Design of Spacecraft structure

8.1 Introduction

The various design elements of a spacecraft structure are illustrated in Fig. 8.1, [Agrawal 1986, chapter 4].

The design of a spacecraft structure can be subdivided into five phases.

1. Determination of spacecraft configuration
2. Initial design of the spacecraft structure
3. Detailed analyses
4. Production of the spacecraft structure
5. Testing

All phases will be discussed in this chapter.

8.2 Determination of Spacecraft Configuration

As part of the determination of the spacecraft configuration the following may be involved:

- Boundary Conditions Launch Vehicle
  - Launch weight
  - Available volume
  - Adapter
  - Payload separation system
  - Launch costs
- Functional requirements
- Mission time (duration)

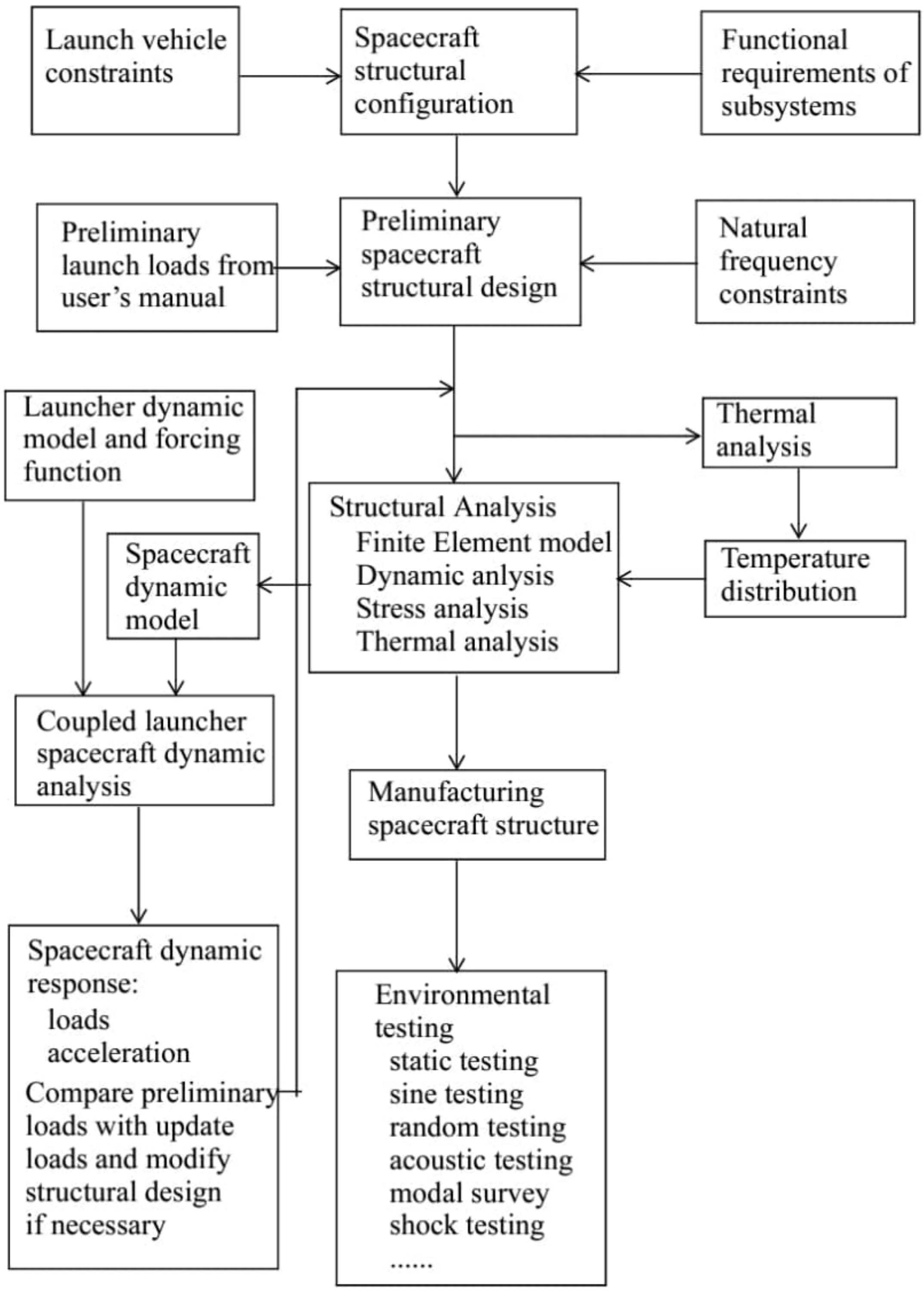

Fig. 8.1 Design flows spacecraft structure design and verification [Agrawal 1986]

