9 Spacecraft Structures

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9.1 Introduction

A satellite structure must fulfill various requirements. First of all, it must resist the loads induced by the launch environment (acceleration, acoustics thermal), met all the functional performances required on orbit such as dimensional stability for example, but it must also interface with some other subsystems such as : thermal control, optical components, electronic equipment, mechanisms, etc. In addition, the structure will be the skeleton used during the assembly process of these subsystems into the satellite and then it must provide very clean interfaces to each individual element in order to simplify the sequence of integration . Finally , the concept must be compatible with the standart manufacturing process and use standart components(sheet-iron, tube,...) every time it is possible.

All these constraints must be taken into acount in the preliminary structural design phase at the beginning of a project when the most important mechanical trade off are done (truss versus shell, materials, integrated panels versus modular platform). A good or a bad structure is determined at the very beginning step of a project .An important part of mission cost is the expense of insertion into space , related to the spacecraft mass.Consequently, A major issue in structural design is to minimize structural weight according to the required reliability level.

Spacecraft structures are mainly divided in two categories :

The ***Primary*** structure or main structure, whose purpose is to transmit loads to the base of the satellite through specifically design components (central tube, honeycomb platform, bar truss, etc.). This structure provides the attachement points for the payload and the associated equipments. Failure of the primary structure leads to a complete collapse of the satellite.

-The ***Secondary*** structures, such as baffle, thermal blanket support and solar panel must only support themselves and are attached to the primary structurewhich guaranties the overall structural integrity. A secondary structure failure is not a problem for the structural integrity, but it could have some important impacts on the mission if it alters the thermal control, the electrical continuity, the mechanisms or if it crosses an optical path.

For the new generation of large satellites, we must consider a third type of structure: ***Flexible appendages*** such as antenna reflectors and solar arrays. These strucures have generally low resonant frequencies which interact directly on the dynamic behaviour of the satellite and require a special care for design.

Finally, some spacecraft structures are more complex than the ones described above, and cannot be described with general rules due to their uniqueness and particular requirements . Among these are the manned spacecraft structures (orbiter and space station) and the future lunar outposts. [Giraudbit 1989]

9.2 Design Philosophy

Spacecraft structural design is a complicated iterative process that involves materials selection, configuration, analysis, and verification testing. Structural design is very dependent on the design requirements of other subsystems such as thermal, propulsion, communications, and power. The design process starts at a conceptual stage and design specifications are based on the mission requirements (see Figure 9.2.1). These specifications may include accommodation of payload and systems, launch requirements, environmental protection, thermal and electrical paths, and stiffness. The challenging tasks of structural design are the extreme mass efficiency and high reliability requirements of the structure.

There are a wide variety of shapes used in satellite structures whose mass, volume, and other structural characteristics are well known, making new design and testing unnecessary. The lack of aerodynamic drag in space allows the use of cubic, cylindrical, octahedral, and polyhedral configurations providing high rigidity and volume capacity. Spin stabilization of spacecraft requires symmetry and appropriate roll-to-pitch inertia ratios for stability. Relying on known designs and less exotic materials will reduce costs.

Modular construction of space vehicles allows quick and cost-effective assembly line production, and increases accessibility and maintainability. Parts are interchangeable and easily accessed. Off the shelf components can be put together to fulfill design specifications. The drawback comes in the form of a weight penalty due to the use of special interfaces. It can be viewed that modular construction is design philosophy as well as a construction technique(see the European Polar Platform project). A non-modular construction has an advantage in weight saving aspects. In a non-modular construction of a space vehicle, one can customize a space vehicle to mass or environmental specifications. Exotic materials can be employed in a non-modular construction more beneficially than in modular construction.

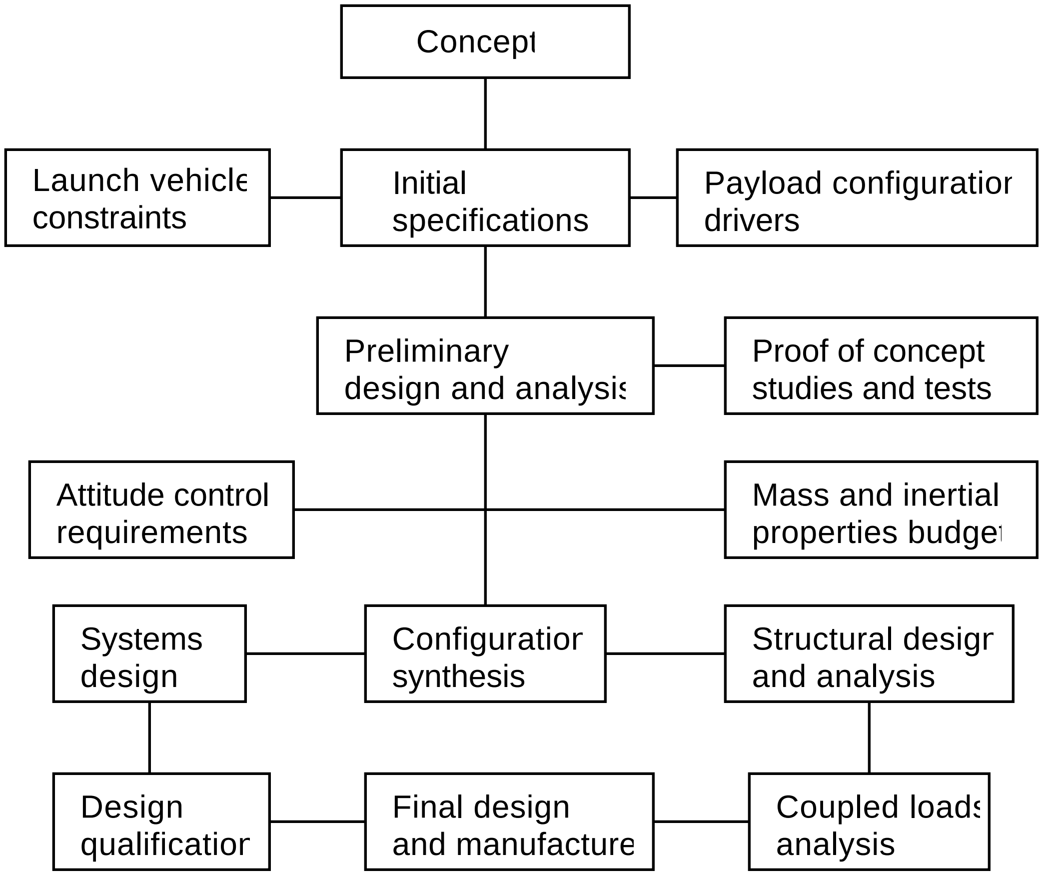


Figure 9.2.1 Design methodology [Fortescue 1991]

Factors of safety for spacecraft structures vary for a number of reasons. Obviously, manned spacecraft require higher factors of safety than do satellites or probes. Structural redundancy is also considered when factors of safety are calculated. If failure of a structural member is non-catastrophic and its load is taken by another member, the factor of safety specified for that member may be relatively low. Mission redundancy also permits lower factors of safety. An example of this is a multiprobe mission where several identical probes are released and failure of one of them is not considered a mission failure.

The typical factors of safety for space structures (unmanned flights) are given in the following list:

a) Test qualified structures

Qualification level	: Flight x 1.45
Yield	: Qualif. x 1.1
Ultimate	: Qualif. x 1.25

b) Computed structures only

Yield	: Flight x 2
Ultimate	: Flight x 3

c) Pressure tanks (fracture analysis)

Yield	: nominal x 1.5
Ultimate	: nominal x 2.0

Of course these values may be changed during the project. Loads and material strengh are subject to uncertainties, often described in terms of mean value and standart deviation σ (Gaussian distribution for instance).On the following example, the factor of safety J will be used to calculate the probability of failure for a load defined at 2σ and a strengh defined at 3σ:

$L_n = \underline{L} + 2\sigma_L$ Nominal load

$S_d = \underline{S} - 3\sigma_s$ Design or admissible strengh

$S_d = JL_n$ J factor of safety

Let $X = S - L$. The probability of failure is the percentage of case where Load > Strength or mathematically P(X≤0). In the case of a gaussian distribution:

$$P(X \le 0) = \frac{\pi}{2}\int_{-\infty}^{K} e^{\frac{u^2}{2}} du \quad \text{with } K = \underline{X}/\sigma_X$$

It can be shown that :

$$K = \frac{\underline{J} - 1}{\sqrt{\left(\frac{\sigma_L}{\underline{L}}\right)^2 + \frac{\sigma_s}{\underline{S}} * \underline{J}^2}} \quad \text{with } \underline{J} = J\frac{1 + 2\frac{\sigma_L}{\underline{L}}}{1 + 3\frac{\sigma_S}{\underline{S}}}$$

Now, if we consider the following deviation (gaussian distributions):
$\frac{\sigma_L}{\underline{L}} = 0.1$ and $\frac{\sigma_S}{\underline{S}} = 0.05$ and a safety factor of J = 1.1, we can deduce the value of K and then the failure probability through a numerical table of Gauss integral.

