: Apply normal force Values: 1,200 N



Name	Type	Min	Max
Stress1	VON: von Mises Stress	2.718e+03N/m^2 Node: 13073	5.522e+07N/m^2 Node: 5315



Name	Type	Min	Max
Factor of Safety	Automatic	4.980e+00 Node: 5315	1.012e+05 Node: 13073

Suggested Activity

back of the envelope kind of calculation of structural load and natural frequencies (vibe) -> FEA

Reference Documents

Launch Services Program Level Dispenser and CubeSat Requirements Document [NASA LSP-REQ-317.01] & [CubeSat Design Specifications Rev 14]

2.2 CubeSat Mechanical Specifications

CubeSat dimensions and features are outlined in the CubeSat Specification Drawings

Note: The CubeSat Inspection and Fit-check Procedure (CIFP) can be used to aid in verifying that the CubeSat meets the dimensional requirements specified in Appendix B. The CIFP can be found on cubesat.org.

These requirements are applicable for all dispensers not utilizing the tab constraint method. CubeSats designed with tabs can find those specific requirements at the PSC website (planetarysystemscorp.com).

2.2.1 The CubeSat shall use the coordinate system as defined in Appendix B. The origin of the CubeSat coordinate system is located at the geometric center of the CubeSat.

> 2.2.1.1 The CubeSat configuration and physical dimensions shall be per the appropriate section of Appendix B.
>
> 2.2.1.2 Note: Extra volume may be available for 3U, 6U, and 12U CubeSats. This extra volume is shown in Figure 3, sometimes referred to as the “Tuna Can” volume. The availability and volume dimensions are dispenser dependent.

2.2.2 The –Z face of the CubeSat will be inserted first into the dispenser.

2.2.3 No components on the yellow shaded sides (see Appendix B CDS drawings) shall protrude farther than 6.5 mm normal to the surface from the plane of the rail.

> 2.2.3.1 Note: Please refer to the CIFP for recommended protrusion measurement technique.

2.2.4 Deployables shall be constrained by the CubeSat, not the dispenser. This requirement originates from requirements of most Launch Providers.

2.2.5 Rails shall have a minimum width of 8.5mm measured from the edge of the rail to the first protrusion on each face.

> 2.2.5.1 Note: An example is shown in Figure 4.

2.2.6 Rails should have a surface roughness less than 1.6 μm.

> 2.2.6.1 Note: This is typically met if the rail material is shown to be properly anodized. Otherwise, if the surface appears rough, more testing may be required.

2.2.7 The edges of the rails should be rounded to a radius of at least 1 mm.

> 2.2.7.1 Note: This is typically met using engineering drawings and manufacturer certification.

2.2.8 The ends of the rails on the +/- Z face shall have a minimum surface area of 6.5 mm x 6.5 mm contact area with neighboring CubeSat rails (as per drawing in Appendix B).

2.2.8.1 Note: If the CubeSat is not sharing the dispenser with another spacecraft, the Launch Provider may choose to waive this surface area requirement.

2.2.9 At least 75% of the rail should be in contact with the dispenser rails. 25% of the rails may be recessed.

2.2.10 Note: Table 1 shows the typical maximum mass for each U configuration.

Table 1: CubeSat Mass Specifications U Configuration Mass [kg]

1U:	2.00
1.5U:	3.00
2U:	4.00
3U:	6.00
6U:	12.00
12U:	24.00

2.2.10.1 Note: Masses larger than the one presented in Table 1 may be evaluated on a mission to-mission basis. Verify constraints with your dispenser provider or Launch Provider.

2.2.10.2 Note: Acceptable masses may vary depending on the dispenser capabilities. Verify capabilities with your dispenser provider.

2.2.11 The CubeSat center of gravity shall fall within the ranges specified in Table 2.

Table 2: Ranges of acceptable center of gravity locations as measured from the geometric center on each major axis

X Axis Y Axis Z Axis
1U + 2 cm / -2 cm + 2 cm / -2 cm + 2 cm / -2 cm
1.5U + 2 cm / -2 cm + 2 cm / -2 cm + 3 cm / -3 cm
2U + 2 cm / -2 cm + 2 cm / -2 cm + 4.5 cm / -4.5 cm
3U + 2 cm / -2 cm + 2 cm / -2 cm + 7 cm / -7 cm
6U + 4.5 cm / -4.5 cm + 2 cm / -2 cm + 7 cm / -7 cm
12U + 4.5 cm / -4.5 cm + 4.5 cm / -4.5 cm + 7 cm / -7 cm

2.2.12 The CubeSat structure should be made from aluminum alloy.

2.2.12.1 Note: Typically, Aluminum 7075, 6061, 6082, 5005, and/or 5052 are used for both the main CubeSat structure and the rails. If materials other than aluminum are used, the CubeSat developer should contact the Launch Provider or dispenser manufacturer.