8.2.1 Boundary Conditions Launch Vehicle

All the requirements and constraints are extensively covered in the user manuals of the associated launch vehicles. These include requirements and constraints concerning:

- The mass to be launched
- The available volume within the nose cone
- Launch vehicle adapter
- Vibrations
- Acoustic loads
- Safety factors

The mechanic and acoustic loads and the safety factors are discussed in previous chapters

8.2.2 Launch mass

The mass (spacecraft + adapter) that can be launched depends on the launch mission. The following general/common launch missions are mentioned below, i.e. ARIANE 5.

- The launching of spacecraft in a Geostationary Transfer Orbit (GTO).
- The launching of spacecraft in a Sun Synchronous Orbit (SSO).
- The launching of spacecraft in a Low Earth Orbit (LEO).
- The launching of spacecraft in an elliptic orbit around the earth.
- The escape of a spacecraft from the gravitation of the earth (escape mission).

The launch possibilities of the standard version of the ARIANE 5 launch vehicle with regards to a certain launch mission are given in the following Table 8.1.

Table 8.1 ARIANE 5 Launch capabilities

Geostationary Transfer Orbit (GTO) (kg)	Low Earth Orbit (LEO) (kg)	Sun-Synchronous Orbit (SSO) (kg)
6800	18000	10000

8.2.3 Available Launch Volume

The volume of the spacecraft that can fit inside the adapter of the launch vehicle depends on the diameter of the adapter and on the size of the fairing. If one spacecraft is being launched then the available volume is obviously the largest. When several spacecraft are launched at once, the volume must be shared. The available volume is extensively covered in the user manual of the launch vehicle.

The space inside the fairing of the Taurus launch vehicle is illustrated in Fig. 8.2.

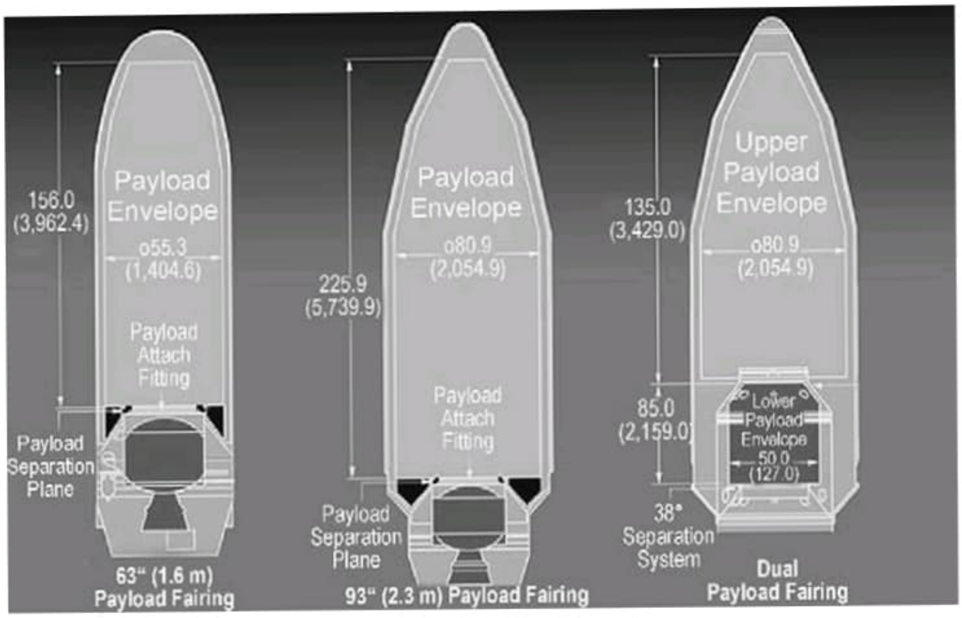

Fig. 8.2 Taurus Fairing Acommondations (courtesy Taurus Launch Systems)

8.2.4 Launch Vehicle Adapter (LVA)

Usually there are various launch adapters and or coupling structures available on which the spacecraft can to be placed. The dimensions of the adapters and coupling structures are outlined extensively in the user's manual of the launch vehicle.