Finally K = 4.37 and $P_{failure} = 10^{-5}$

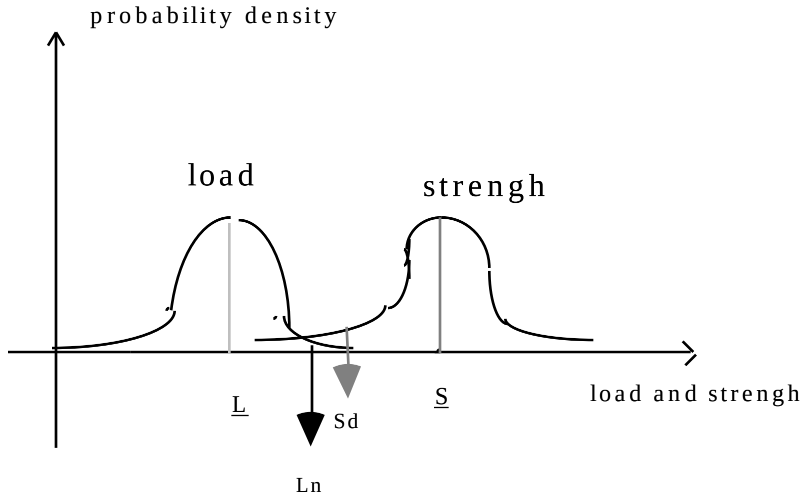


Figure 9.2.2 Warner Diagram

9.2.1 Design Rules Summary

The constraints to take into account at the beginning could be summarized as follows:

- Good knowledge of the environment (launch and orbit)

- Simple structural elements with simple function,

- Modularity, simple acess and assembly,

- Easy analytical predictions,

- Simultaneous concern of static, dynamic, and thermoelastic problems,

- Growth potential,

- Mechanical decoupling : isostaticity , decoupling of primary and secondary structure, simple interfaces,

- Taking into account the thermal requirements (thermal and structural team have to work very closely all the time),

- Local interface flexibility to consider the global stiffness budget,

- Precise choice of materials: specific strengh and stiffness, outgasing, thermal conductivity, electrolytic corrosion, electromagnetic compatibility, availability, cost, radiation resistance,, influence of humidity,...

- Handling hard points,

- Manufacturing process in accordance to unitary production.

These common sense rules generally lead to simple structures with very competitive costs.

9.3 Space Environment

The space environment poses a variety of extreme conditions that can seriously degrade or even cause failure of many materials and structures. Sublimation and outgassing, thermal and radiation effects, and oxidation must all be considered in the structural design. For craft in low Earth orbit, the environment is particularly harsh.

9.3.1 Vacuum Properties

The vacuum of space surrounding Earth ranges from a pressure of 1.3 x 10^{-7} kPa at 200 km, to less than 1.3 x 10^{-12} kPa beyond 6,500 km. Under these conditions, polymers may decompose and metals sublimate (lose molecules). The rate of sublimation is given by:

$$G = 5.04x10^{3} P \sqrt{\frac{M}{T}}$$

where:

G = amount of sublimated material, grams/cm^3-day
P = Vapor pressure of evaporating material, mm Hg
M = Molecular weight of the material
T = Absolute temperature, K

Sublimation can cause the growth of whiskers, which can create short circuits, or lead to deposits on optical and thermal systems, which may ruin data transmission or overheat and destroy the craft. Certain materials have high sublimation rates at low temperatures (less than 200° C), and therefore should be avoided. Zinc, tin (used in electrical solder), magnesium, and cadmium are examples. Also, composite matrices have a higher vapor pressure than metals, thus having a higher sublimation rate. This tends to make composites less desirable for long duration missions, though this may improve with new composite technologies.

Outgassing, the release of absorbed gasses by a structural surface in a vacuum, is a problem common to most materials. Released particles will settle on other parts and can cause malfunctions (destruction of thermal coating, contamination). Polymeric materials must possess non-outgassing characteristics for spacecraft applications. This problem can be reduced by "bake-out" processing, putting the material in a vacuum at high temperature. Because most lubricants outgas in space, friction is greatly increased and some materials may undergo cold molecular welding. Finally, composite materials have a high absorption rate of humidity which can cause serious outgassing problems and lower their structural performances.

9.3.2 Temperature Concerns

Related to the sublimation rate (as T increases, so does G), and vitally important are temperature variations and extremes in space. Without an atmosphere, thermal energy can only be transferred through conduction and radiation, with temperatures ranging from -160° C to +180° C. This will vary for each spacecraft, depending on its spin rate and the type of thermal control system. Passive systems make use of surface absorptance/emittance (α/ε) properties. For example, anodized aluminum or white surfaces have low (α/ε) ratios, while black objects have a ratio of about unity. Solar absorbers, such as polished metal, have (α/ε) ratios greater than unity.

The highest temperatures affecting structural design typically arise from atmospheric entry or robust propulsion systems. These conditions require the use of special materials, tailored insulation, or both. The Space Shuttle uses tiled insulation on its exposed aerodynamic surfaces. Most of these areas have normal aluminum skin-stringer or honeycomb panel beneath, though the most critical locations (e.g., stagnation points) use titanium.

For spacecraft without these two causes of extreme heating, the temperature conditions are relatively benign. Cold environments, such as among the outer planets, will generally increase the yield strength, tensile strength, and Young's modulus of a material. Effects on ductility and toughness, however, vary with the material. This requires that brittle failure by shock be examined. Cryogenic fuel storage also necessitates a material with good low-temperature properties.

Figure 9.3.1 presents the temperature limits of several structural materials, according to present and projected technology.

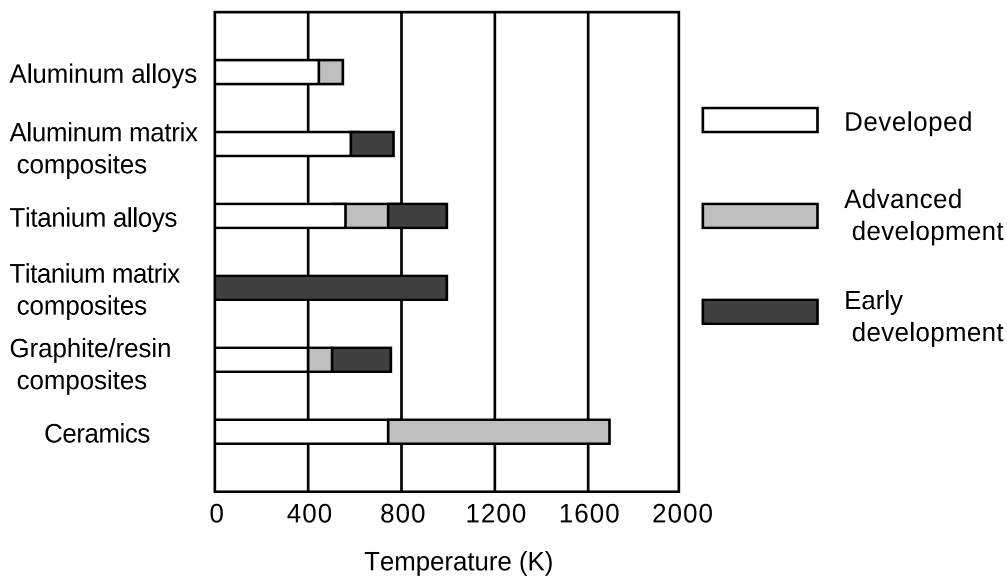


Figure 9.3.1 Material temperature limits.