2.2.13 Any aluminum CubeSat external surfaces, such as rails and standoffs that are in contact with the dispenser rails, shall be hard anodized to prevent any cold welding within the dispenser.

2.2.14 If a CubeSat shares a dispenser with another CubeSat(s), each CubeSat shall employ a mechanism to encourage separation from neighboring CubeSats within the dispenser.

2.2.14.1 Note: Any mechanism that will provide separation is acceptable. The common assumption with separation springs is that “stronger is better”. This is not always the case. Stronger separation springs can overpower the CubeSat dispenser deployment spring force during ejection and yield unpredictable separation characteristics, possibly re-contacting neighboring CubeSats. On the other hand, lower force springs may not have sufficient energy to separate the CubeSats the required amount. The general guideline is to select a separation spring with a max force less than 6.7 N (1.5 lbf) but with a stroke length greater than 2.5 mm (0.1 inches)

2.2.14.2 The separation mechanism shall not extend beyond the level of the standoff in a stowed configuration.

2.2.14.3 Note: The most common placement of the CubeSat separation mechanism is centered on the end of the two standoffs on the CubeSat’s –Z face as per Figure 5.

2.2.14.4 Note: A separation mechanism is not required for CubeSats that do not share a dispenser with another CubeSat(s).

Structural Requirements Excerpt from NanoRacks External CubeSat Deployer (NRCSD-E) Interface Definition Document (IDD) [NR-NRCSD-S0004]

4.1 Structural and Mechanical Systems Interface Requirements

The NRCSD-E is designed to house 6U of payloads in each of its six silos, for a total volume of 36U. It can accommodate any combination of CubeSats from 1U to 6U in length, up to a maximum volume of 6U in the 1x6x1U form factor. The only dimensional requirement that vary between the form factors is the total length (Z-axis dimension), which is specifically noted in the requirements herein. This section captures all mechanical and dimensional requirements to ensure the payloads interface correctly with the NRCSD-E and adjacent CubeSats.

4.1.1 CubeSat Mechanical Specification

1) The CubeSat shall have four (4) rails along the Z axis, one per corner of the payload envelope, which allow the payload to slide along the rail interface of the NRCSD as outlined in Figure 4.1.1-1.

2) The CubeSat rails and envelope shall adhere to the dimensional specification outlined in Figure 4.1.1-1.

Note: Any dimension followed by 'MIN' shall be considered a minimum dimensional requirement for that feature and any dimension followed by 'MAX' shall be considered a maximum dimensional requirement for that feature. Any dimension that has a required tolerance is specified in Figure 4.1.1-2. The optional cylindrical payload envelope (the "tuna can") must be approved for use by NanoRacks and special accommodations may be required if utilizing this feature.

3) Each CubeSat rail shall have a minimum width (X and Y faces) of 6mm.

4) The edges of the CubeSat rails shall have a radius of 0.5mm +/- 0.1mm.

5) The CubeSat +Z rail ends shall be completely bare and have a minimum surface area of 6mm x 6mm.

Note: This is to ensure that CubeSat +Z rail ends can serve as the mechanical interface for adjacent CubeSat deployment switches and springs.

6) The CubeSat rail ends (+/-Z) shall be coplanar with the other rail ends within +/- 0.1mm.

7) The CubeSat rail length (Z axis) shall be the following (+/- 0.1mm):

a. 1U rail length: 113.50mm

b. 2U rail length: 227.00mm
c. 3U rail length: 340.50mm
d. 4U rail length: 454.00mm
e. 5U rail length: 567.5mm
f. 6U rail length: 681 to 740.00mm

Note: Non-standard payload lengths may be considered. Any rail length differing from the above dimensions must be approved by NanoRacks and recorded in the payload unique ICA.

8) The CubeSat rails shall be continuous. No gaps, holes, fasteners, or any other features may be present along the length of the rails (Z-axis) in regions that contact the NRCSD-E rails.

Note: This does not apply to roller switches located within the rails. However, the roller switches must not impede the smooth motion of the rails across surfaces (NRCSD-E guide rails, fit gauge, etc.).

9) The minimum extension of the +/-Z CubeSat rails from the +/-Z CubeSat faces shall be 2mm.

Note: This means that the plane of the +/-Z rails shall have no less than 2mm clearance from any external feature on the +/-Z faces of the CubeSat (including solar panels, antennas, etc.).

10) The CubeSat rails shall be the only mechanical interface to the NRCSD-E in all axes (X, Y, and Z axes).

Note: For clarification, this means that if the satellite is moved in any direction while inside the NRCSD, the only contact points of the payload shall be on the rails or rail ends. No appendages or any part of the satellite shall contact the walls of the deployer.

11) The CubeSat rail surfaces that contact the NRCSD-E guide rails shall have a hardness equal to or greater than hard-anodized aluminum (Rockwell C 65-70).

Note: NanoRacks recommends a hard-anodized aluminum surface.

12) The CubeSat rails and all load points shall have a surface roughness of less than or equal to 1.6 μm (ISO Grade N7).