8.2.5 Payload Separation System

The payload separation system consists of either:

- Pyrotechnique cutting device(s).
- Clampband. The Clamp Band (CB) consists of two halves of steel bands fixed by two connecting bolts. The tensile stress in the band exerts pressure on the clamp that connects the adapter with the spacecraft. The Pyro Bolt Cutters cut through the connecting bolts so that the half bands come loose. The separation springs are loaded in such a way that the spacecraft can separate safely from the adapter. The Clamp Band grasping system grasps the two halves of the Clamp Band and thus prevents the spacecraft from clinging behind it.

Both systems will introduce high shock loads on the spacecraft.

8.2.6 Functional requirements spacecraft

The functional requirements can vary immensely and are strongly dependent on the mission. The various missions can be:

- Sun-observation
- Planetary
  - Fly-by
  - Orbiter
  - Lander
- Geostationary spacecraft
  - Communication
  - Television
  - Meteorological
  - Earth observation
  - Navigation
- LEO spacecraft
  - Telecommunications
  - Meteorological
  - Earth observation
  - Navigation
  - Micro-gravity
- Astronomy
  - Space observatory
- Fields and particles
- ……………

8.3 First Design Spacecraft Structure

For the initial preliminary design of the spacecraft structure, the following aspects must be considered:

- Design Launch Loads
- Factors of Safety
- Stiffness requirements
- Materials
- Basic Design

  The preliminary sizing of the structural components is illustrated in Fig. 8.3.

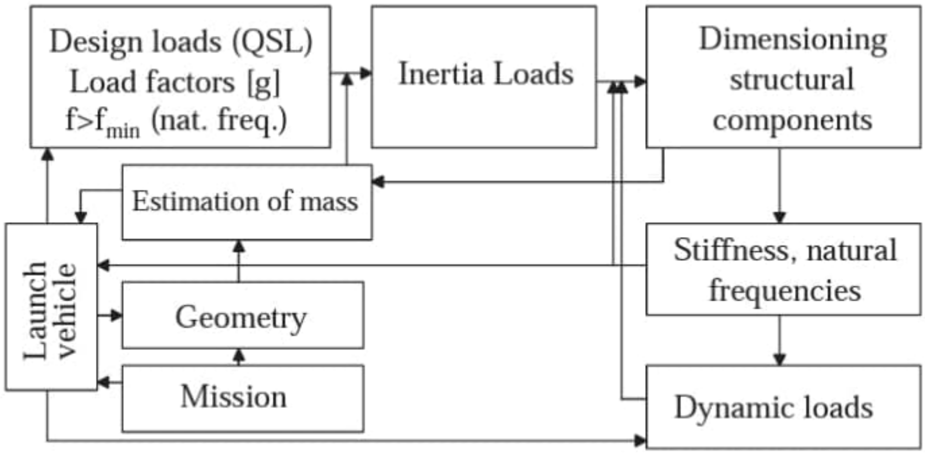

Fig. 8.3 Preliminary sizing structural components

The mission requirements in combination with the functional requirements will lead to a preliminary geometry and associated mass distribution. The selected launch vehicle will specify launch loads, stiffness, geometry constraints, etc. The quasi-static load factors will be applied assuming a minimum natural frequency compliance. The quasi-static loads factors applied to the mass distribution will result in internal load distribution. This load distribution is a starting point for sizing the structural members of the spacecraft primary and secondary structures. After that, the stiffness of the primary structure must be checked against launch vehicle stiffness requirements. If the minimum of the longitudinal and lateral natural frequencies comply with the requirements, dynamic loads may to be applied.

8.3.1 Design Loads

The various mechanical loads are not all equally important and depend on the type of the mechanical structure: i.e. does it concern a primary structure, the spacecraft structure or other secondary structures (such as solar panels, antennas, instruments and electronic boxes). Preparations on the ground, the launch, and the operations in the orbit around the Earth set various types of requirements, such as [Sach 1988]:

- natural frequencies
- steady-state (semi-static) acceleration
- sine excitation
- random excitation
- acoustic noise
- transient loads
- shock loads
- temperatures

Natural frequencies
The natural frequency is a governing design requirement for all parts of the spacecraft. This requirement is imposed in order to limit the dynamic coupling of the spacecraft with the launch vehicle.