9.3.3 Radiation Effects

Electromagnetic and particle radiation, such as protons and electrons from radiation belts, solar emissions, and cosmic radiation, can remove structural material. The amount is usually no more than 1 mg/cm^2, which has no serious effect on the design of most structures. Thin films, however, such as a solar sail, must account for this degradation. Radiation also reduces the ductility of most materials. This must be anticipated for long-duration or high-exposure missions.

9.3.4 Oxidation Effects

In low Earth orbit, high energy, neutral atomic oxygen atoms (ATOX) and ionizing radiation can severely degrade polymeric materials by reacting with their organic molecules.

Similarly, thin organic films, advanced composites and metallized surfaces can suffer from oxydation effects.For instance, Kapton erodes as much as 2.8 μm for every 10^{24} atoms/m^2 of atomic oygen fluence(the fluence over a period T is equal to T x density of ATOX x satellite velocity). At approximately 200 km (125 miles), their concentration varies from $2x10^9$ to $8xl0^9$ atoms/cm^3, depending on solar activity. They can strike a spacecraft with a relative velocity greater than 9 km/s. ATOX can also lead to further outgassing, and this effect can not be eliminated before launch. However, research is being done to characterize and predict the outgassing behavior of various materials. Development of coatings such as silica can reduce this effect.

9.3.5 Magnetic Properties

The magnetic field produced by the Earth requires that earth orbiting spacecraft must use a non-magnetic material for most of its structure. The alternative is to accept the magnetic dipole moment induced by motion through the magnetic field, causing an orientation change unless resisted by attitude control mechanisms.

9.4 Materials

9.4.1 Properties

The selection of materials for structural applications is a crucial step in the design process. A simple selection based on strength / density is not sufficient. It is important to consider many other properties such as stiffness, stress corrosion resistance, fracture toughness, fatigue resistance(minor issue for all short-life spacecraft), thermal characteristics, sublimation, electrical and magnetic properties, ease of manufacture, availability and cost.

9.4.1.1 Specific Strength (S_y / ρ)

The specific strength of a material is defined as the stress that causes a 0.2% elongation at the elastic limit divided by the the density. This parameter is useful for preliminary comparisons between materials. Titanium alloy metals and fiber-reinforced composites (kevlar,HT carbon) typically have high specific strengths. Composites materials generally present a high specific strengh ratio if they are unidirectional. The performances of an isotropic composite material are much lower (30%), which should be taken into account during the design.

9.4.1.2 Specific Stiffness (E / ρ)

The specific stiffness of a material is defined as the Young's modulus divided by the density. This parameter is useful to select an efficient material with respect to mass. Table 9.4.1 shows three common load cases and their respective material efficiency criteria. These material efficiencies are given for typical structural materials in Table 9.4.2. Note that aluminum and titanium alloys have similar structural efficiencies for all three load cases but that titanium has a higher specific strength.

Figure 9.4.1 Typical Stress / strain Diagram for Ductile and Brittle Materials [Larson 1992]

9.4.1.3 Thermal Characteristics

Thermal conductivity and thermal expansion coefficients are critical parameters to consider when selecting a material for a structural application. Thermal conductivity is important as thermal conduction or insulation is often a secondary function of the structure of a spacecraft. The thermal expansion coefficient is also an important parameter. Large thermal stresses can be induced when two materials with differing thermal expansion coefficients are used in the same structure. Also, it may be desirable to minimize thermal expansion for delicate instruments such as space telescopes. This can be done with composite materials quite effectively as a structure can be designed to use the directionally dependent positive and negative expansion coefficients of graphite / epoxy. A net expansion coefficient of zero for a structure is theoretically possible within certain temperature ranges(Kevlar 49 and UHM carbon are used in that way).

9.4.1.4 Fracture and Fatigue

Fracture and fatigue resistance becomes more important as safety factors are reduced and as structural efficiency is increased. Microcracking is assumed to exist in all structures and the designer must ensure that failure from these flaws does not occur throughout the service life of the structure. Every crack tip is a stress concentration and cracks will propagate if local stresses are high enough. The critical crack length is defined as the length of the largest crack that will not propagate at a given stress level. The designer must calculate that the design and material can withstand non-catastrophic cracking up to a certain size. Also, non-destructive testing techniques must be used to demonstrate that no cracks above the critical size exist before launch. These tests are usually done by ultrasonic sounding(reflection and transmission), X-ray, thermography, holography.However, it should be pointed out that for most commercial spacecraft, fatigue is not a dimensioning parameter because of the relatively short duration of the mission.

9.4.1.5 Ease of Manufacture

It is important that the designer consider the manufacturing process when designing spacecraft structures. Some composite material structures may be prohibitively expensive to manufacture and could be made of less exotic materials and geometries while still performing the same function. Some materials such as beryllium and aluminum-lithium alloys can present toxic or dangerous conditions during manufacture. Late modifications often occur during assembly of individual components. The designer should allow for such modifications to occur at the assembly level if necessary.

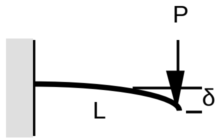
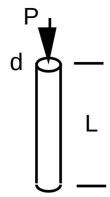
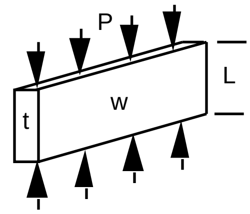
Loading Configuration	Beam Deflection	Strut Buckling	Panel Buckling
			
Characteristic Equations	$\delta = \frac{PL^3}{3EI}$ $I = \frac{wt^3}{12}$ $mass = Lwt\rho$	$P_e = \frac{\pi^2 EI}{L^2}$ $I = \frac{\pi d^4}{64}$ $mass = \frac{\pi d^4}{4} L\rho$	$P = const \frac{E}{1-\sigma^2}\left(\frac{t}{w}\right)^2 tw$ $mass = Lwt\rho$
Efficiency $\left(\frac{load}{mass}\right)$	$\frac{P}{mass} = const\left(\frac{\delta t^2}{L^4}\right)\frac{E}{\rho}$	$\frac{P}{mass} = const\left(\frac{P_e}{L^4}\right)^{1/2}\frac{E^{1/2}}{\rho}$	$\frac{P}{mass} = const\left(\frac{P_e}{L^4}\right)^{2/3}\frac{E^{1/3}}{\rho}$
Material Efficiency	$\frac{E}{\rho}$	$\frac{E^{1/2}}{\rho}$	$\frac{E^{1/3}}{\rho}$
Loading Coefficient	$\frac{\delta t^2}{L^4}$	$\left(\frac{P_e}{L^4}\right)^{1/2}$	$\left(\frac{P_e}{L^4}\right)^{2/3}$

Table 9.4.1 Structural configurations for material selection.

Material	Density ρ (kg/m^3)	Young's modulus E (GPa)	Yield strength Sy (MPa)	$\frac{E}{\rho}$	$\frac{E^{1/2}}{\rho}$	$\frac{E^{1/3}}{\rho}$	$\frac{S_y}{\rho}$	Thermal expansion (μm/m K)	Thermal Cond. (W/m K)	Fracture toughness (MPa-m)	Fatigue strength (MPa)
Aluminum alloy											
6061 T6	2800	68	276	24	2.9	1.5	98.6	23.6	167	186	97
7075 T6	2700	71	503	26	3.1	1.5	186.3	23.4	130	24	159
Magnesium alloy											
AZ31B	1700	45	220	26	3.9	2.1	129.4	26	79		
Titanium alloy											
Ti - 6Al - 4V	4400	110	825	25	2.4	1.1	187.5	9	75	500	
Beryllium alloys											
S 65 A	2000	304	207	151	8.7	3.4	103.5	11.5			
S R 200E			345								
Ferrous alloys											
INVAR		150						1.66			
AM 350	7700	200	1034	26	1.84	0.8	134.3	11.9	40/60	550	
304L Ann	7800	193	170	25	1.8	0.7	21.8	17.2			
Fiber Composites											
Kevlar 49 0deg	1380	76	1240	55	6.3	3.1	898.5	-4			
/epoxy 90deg	1380	5.5	30	4	1.7	1.3	21.7	57			
Graphite 0deg	1640	220	760	134	9.0	3.7	463.4	-11.7			
/epoxy 90deg	1640	6.9	28	4.2	1.6	1.16	17.1	29.7			

Table 9.4.2 Material properties

9.4.2 Metals

9.4.2.1 Aluminum

The alloys of aluminum represent the majority of spacecraft structural materials. A combination of high stiffness to density ratio, excellent workability, non-magnetism, moderate cost, high ductility, high corrosion-resistance, and availability in numerous forms makes it the best choice for most uses. Its low yield strength is the only appreciable disadvantage.