4.1.2 CubeSat Mass Properties

1) The CubeSat mass shall be less than the maximum allowable mass for each respective payload form factor per Table 4.1.2-1.

Note: The requirement driver for the CubeSat mass is the ballistic number (BN), which is dependent on the projected surface area of the payload on-orbit. The mass values in Table 4.1.2-1 assume no active or passive attitude control of the payload once deployed. If the CubeSat has attitude control capabilities or design features, then the operational ballistic number (BN) drives the mass requirement. If applicable, this shall be captured in the unique payload ICA.

2) The CubeSat center of mass (CM) shall be located within the following range relative to the geometric center of the payload: a. X-axis: (+/- 2cm) b. Y-axis: (+/- 2cm) c. Z-axis: i. 1U: (+/- 2cm) ii. 2U (+/- 4cm) iii. 3U (+/- 6cm) iv. 4U (+/- 8cm) v. 5U (+/- 10cm) vi. 6U (+/- 12cm)

4.1.3 RBF/ABF Access

1) The CubeSat shall have a remove before flight (RBF) feature that prevents the CubeSatfrom powering on when the inhibit switches are not depressed. The NRCSD-E has access ports only on the -X face of the dispenser. CubeSats in silos without the access panels should have timers implemented post RBF removal to prevent powering on of the spacecraft. The access port dimensions are defined in Figure 4.1.3-1.

Note: There is no physical access to the payload after integration into the NRCSDE besides what can be accessed from the access ports.

4.1.4 Deployment Switches

1) The CubeSat shall have a minimum of three (3) deployment switches that correspond to independent electrical inhibits on the main power system (see section on electrical interfaces).

2) Deployment switches of the pusher/plunger variety shall be located on the rail end faces of the CubeSat's -Z face.

3) Deployment switches of the roller/lever variety shall be embedded in the CubeSat rails (+/- X or Y faces).

4) Roller/slider switches shall maintain a minimum of 75% surface area contact with the NRCSD-E rails (ratio of switch contact to NRCSD-E guide rail width) along the entire Z axis.

5) The CubeSat deployment switches shall reset the payload to the pre-launch state if cycled at any time within the first 30 minutes after the switches close (including but not limited to radio frequency transmission and deployable system timers).

6) The CubeSat deployment switches shall be captive.

7) The force exerted by the deployment switches shall not exceed 3N.

8) The total force of all CubeSat deployment switches shall not exceed 9N.

4.1.5 Deployable Systems and Integration Constraints

1) CubeSat deployable systems (such as solar arrays, antennas, payload booms, etc.) shall have independent restraint mechanisms that do not rely on the NRCSD-E dispenser.

Note: Passive deployables that release upon ejection of the CubeSat from the NRCSD are considered on a case-by-case basis.

2) The CubeSat shall be capable of being integrated forwards and backwards inside of the NRCSD (such that the +/-Z face could be deployed first without issue).
Note: In general, the deployables should be hinged towards the front of the deployer to mitigate risk of a hang-fire should the deployables be released prematurely while the CubeSat is still inside the NRCSD.

4.1.6 Deployment Velocity and Tip-Off Rate Compatibility

1) The CubeSat shall be capable of withstanding a deployment velocity of 0.5 to 2.5 m/s at ejection from the NRCSD-E.

2) The CubeSat shall be capable of withstanding up to 5 deg/sec/axis tipoff rate.

Note: The target tipoff rate of the NRCSD-E is less than 5 deg/sec/axis. Additional testing and analysis are being completed by NanoRacks to refine and verify this value. If a payload has specific tipoff rate requirements, these should be captured in the unique payload ICA.

4.4.9 Materials

4.4.9.1 Stress Corrosion Materials
Stress corrosion-resistant materials from Table I of MSFC-SPEC-522 are preferred. Any use of stress corrosion-susceptible materials (Table II) shall be pre-coordinated with NanoRacks and documented in the ICA. Any use of Table III materials shall be avoided.

4.4.9.2 Hazardous Materials
Satellites shall comply with NASA guidelines for hazardous materials. Beryllium, cadmium, mercury, silver and other materials prohibited by SSP-30233 shall not be used.

4.4.9.3 Outgassing/External Contamination
Satellites 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. Since the satellite will be in close proximity to the ISS for anywhere from 21-90 days, a more thorough outgassing analysis is performed. This outgassing

analysis, performed by the ISS Space Environments group, uses ASTM 1559 data to characterize any potential material issues.

Note: A Bill of Materials (BoM) must be provided to NanoRacks to verify all materials requirements are met. The BoM shall be provided in the template specified by NanoRacks, and must include the vacuum-exposed surface areas of all non-metals. The ISS Space Environments Team screens the BoMs to ensure there are no external contamination concerns due to high-outgassing components. A bake-out is not required. The NASA website linked below is a useful source for obtaining outgassing data for materials.

https://outgassing.nasa.gov/