Semi-static and low frequency sinusoidal loads
The design of the primary structure is determined to a large extent by the semi-static and low frequency sinusoidal loads (up to approximately 50Hz).

Sinusoidal and random loads
To a large extent, the sinusoidal and random loads determine the design of secondary structures (solar panels, antennas, electronic boxes).

Acoustic loads
Light structural parts with relatively large surface areas (such as solar panels and spacecraft antennas) are more sensitive to acoustic loads than sinusoidal and random base excitation.

Shock loads
Deployable structures experience high shock loads; for example during latch-up of hinges in the required final position of these mechanisms. This is especially the case when the deployment velocities are too high.

Temperatures
Temperature variations usually cause high thermal stresses in the structures. In general, the various coefficients of expansion are accounted for in the choice of the structural materials. Thermal deformations are taken into account when working with structures that must be aligned with each other.

Random Loads
The design of instruments and electronic boxes are determined by the random base excitation.

8.3.2 Stiffness requirements (natural frequencies)

Additionally, the natural frequencies of the spacecraft must be such that the fundamental natural (undamped) frequencies in all directions are larger than the lowest frequencies generated by the launch vehicle that excite the spacecraft. Due to the difference in natural frequencies, the spacecraft is dynamically uncoupled from the launch vehicle and will display rigid behaviour in the lower frequency regions.

In the Table 8.2 the required lowest frequencies in the lateral and the launch direction are given for various launch vehicles. In general, these frequencies are valid when the spacecraft is considered to be clamped at the interface between the spacecraft and the launch vehicle.

Table 8.2 Examples stiffness requirements (Continued)

Launch Vehicle Launch System	**Required Lowest Natural frequencies (Hz)**	
Direction	Launch direction	Lateral
STS	13	13
DELTA 6925/7925	35	15
ARIANE 5		9–10[a]
<=4500 kg	31	
> 4500 kg	27	

a. Depends on the launcher spacecraft interface

8.3.3 Quasi-static loads

The structure of the spacecraft is designed to support the maximum quasi-static loads (QSL), including a factor of safety. The quasi-static loads are a combination of the steady-state static loads and the low frequency sinusoidal loads. Quasi-static loads can be used to dimension the spacecraft, provided the minimum frequency requirements with respect to mode shapes in the launch direction and the lateral direction are fulfilled.

In the ARIANE 5 user manual the sizing loads for spacecraft with a weight ≤ 5000 kg are specified in the following way:

Table 8.3 Quasi-static load factors

Flght event	**Load factors, acceleration (g)**			
	Launch direction		**Lateral**	
	Static	**Dynamic**	**Static**	**Dynamic**
Lift-off	–1.7	±1.5	0.0	±1.5
Maximum dynamic pressure	–2.7	±0.5	0.0	±2.0
P230 Burn-out	–4.25	±0.25	±0.25	±0.25
H155 Burn-out	–3.6	±1.0	±0.1	0.0
H155 Thrust tail-off	–0.7	±1.4	0.0	0.0

- The minus sign refers to a pressure force in the launch direction.
- The quasi-static loads are applied for the centre of gravity of the spacecraft.
- Gravity has been taken into account.
- The spacecraft must fulfill the stiffness requirements.
- The centre of gravity of the spacecraft must be located in a certain area to prevent overloading of the spacecraft adapter. This depends of course on the adapter used.

8.3.4 *Mass Acceleration Curve (MAC)*[1]

In Fig. 8.4, the design loads for the structural analyses, which depend on the effective masses, for the components and instruments are given. Components with a low effective mass experience higher acceleration. This observation is valid both for transient as well as random (stochastic) loads. The "mass acceleration curve" (MAC) is primarily based on experiences (data) from previous projects. In most cases, a MAC can be derived for a launch vehicle that can subsequently be used for most components and instruments. The MAC is an upper bound acceleration level for all components of a given mass, regardless of location, orientation, or frequency. In general, it is assumed that the lowest natural frequencies are $f_n \geq 100$ Hz.