Aluminum-lithium (Al-Li) alloys have the potential of reducing launch vehicle weight by as much as 30%, and are being produced by several manufacturers. These materials can have a tensile strength over 100 ksi as well as a cryogenic strength greater than any other aluminum alloy, and have high weldibility. This is an important consideration for cryogenic fuel tanks.

Use of Al-Li sheet as a laminate with a fiber/epoxy has also been developed. The fiber/epoxy is sandwiched between layers of aluminum, combining the strength and resistance to fatigue of fiber with the benefits of the alloy.

9.4.2.2 Steel

Most structural steels are penalized by their magnetic properties, although austenitic stainless steels are non-magnetic. Unfortunately, the stiffness to density ratios for austenitic steels are lower than those of aluminum alloys. Nonetheless, these materials can still be utilized for their high strength in instances where titanium is not desirable, perhaps for machining or temperature reasons.

9.4.2.3 Titanium

Titanium, a non-magnetic material is used in many applications where aluminum does not possess the required strength. Though somewhat more difficult to machine, it has a substantially greater yield strength, a higher stiffness to density ratio, and is particularly suitable for low-temperature applications (e.g., cryogenic fuel storage). At high-temperatures, titanium is surpassed by steels, though it still exceeds aluminums capabilities. Research may greatly increase the ease of manufacturing complex titanium components.

A relatively new class of titanium based materials are the intermetallic titanium alumides, or tialuminides. These low density materials exhibit high strength at temperatures greater than 700° C and resist oxidation at all temperatures. Primary uses are as a composite matrix material and in honeycomb structures. Unfortunately, tialuminides react poorly with hydrogen, becoming brittle. This limits their utility in vehicles that use hydrogen to actively cool the airframe unless protective coatings are used.

9.4.2.4 Magnesium

Though its stiffness to density ratio is close to aluminum, this material and its alloys are prone to brittle fracture. This reduces its applications to those where its barely higher yield strength may be of some use, or where its good low-temperature behavior is paramount. As already mentioned, magnesium sublimes relatively quickly in vacuum (0.04 in/yr at 180° C). Exposure to temperatures less than 250° C can cause it to lose static strength, although electroplating or coatings can prevent this.

9.4.2.5 Beryllium

With a density approximately 60% of aluminum and a stiffness to weight ratio six times better than aluminum or titanium, this material has many potential applications. Being stiffer than other materials it can be useful in avoiding resonant frequencies that may occur between a satellite and its booster during launch. It is non-magnetic, has a high elastic modulus (44 Msi), and a high yield strength. Using beryllium instead of aluminum can reduce weight significantly, and its high thermal conductivity makes it an excellent choice for components that will conduct heat. It is also non-reactive with hydrogen.

However, beryllium is extremely anisotropic and sensitive to damage, and is twice as brittle as aluminum. Its relatively low fracture toughness at cryogenic temperatures is a potential drawback, but development of beryllium-aluminum alloys may improve this. It also needs special facilities and tools for machining due to the toxicity of its dust, and is thus very expensive.

9.4.3 Composites

Composite materials are quickly becoming the material of choice for aerospace applications. They possess stiffness to weight ratios beyond all metals, making them useful for damping unwanted vibrations. With a negative axial coefficient of thermal expansion, they allow for structures that will not deform in the temperature extremes of space. Their thermal conductivity exceeds copper, and so they also provide lightweight thermal management and heat sinks. Where stiffness is critical, as in telescopes, antennas, and reflectors, carbon fiber composites are another natural choice. Metal matrix, Carbon-Carbon, and Ceramic-Matrix composites are best for high temperature applications, such as re-entry vehicle skins, since they can withstand temperatures in excess of 2500° F. with no active cooling.

However, there are some drawbacks to these materials. Effective oxidation coatings must developed, as well as manufacturing techniques for large scale structures. Grounding of electrical systems is done by adding conductive strips, which increase the mass of the structure. Nicks and dents that can be repaired or ignored in structural metals can destroy the integrity of the fibers and render the composite unusable. In space, with little or no inspection and maintenance, and where failure of primary structure can have devastating consequences, composites are typically judged too unreliable for use as more than secondary structure. But judicious use of composites in secondary structure can still result in much mass savings.

Another concern of laminated composites is their reaction to temperature changes. Uniform changes can induce substantial internal stresses caused by different expansion rates between the fibers and matrix. Temperature differentials can produce more pronounced warping than in an isotropic material.

Hereafter are some applications of fiber reinforced composites:

a) Glass
   High strengh, low cost, fatigue insensitive.
   - Solid rocket engine casing

- Pressure vessels
- Thermal decoupling

b) Kevlar
   High strengh, low cost, impact resistant, radio frequency transparency.
   - Solid rocket engine casing
   - Pressure vessels
   - Shrouds
   - Bi-grille reflector antenna

c) HT-CFRP
   High strengh, fatigue insensitive, low cost.
   - Launcher interstages

d) HM-CFRP
   Stiffness and strengh, reasonable cost.
   - Optimized structure
   - Solar array
   - Antenna reflector

e) UHM-CFRP
   High stiffness, low CTE, very high cost (10 times HT cost).
   - Sophisticated thermo-elastic stable structure
   - Telescope
   - Antenna reflector and tower
   - wave guide

9.4.4 Ceramics

Currently, ceramic materials are too brittle for use in primary structure. However, their excellent ability to withstand high temperatures have led to applications as turbine blades. Research may extend the high-temperature performance of ceramics considerably, though its use will probably be limited to secondary structure.

9.5 Structural Elements

The structural elements most commonly used for spacecraft are columns, frames, trusses, plates, and shells. Pressure vessels are also used, especially for fuel tanks.

Columns, including those that are the components of a truss, are often necessitated by the axial nature of propelled spacecraft. Weight concerns, meanwhile, demand small cross-sections. The result are columns designed close to the limits of instability. To withstand axial loads with no eccentricity, the best cross-section must be axisymmetric and possess a large radius. Round tubular sections are normally used because they provide adequate torsional stiffness and they are less massive than solid sections. In cases of large columns, the tube itself may rely on the components it protects to provide some lateral support.

A simple and useful calculation is the determination of the margin of safety for a circular cylinder:

1) Reference stress

From the theory of shells : $\sigma_{ref} = \frac{1}{\sqrt{3(1-\nu^2)}} \frac{Et}{R}$

with E : Young's modulus
t : Thickness
R : radius
ν : Poisson's ratio

2) Compression and bending critical stresses

- Compression:

The critical stress is given by $\sigma_c^* = \gamma_c \sigma_{ref}$ where γ_c is a correlation factor for compression given by :

$$\gamma_c = 1 - 0.901\left(1 - \exp\left[-\frac{1}{16}\sqrt{\frac{R}{t}}\right]\right)$$

- Bending:

The critical stress is given by $\sigma_b^* = \gamma_b \sigma_{ref}$ where γ_b is a correlation factor for bending given by :

$$\gamma_b = 1 - 0.731\left(1 - \exp\left[-\frac{1}{16}\sqrt{\frac{R}{t}}\right]\right)$$

3) Margin of safety for a combined load case

The two critical stresses σ_c^* and σ_b^* are determined as above. The actual stresses are determined as shown on the following figure :

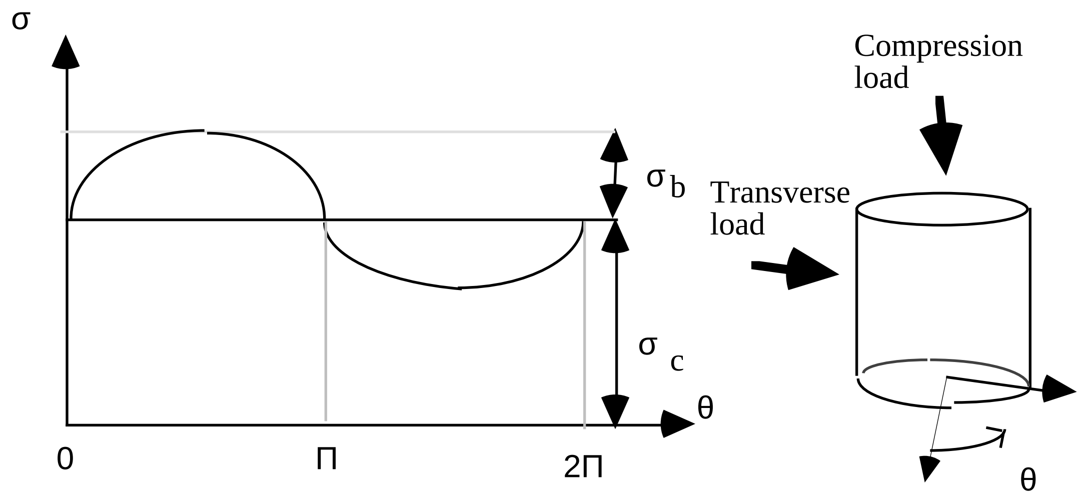


Figure 9.5.1 Axial Stress Distribution in a Cylinder Subject to a Combined Bending and Compression Loading

The recommended interaction equation for the margin of safety in the case of combined compressive load and bending is given by :

$$\left[\frac{\sigma_b}{\sigma_b^*} + \frac{\sigma_c}{\sigma_c^*}\right]^{-1} - 1 = \text{M.S.}$$

where σ_c and σ_b are respectively the pure compression actual stress and the pure bending actual stress.

Interpretation:					
		MS	$<$	0	Failure
0	$<$	MS	$<$	0.5	Optimum design
0.5	$<$	MS	$<$	1.5	standart but efficient design
1.5	$<$	MS	$<$	∞	Non-efficient design

Trusses and frames are similar. Both distribute a load throughout a sparse arrangement of discrete elements. Trusses are constructed with fittings at the ends of the members, while frames are molded or otherwise formed as one piece. Various fasteners can attach smaller frames together, making a large one. Frames are generally used for smaller spacecraft (satellites and probes), while trusses are more applicable to large vehicles (manned structures and launch vehicles).

In plate and shell configurations, a skin-stringer or honeycomb sandwich configuration is able to transmit loads while minimizing weight. The honeycomb section is slightly more efficient than skin stringer and less prone to buckling. Honeycomb sections are made of two thin (0.5-1.0 mm) aluminum or composite sheets bonded on either side of a honeycomb core. These sections provide greater micrometeorite protection than skin stringers, especially if the core is a low-density solid (e.g., plastic foam). Additionally, honeycomb sections exhibit high damping. However, the fabrication of a honeycomb plate is more complicated than a skin-stringer because fittings, inserts, holes and edgings must be carefully designed. The panel's strength depends on the core's resistance to damage and the adhesive between the core and plates, which makes crack propagation an important concern. Also, temperature differentials across the honeycomb panel's cross section can cause substantial internal stresses. In practice, skin-stringer and honeycomb panels are both used extensively, the former especially for irregular shells, such as nose caps and aerodynamic surfaces.

Analytic calculation of structural plates made out of honeycomb can easily be derived from isotroic elements formula. The actual thickness and Young’s modulus of the honeycomb are replaced in the formula by “equivalent” thickness and Young’s modulus depending on the honeycomb thickness and on the thickness of its faces.It should also be noticed that sandwich have several buckling modes that should be checked during the design process: intracellular buckling, face wrinkling and shear crimping.

Another crucial spacecraft structure are the fuel tanks. Propellant tanks are usually constructed of a thin gauge metal, and unlike other structural parts, can not be riveted together. Instead, welding or filament winding is required. The propellant itself is necessary for structural stability, possibly supplemented by stringers or struts. Additionally, baffles may be required to prevent the sloshing of the fuel. In general, the fuel tank mass may be estimated to be 10% of the total propellant mass.

The most commonly tanked fuels are hydrogen (for its high specific impulse (I_{sp}) and use in nuclear rockets), helium (used as a pressurizer), oxygen (an oxidizer and ECLSS component), and hydrazine (for its high storability). The first three fuels do not react with normal structural materials, but they must be stored at extremely low temperatures. Thus, the wall of the tank will also be subject to the low temperature, so design must anticipate brittle behavior. To minimize the flux of thermal energy to the stored liquid, large tanks with high volume to surface area ratios, such as spheres, are desirable for all cryogenic fuels. Additionally, foam-type insulation is typically applied to the exterior of the tank. Hydrogen is especially sensitive to temperature and will boil off if the insulation is not adequate. Also, spontaneous conversion from ortho- to para-hydrogen will induce losses, unless a catalyst is used to convert the fuel before storage. One material that cannot be used in conjunction with hydrogen is titanium and its alloys. Hydrazine does not require cryogenic storage, and thus does not lose large quantities to boil-off. For this reason it is preferred for landers and other vehicles where propellant storability is an important concern. Hydrazine can be stored in tanks constructed of most structural materials, though magnesium and some aluminum alloys are not acceptable.

9.6 Structures Subsystem Interfaces

Based on the information provided in the previous sections, a general structures development/design process can be formulated. Figure 9.6.1 shows the structures subsystem interface document. This is divided into two separate categories, Inputs and Outputs, to allow for easy referencing. The following sections discuss the many design considerations that are used in the design of a spacecraft structure.

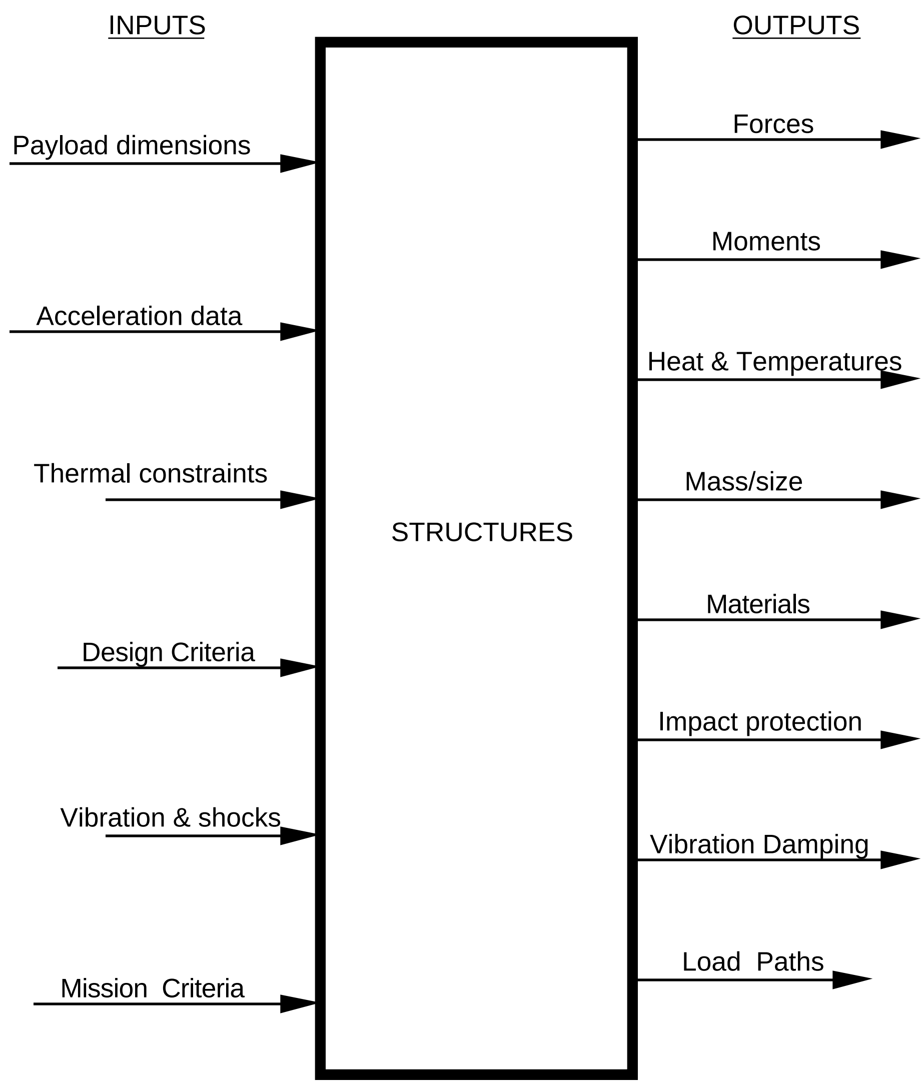


Figure 9.6.1 Structures subsystem interface.