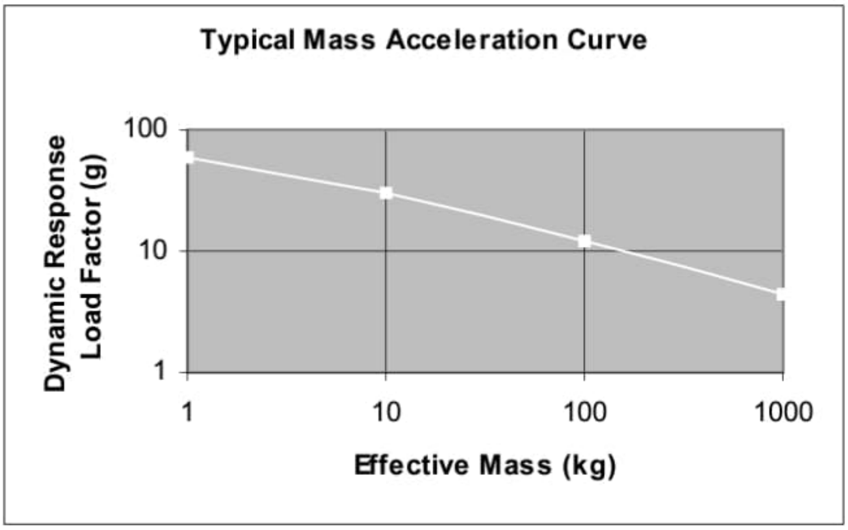

Fig. 8.4 Typical mass acceleration curve [NASA PD-ED-1211]

Example

We have a component of about 10 kg and the lowest natural frequencies are above 100Hz. The "static" load factors to design that component can be taken from the MAC (Fig. 8.4) and are about 40g.

End of example

1. The term "Mass Acceleration Curve" is frequently used by the NASA in many "Spacecraft loads" documents. MAC is also the abbreviation of the term "Modal Assurance Criteria".

8.3.5 Random Loads

The (test) qualification random loads depend on the type of launch vehicle and spacecraft.

In the "General Environmental Verification Specification for STS & ELV, Payloads, Subsystems and Components", [Baumann 1996], a distinction is made between:

- Spacecraft
- Instrument (subsystem) (≤ 68kg, 150lbs)
- Component (≤ 22kg, 50lbs)

In the following Table 8.4, Table 8.5 and Table 8.6 "Random vibration levels" are specified. It must be noticed that the higher the mass of the component the lower the random vibration levels.

Table 8.4 Spacecraft "Random Vibration Levels"

Frequency spectrum (Hz)	PSD acceleration (g^2/Hz)
20–800	0.008
800–1000	7.6 dB/octave
1000–1300	0.014
1300–2000	–13.6 dB/octave
2000	0.002
Grms	4.1 g

8.3.6 Factors of Safety

The factors of safety are used to account for uncertainties that cannot be fully analysed.

The qualification loads are often used as design loads, and subsequently the factors of safety for "yield" and "ultimate" are applied.

Table 8.5 Instrument "Random Vibration Levels"

Frequency spectrum (Hz)	PSD acceleration (g^2/Hz)
20	0.01
20–50	5.5 dB/octave
50–800	0.053
800–2000	–5.5 dB/octave
2000	0.01
Grms	8.26 g

Table 8.6 Component "Random Vibration Levels"

Frequency spectrum (Hz)	PSD acceleration (g^2/Hz)
20	0.026
20–50	6 dB/octave
50–800	0.16
800–2000	–6 dB/octave
2000	0.026
Grms	14.1 g

The factors of safety that were used in the ENVISAT spacecraft project are illustrated in Fig. 8.5.

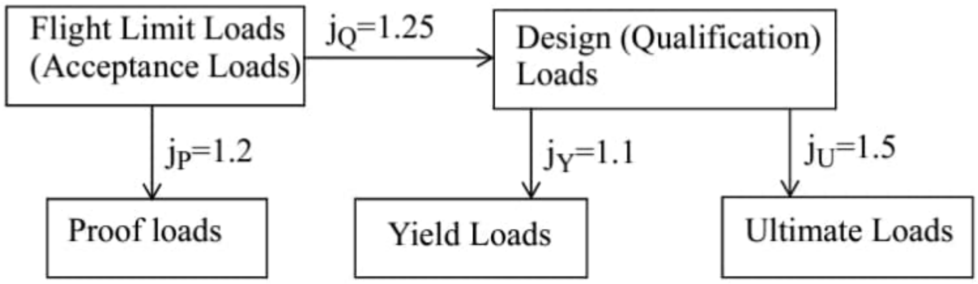

Fig. 8.5 ENVISAT Safety factor philosophy

8.4 Basic Design Supporting Structure

The spacecraft structure is designed to mechanically support the service systems, the payload or the instruments and is essentially the backbone of the spacecraft. In addition, the structure ensures that the instruments are properly aligned with each other and that the surfaces of certain structural elements are finished in such a way that they meet the requirements set by the thermal subsystem.