9.6.1 Launch Constraints

9.6.1.1 G Loadings

G loadings are the weight related accelerations a space vehicle experiences for an extended period of time and that can be considered as quasistatic. These differ from vibrations and shocks, which are accelerations of shorter period and are several orders of magnitude greater. An example of typical g loadings is shown in Table 9.6.1.

G loadings during launch depend greatly on the propulsion system in use. Solid motors are usually not throttleable and are often designed to deliver large amounts of thrust in a relatively short burn. Liquid fueled boosters ability to throttle up to full thrust can alleviate unnecessary loadings that are experienced when solid rocket motors are used. Additionally, the ability of throttling may give a weight advantage that can offset the added weight of equipment related to throttling thrust. This is an important consideration since load sensitive payloads require special packing devices.

Nuclear fueled and related engines have a long duration burn when compared to chemical motors, and will not have as high g loadings. This is also true for spacecraft which have throttleable motors for orbit transfers. A possible negative result of the use of these types of engines is the inducement of material fatigue due to repetitive thrusting combined with extended periods of thrusting.

Finally, payload fairing are typically not jettisoned until the dynamic pressure drops to 0.5 N/m^2 (at this altitude the aerodynamic pressure is equal to the solar radiation pressure).

Flight Event	Longitudinal Axis	Lateral Axis
Maximum Dynamic Pressure	-3 g (compression)	±1.5g
Second engine cut off (SECO)	- 7 g (compression)	±1.0g
Thrust Lift Off	+2.5g (tension)	±1.0g

Table 9.6.1 Flight Limit Quasi Static Loads (Ariane IV) at the CG of the payload

9.6.1.2 Ignition, cut off (chugging) and Stage Separation Shocks

The primary source of loads occur during the launch phase and during stage separations. These loads can be reduced with the use of throttling in liquid fuel rockets. The actual loads applied to the payload depends on the particular launcher used, but the expected flight limit loads are available through the launcher agency. The spacecraft should be designed with of factor of safety between 1.25 and 1.4 in relation to these flight limit loads. Shocks may also arise in orbit from pyrotechnics and appendage deployments. Although these are small when compared to the shocks encountered during motor burn, the orientation of these minor shocks may be serious. For example, a component may be designed to withstand launch shocks in the longitudinal direction, but shocks from other events may be oriented differently from the longitudinal axis. A material's maximum tensile strength could be important in considering these effects, but many other kind of failure are likely to be induced by onboard shocks(electrical failure, solar panel release failure, local geometrical deformation). The actual effects of these shocks can be estimated with an impact loading and transient analysis and, if necessary, the design should be modified to handle these effects.

9.6.1.3 Vibrations and Resonant Frequencies

The natural frequency of a spacecraft must not match those of the launch vehicle if strong and potentially destructive coupling is to be avoided. Typically, it is desirable for the spacecraft to have resonant frequencies above 30 Hz (to avoid coupling with POGO) in the longitudinal axis and above 10 Hz laterally. Meeting the minimum allowable frequency usually allows for simpler and lighter designs. A rule is to consider that dynamic decoupling is achieved between the satellite and the launcher when the first frequency of the satellite is than the first frequency of the launcher multiplied by $\sqrt{2}$ .This rule is also valid for the decouling between payload and main platform on the large satellites. Fatigue induced yielding could also be a major design factor when harsh vibrational environments are encountered. Table 9.6.2 lists the goal structural natural frequency bands for spacecraft being launched by the Ariane IV and Delta launch vehicles.

Axis	Launcher	Spacecraft	Internal Equip.
Ariane IV			
Thrust	>31	31-60	<80
Lateral	>10	10-40	<60
Delta			
Thrust	>35	35-65	<80
Lateral	>15	15-45	<60

Table 9.6.2 Design-goal structural natural frequency bands (Hz) [Fortescue 1991]

Random vibrations and acoustic vibrations also occur at launch and should also be considered in the design of any spacecraft. Random vibrations are generated by mechanical parts in movement,gust of wind, combustion phenomena, and structural elements excited by the acoustic environment. These vibrations are transmitted to the spacecraft by the launch vehicle structure. Acoustic vibrations are generated by engine noise, buffeting, and aerodynamic noise. These vibrations propagate primarily through the atmosphere within the spacecraft at launch, but are eventually transmitted as high-frequency structure-borne noise during ascent. Tables 9.6.3 and 9.6.4 list the flight levels of random and acoustic vibrations for the Ariane IV.

Frequency (Hz)	Level (PSD)	Overall (rms)
5 - 150	+6 dB / oct to $0.04g^2$ / Hz	7.3 g
150 - 700	$.04g^2$ / Hz	-
700 - 2,000	-3dB / oct	-

Table 9.6.3 Random Vibration Flight Limit Environment (Ariane IV)

Octave Band (Center Frequency, Hz)	Acceptance Level (0 dB ref.)
31.5	114
63	120
125	131
250	136
500	139
1,000	133
2,000	128
4,000	121
8,000	120
Overall	142

Table 9.6.4 Acoustic Vibration Flight Level (Ariane IV) in fairing

9.6.1.4 Dynamic Loadings

In addition to the thrust loads applied at launch and stage separation, the spacecraft will also experience dynamic loadings due to lift, drag, and weight. Lift from a cylindrical launch vehicle will always be a lateral force. Since hoop stresses in a thin skinned vehicle will cause buckling more easily than the longitudinal stresses, they must be minimized to lighten the vehicle and prevent undesirable loads.

The drag force on a launch vehicle has a profound effect on the nose of the launch vehicles. Pressure distributions must be obtained for the analysis of the aerodynamic forces on the launch vehicle. The point where the aerodynamic dynamic pressure is at it's greatest, q_{max}, is a function of altitude (density) and freestream velocity. At q_{max} drag forces also peak. Pressure distributions at this condition may be analyzed for a first cut type of calculation for a sizing design analysis.

During the flight, the weight of the launch vehicle will decrease and will have an effect on the center of gravity placement and the stability of the launch vehicle. Center of gravity movement will be caused by stage separations, fuel burn, transfer of fuel between tanks, and jettisoned expendables, but this movement can be easily predicted. The movement of the center of gravity should be analyzed thoroughly because it may cause a shift in the load paths within the vehicle.

9.6.1.5 Launch Vehicle/Volume Constraints

Volume limitations are encountered with every payload, especially for inter-planetary vehicles. Solar arrays, magneto sensors, high heat generating components, antennas, and other oversized appendages are often stowed or tucked away within the allowable space of the launch vehicle. These appendages are later deployed with the use of springs, pyrotechnics, or angular momentum. The space vehicle must be designed to fit in the payload area of the launch vehicle. The capabilities of current and planned launchers is summarized in Table 9.6.5. Usually, the length of the payload bay is not specified and many vehicles can accommodate long payloads for special missions.

Country	Vehicle	Fairing envelopes dia(m)/length(m)	Deliverable Payload weight (kg) to Orbit			
			CEO/LEO	GTO	GEO	Interplanetary
China	Long March 1-D	1.9/2.8	860	200	100	
	Long March 3	2.3/3.1	5,000	1,500	730	
Europe	Ariane III	3.2 / 8.6	3,350	1,600		
	Ariane IV-40	3.6/12.4	4,900	1,900		
	ArianeV	3.6/13.7	20,000	7,400		
Japan	Mu-3S II	1.64	840			
	N-2	2.4	2,200	300		
	H-1A	2.2/6.5	3,200	1,100	550	
	H-2	3.7/10	10,500	4,000	2,500	
USA	Atlas-Centaur	2.9/7.7	5,580	2,250	450	1,300
	ALS (1 booster)	8.3	46,800			
	ALS (2 boosters)	10.6	88,200			
	Delta 3910/PAM	2.4	2,475			
	Delta 3920/PAM	2.4	2,600	1,200		
	Delta 3924	2.4	1,090			
	HLLV	12.37/29.7	126,000			
	Saturn 1 B	6.6	18,000			
	Saturn 5	6.7	83,000			42,000 (lunar)
	Scout D	1.2	270	54		
	Shuttle	4.6/18.3	24,400			
	Shuttle-C	4.6/24.7	64,000			20,400 (lunar)
	Titan 3	3.6/12.4	14,400	5,000	1,360	
	Titan 4	4.5/17	17,700	6,350	4,540	

Table 9.6.5 Launch Vehicle Parameters

9.6.2 Orbit and Mission Constraints

9.6.2.1 Thermal Gradients

Both internal and external thermal gradients exist in a spacecraft. The internal thermal gradients usually originate from subsystems such as propulsion, electronics, power, and communications. External thermal gradients are largely due to the sun. For interplanetary vehicles, other factors such as radiative planetary surface reflections (albedo) and aerothermodynamic heating may have to be considered.

The existence of these thermal gradients can cause thermal stresses and then geometrical distortions. If left unchecked, these stresses will cause deformations of some structural members which may disturb the alignments and stability of onboard sensors. Therefore, the heat generated within the spacecraft must be ejected from the craft. The spacecraft structure is commonly used to provide some thermal control. This is accomplished by placing heat generating components on highly thermal conductive structural members which have a conducting path to the surface of the vehicle where radiative dissipation will take effect.

9.6.2.2 Debris Protection

Since all types of spacecraft are in danger of being impacted by either micrometeorites or some type of space debris, the structure of spacecraft must be designed to withstand the majority of these impacts. Actually, the probability of a piece of debris impacting a space system is given by the kinetic theory of gases as

$$PC = 1 - e^{(-SPD.AC.T.V_{rel})}$$

where spd is the spatial density of debris, AC the cross-sectional area of satellite at risk, Vrel the relative velocity and T the mission duration.This formula is valid for LEO rather than GEO.

There would be a prohibitive weight penalty if the structure was designed for all possible impacts, thus the design is usually based on the assumption that the spacecraft will not be impacted by debris larger than a predetermined size. A size that has been used for space structures designed for use on or around the moon is a 1 gm micrometeorite with a 1.56 cm diameter.

There are a few different structural methods for debris protection, two of which are sacrificial bumpers and multilayered protection. The theory behind sacrificial bumpers is that a bumper can be designed to vaporize both a micrometeorite of known mass and the local bumper wall upon impact. Unfortunately, if the mass of the debris is smaller than the bumper is designed for, the bumper will not completely vaporize and the structure can actually be damaged by subsequent hypervelocity impact of bumper particles.The optimum back-up sheet (second wall) thickness, t_b, for aluminium, is given by the following equation (taken into consideration that the shield and the debris have to be vaporized during the impact):

$$t_b = \frac{Cmv}{S^2}$$

Where t_b is the back-up sheet thickness in cm, m is the projectile mass in g, v is the particle velocity in km/s, S is the spacing in cm, and C=41.5+/- 14.0.According to the current data and the predictions about the orbital debris population ,t_b for the Space Station is equal to 0.31 cm in 1995 and 1.8 cm in 2010.

The other method is to combine thermal and micrometeorite protection and use Multilayered Insulation (MLI). This method is implemented by wrapping the structure with several layers of thermal protection material separated by layers of low thermal conductivity impact protection material. This method is typically less massive, while still as effective as using aluminum bumpers.

9.6.2.3 Deployable Appendage Constraints

Thermal flutters caused by the spacecraft's spinning in an axis perpendicular to the radiation of the sun may cause an encroachment onto the vibrational design limits of the spacecraft. For a sensor or weapons platform pointing at objects far away, even tiny structural vibrations can cause large errors. For instance, some instruments on the Hubble Space Telescope are troubled by the snap and subsequent vibration as its solar panels expand in sunlight and contract in darkness. Booms and antennas are areas where thermal flutter can be a design factor, so there will be stiffness requirements and vibration limits applied to deployed appendages.Other sources of micro-perturbations are commonly the reaction wheels, the tape recorders, the instrument coolers using mechanical cycles (Stirling,..). It may also be necessary to include passive or active vibration suppression in the design of the spacecraft, depending on the spacecraft's function and the levels of vibration expected.

9.6.2.4 Aerobrake or Aerothermodynamic Heating

Some spacecraft will require an aerobrake to be incorporated in their design. A detailed discussion of aerobrake design is located in another portion of this manual.

9.6.3 Spacecraft Design Criteria

9.6.3.1 Mass Distribution

The distribution of subsystem components within the spacecraft is critical because it determines the center of gravity and the moments of inertia about the craft's principle axes. Mass distribution of the components should provide minimal moments of inertia in order to provide optimum attitude control characteristics, and interfacing with the attitude control team is important. The actual center of gravity placement will be dependent on the location of the thrust line.

Scientific sensors often need large field of views, and thus may require a boom. Solar arrays need a surface attachment with a large field of view or are deployed on a boom. Antennas, particularly high gain antennas, also need unobstructed fields of view .

9.6.3.2 Mass Estimation

Although a significant design process is required to determine the attributes of a spacecraft structure, for a preliminary design the critical output is the structure's mass, and that can be estimated from other preliminary results. Inputs into the mass estimation include the various mission design parameters and outputs from the other subsystem analyses.

The outputs from the other subsystems to the structural design will include thermal loads, electrical and data transmissions and most significantly, forces and torques, which are driven by the masses of these subsystems. Because of this, the principle factor in the estimation of the mass of the structure is the sum of the masses of the other subsystems.

Analysis of the structural masses of various spacecraft is complicated by the wide variety of definitions of the structure used in the literature. While the primary structure is always included, the listed mass of the structure may or may not include the secondary structure, brackets and attachments, booms, various mechanisms and hardware, ballast, fasteners, and other miscellaneous components. Even cabling, solar arrays, and the entire thermal system are part of the structure by some definitions. A list of the masses of a few satellites is given in Table 9.6.6.

Spacecraft	Structural Mass (kg)	Total Dry Mass (kg)	Comments
ATS-6	212.1	1,297.30	
Mariner 5	30.999	244.563	
Pioneer 6	17.46	137.94	includes booms
Pioneer 9	18.10	147.20	includes booms
Intelsat V	183.1	749.80	includes thermal
Magellan	246.3	1,046.0	
Galileo	255.2	1,051.8	
Mars Observer	231.21	1,011.09	includes mech.

Table 9.6.6 Sample satellite structural masses

Based on the data presented in Table 9.6.6, an empirical equation relating structural mass to non-structural mass can be derived. That equation is:

$$M_S = 0.032M_N^{1.084}$$

where M_N is the total mass of the other subsystems, and M_S is the mass of the structural system, including booms and mechanical hardware not specifically part of another subsystem.

For an idea of the mass of the primary bus structure, Table 9.6.7 lists this value for some spacecraft.

Spacecraft	Primary Structure Mass (kg)
ATS-6	192.4
Pioneer 6	7.01
Pioneer 9	7.03
Magellan	123.8
Mars Observer	135.81

Table 9.6.7 Sample masses of the primary bus structures of some satellites

Table 9.6.8 presents subsystem mass data for several spacecraft. The relationship for structural mass that results from this data is:

$$M_S = 0.315 M_N^{1.084}$$

where here M_S represents the mass of the structural, thermal and cabling systems.An other commonly used rough estimation of the structural mass is that it should be, for an optimum design, between 7 and 10 % of the spacecraft total mass.

Subsystem Masses (kg)						
Satellite	Structure	Telemetry & data	Power	Experiments	Guidance & Control	Total
Ranger 6-9	120	40	114	27	47	356
Ranger 1-5	88	30	73.5	24	56	277
IMP A-C	19.5	6	18	1	16.5	61
IMP D-E	17	7.5	19	1	9.5	93
Mariner-R	60.5	38	46.5	24	21.5	198
Mariner-64	62	51	70.5	37	20.5	252
OGO A-E	182	71.5	81.5	49	86	470
RELAY	20.5	20.5	35	1.5	5.5	83
SYNCOM	9	14.5	6	4	0	38
SURVEYOR 1-7	133.5	24.5	45	38	12	356
OSO	70	19.5	23.5	28	113	254
ESRO-II	24.5	10	14	5	21.5	75
ESRO-I A	34	12.5	16	2.5	20	85
HEOS A-1	40	8.5	17	8.5	27	101
ESRO-I B	34	12.5	16	2.5	20	85
HEOS A-2	41	9	18.5	11	25.5	105
TD	184	33	35.5	86.5	116	455
ESRO IV	22	11.5	20	8.5	32	94

Table 9.6.8 Mass Break-down of satellites

Estimation of the mass of the structure can be accomplished by the use of an empirical function of the total mass of the other subsystems. The simplistic nature of this function curtails its usefulness in many situations. Some particular design requirements that will not be reasonably estimated using this relationship include:

<u>Manned systems</u> - The restrictions provided by ECLSS and the high contained volumes required produce a very different set of constraints from unmanned

systems. An approximation as a fuel tank may produce a meaningful estimate for some uses.