8.4.1 Design criteria

The design criteria for the structural elements are:

- Mass (minimum)
- Reliability
- Design costs (e.g. engineering hours)
- Production costs (including moulds and templates)
- Ease of inspection (e.g. NDI)
- Ease of reproduction

- Possibility to repair
- Modification possibilities of hardware (H/W) in a late design phase

These criteria have been presented in an arbitrary order. The mass is important, however, sometimes costs are even more important. Each product has its own order for these criteria.

8.4.2 Standard structural elements of spacecraft structures

A spacecraft is generally constructed with the following standard structural elements:

- Bending beams
- Tension and compression members (Struts)
- Ring frames
- Plates/panels
  - Rectangular
  - Circular
  - Annular
- Cylindrical and conical thin-walled shells
  - Moncoque
  - With rings and stiffeners
  - Corrugated
  - Composites
- Sandwich and composite structures
- Pressure vessels[1]
- Fuel tanks
- ...

The design of a structural element is specified by three aspects [Ashly 2003]:

1. The functional requirements
2. The geometry
3. The properties of the materials used

The performance p of a structural element is described by the following equation

---

1. A pressure vessel stores 19130 J of energy or more. The energy is based on the adiabatic expansion of an ideal gas.

$$p = f\left[\begin{pmatrix}\text{Functional}\\ \text{requiremnts, } F\end{pmatrix}, \begin{pmatrix}\text{Geometrical}\\ \text{parameters, G}\end{pmatrix}, \begin{pmatrix}\text{Material}\\ \text{properties, } M\end{pmatrix}\right] \tag{8.1}$$

Equation (8.1) is separable, hence

$$p = f_1(F)f_2(G)f_3(M) \tag{8.2}$$

where $f_1(F)f_2(G)$ is the structural index and $f_1(M)$ is the efficiency coefficient or material index.

The provisional sizing of a space structure can be done quickly and efficiently provided that straightforward methods (Manual, EXCEL®, Mathcad®, Maple®, MATLAB®, etc.) are used to calculate the aforementioned elements.

Important aspects for the determination of the dimensions of a space structure are:

- Design loads
  - Handling loads
  - Launch loads
  - In-orbit loads
- Test loads
- Internal pressures (i.e. fuel tanks)
- Minimum requirements with respect to natural frequencies
- Thermo-elastic deformations
- …

The most important failure modes are:

- Exceeding the yield stress
- Exceeding the ultimate strength
- Stability (against buckling), locally or generally
- Fracture mechanics (pressure vessels, manned space flight)
- Fatigue
- …

Margins of safety (MS, or MOS) and factors of safety have a different meaning. For a given factor of safety the "probability of failure" or in other words, the reliability of the structure, can be determined. With the aid of the concept of MS we determine the functionality of the structure. The MS is defined as follows:

$$\mathrm{MS} = \frac{\sigma_{\text{allowable}}}{j\sigma_{\text{actual}}} - 1\,, \tag{8.3}$$

where MS is the margin of safety and $\sigma_{\text{allowable}}$ is the allowable stress. The allowable stress is the maximum permissible stress before failure. The σ_{actual} is the resulting stress from a certain load and j is the factor of safety.

At failure, the applied stress is larger than the allowable stress. The MS value is then negative. From this it follows that the MS value must be greater than zero.

In Table 8.7 the significance of the MS value is shown.

Table 8.7 Significance of MS values

Margin of safety	Significance
MS<0	Failure
0<MS<0.5	Optimal design
0.5<MS<1.5	Good design
MS>1.5	The design may be easily improved

For a combination of load conditions, the value for the MS can be determined as shown in Table 8.8.