Re-entry vehicles - The mass of the heat shielding is sometimes included in the structure. More information on this can be found in the aero-braking section. The vast range of accelerations possible during aero-braking (the Galileo probe will peak at 231 g's) requires a more complex analysis. As an example for this type of structural mass, the Galileo probe has a structural mass of 90.7 Kg., and a total mass of 120.6 Kg.

Spacecraft with significant aerodynamic behavior - While this is a subset of re-entry vehicles, this is an even more complex system and requires an even more involved analysis. There is much effort underway to model these systems, especially to assist in the design of shuttle derivatives. For more information see MacConochie et al.

Large structures - These type of structures are dissimilar to other spacecraft types, and require a very different design analysis.

Table 9.6.9 presents the total weight of several spacecraft.

Spacecraft	Total Weight (lbs., dry)
Apollo CSM (wet)	51,000
Apollo LM (wet)	33,000
ATS-3	805
ATS-6	3,090
Comstar-l	1,800
Echo I	166
FLTSATCOM	2,216
Gemini capsule	7,000- 8,374
HS 376	1,300
Intelsat 2	192
Intelsat 4A	1,750
Landsat	1965 - 2100
Lunar Orbiter	853
Mariner 2 (Venus)	447
Mariner 3 (Mars)	575
Mariner 9 (Mars)	2,200
Mariner 10 (Venus & Mercury)	1,108
Mercury capsule	2,000-3,000
Pioneer 10 & 11	570
Ranger	840
RCA Satcom	1,021
Shuttle Orbiter	150,000
Skylab	167,849
Surveyor	2,194 (596 after landing)
Viking orbiter	5,125
Viking lander	2,353
Voyager 1 & 2	1,797

Table 9.6.9 Spacecraft Weight

9.6.3.2 Electrical Grounding

Many spacecraft components need grounding to prevent internal circuitry from burning out due to environmental electrical fluctuations or the regular operation of the spacecraft itself. Electrical environments can be caused by solar flares, traversing through the earth's Van Allen belts, and electrical switching within the spacecraft. Electrical grounding is a similar situation as the thermal conduction case. Both require a path of conducting material which will lead to the main structure of the spacecraft. Fortunately, a good electrical conductor is usually a good thermal conductor.

9.7 Design Verification

The structural design of any spacecraft must be verified, either by a complete set of tests or by modelling and analysis supported by limited testing. The actual method of verification will determined by such factors as physical size and the number of qualification models built. In other words, the structure may be too large to fit onto a test facility or it may be desirable to limit testing if only the actual flight model is built. An outline of the common method of design verification is in Figure 9.7.1.

With the use of finite element analysis (FEA) software packages such as NASTRAN, it is possible to model structures mathematically in great detail, and to examine their behavior under all possible static and dynamic load conditions. For instance, in a dynamic simulation, the structure's natural frequencies can be assessed and relative phase information of deflection shapes at different locations within the structure can be indicated. Once a complete set of finite element analyses has been performed on the whole structure and other subsystems, a reduced model, which demonstrates similar characteristics to the larger version, is used for incorporation into the overall spacecraft simulation. This model is then incorporated into a coupled analysis model of the launcher-spacecraft combination so that a full examination of the complete launch configuration can be made.However, the results of finite elements analysis might present some important discrepancies with the actual data. As small joints and mechanisms are difficult to modelize and subject to high concentration of stresses, the finite element analysis can sometimes lead to errors up to 40%.In dynamic models, where such details are less important, the average error rate is around 10%.The fact is that these error rates could be corrected by a more and more detailed model.Nevertheless, such an improvement requires time and money, which are actually the factors that define the limit the use of FEM in spacecraft design. Usually. a good solution to this problem is to use test data to correct the mathematical model.

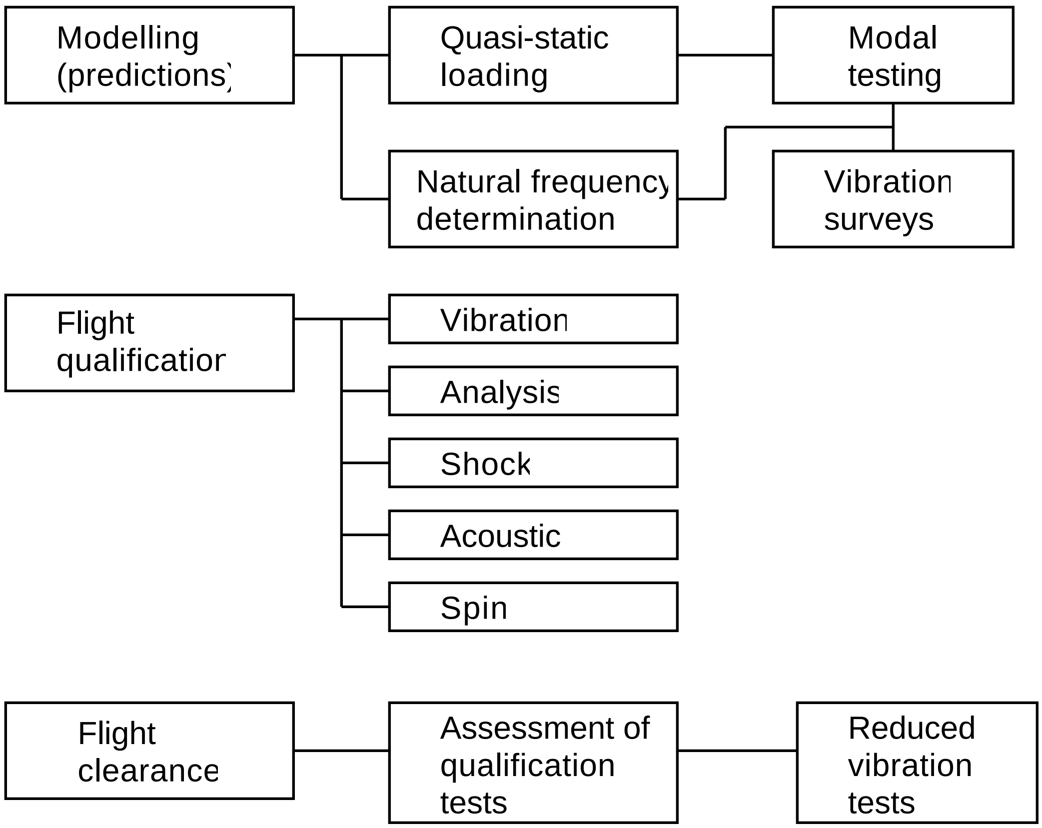


Figure 9.7.1 Structure Verification [Fortescue 1991]

The structural testing required to qualify an assembly for launch is often accomplished by subjecting a prototype to static and vibration loads in excess of those anticipated for flight and the actual flight unit is subjected to near flight levels (approximately 110%). The vibration testing is carried out in each of the three axes independently and is used to determine the response of the structure to vibration, which includes determining its natural frequencies. For large structures that can not be excited by a single vibrator, a test is performed using small vibrators positioned within the structure. The placement of these vibrators is done with information obtained in the FE model.

In addition to vibration testing, the structure is also subjected to shock, quasi-static load, and spin testing. Shock tests, used to simulate shocks due to releases or deployments, are performed by feeding a simulated shock into the vibrator to excite the spacecraft. Since a full static load test can not be realistically performed, quasi-static load testing is performed by attaching loads to appropriate nodes of the structure and comparing the resultant deflections to the FEA predictions. For any spacecraft that will undergo spinning in operation, spin testing is performed to test its structural integrity and to allow the dynamic balance of the spacecraft to be measured.

9.8 Cost Modeling

A good analysis of spacecraft design cost is provided in Chapter 20 of "Space mission analysis and design" W.J.Larson.First, the cost of the structure / thermal subsystem can be calculated in terms of Fiscal Dollars for a given year.Second, methods are given to translate this sum in actual cost, which takes into account not only the value of the subsystem but also the time spreading of the funding and the inflation during the project.

9.9 Future Subsystem Report Recommendations

It will be important to better define the output process through specific methodology, such as Finite Element stress analysis. The mass estimation section can be expanded by obtaining more data. The materials section should be updated as new materials are developed. Other areas to look into are relative costs of materials and configurations, lunar or non-terrestrial materials for space construction, and structures for large space assemblies such as space stations.

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