Table 8.8 Combined loadcases

Margin of safety	Combined load cases
$MS = \frac{1}{j}\left(\frac{\sigma_{compr}}{\sigma^*_{compr}} + \frac{\sigma_{bend}}{\sigma^*_{bend}}\right)^{-1} - 1$	Combination of compression and bending loads
$MS = \frac{1}{j}\left(\frac{\sigma_{compr}}{\sigma^*_{compr}} + \frac{\sigma_{tors}}{\sigma^*_{tors}}\right)^{-1} - 1$	Combination of compression and torsion loads
$MS = \frac{1}{j}\left(\frac{\sigma_{compr}}{\sigma^*_{compr}} + \frac{\sigma_{bend}}{\sigma^*_{bend}} + \left\{\frac{\sigma_{tors}}{\sigma^*_{tors}}\right\}^2\right)^{-1} - 1$	Combination of compression, bending and torsion loads

where $\sigma_{...}$ is the resulting stress, $\sigma^*_{...}$ is the allowable stress and j is the factor of safety.

8.4.3 Selection of materials

A very important step in the design process is the selection of materials for a spacecraft structure. The choice has significant consequences for the mass, the production costs, etc. The operational conditions of the spacecraft, the ability to retain its shape and the reliability of the structure are some of the parameters used for the selection of materials. The most important material properties are:

- Strength and stiffness
- Specific weight

- Ultimate strength
- Fatigue strength
- Technical constraints (elasticity, weldability, stress concentrations, etc.)
- Effect of the environment on the material properties
- Thermal conductivity
- Electrical conductivity or resistance
- Availability
- Costs

With regards to structural elements that are subjected to a tensile stress, the following simple expression can be derived:

$$M = AL\rho = j\frac{N}{\sigma_u}L\rho\,, \tag{8.4}$$

in which M is the mass of the structural element, A is the cross-section of the structural element, L is the length of the structural element, N is the normal (tension) force, σ_u is the ultimate stress of the applied material, ρ is the density of the applied material and j is the factor of safety.

It appears from the previous expression that the mass of the structural element decreases while the specific strength $\frac{\sigma_u}{\rho}$ increases.

For thin-walled structural elements (monocoque, sandwich, with stiffeners) subjected to an axial compression load, the mass M for various buckling conditions can be expressed as follows:

$$M = 2\pi R\sqrt{\frac{jN}{2\pi\psi E}}, \tag{8.5}$$

in which R is the radius of the monocoque shell and ψ is a constant that depends on the boundary conditions.

For thin-walled structural elements the mass of the structural element decreases while the specific stiffness $\frac{\sqrt{E}}{\rho}$ increases.

Analogous expressions can be derived for different loads and structural elements.

Ashly defines in [Ashly 2003] the structure efficiency (SE) or material index which stands for:

$$\text{Structure Efficiency} = \frac{\text{Load carried by the structure}}{\text{mass of the structure}}$$

$$\text{SE} = f\left(\frac{E^i}{\rho}\right),\ i = 1, \frac{1}{2}, \frac{1}{3} \tag{8.6}$$

In Table 8.9 an example of the structure efficiency are given.

Table 8.9 Structural efficiency or material index (Continued)

Structural element	Load	SE	Remarks
Beam, Plates and shells	Tension and pure compression	$f\left(\frac{E}{\rho}\right)$	$L < L_{crit}$
Sandwich panels and shells (symmetric)	Tension and pure compression	$f\left(\frac{E}{\rho}\right)$	$L < L_{crit}$
	Bending	$f\left(\frac{E}{2\rho_f}\right)$	Strength
		$f\left(\frac{E}{3\rho_f}\right)$	Stiffness
Beams	Buckling and bending	$f\left(\frac{E^{\frac{1}{2}}}{3\rho_f}\right)$	
Plate and shells	Buckling and bending	$f\left(\frac{E^{\frac{1}{2}}}{3\rho_f}\right)$	

When assessing the material choice for the various structural elements, the effect of the temperature on the material properties must be taken into account. For this reason, lightweight metal alloys and composite materials are often used for the construction of a spacecraft.

8.5 Detailed Analyses

The structural analyses are, in general, done with the finite element method. There are many commercially available finite programmes that are used in conjunction with compatible pre- and post processors. Examples of commercially available finite element packages are:

- MSC.Nastran
- MSC.Marc
- ABAQUS
- .....

and examples of pre- and post processors are:

- MSC.Patran
- FEMAP

- ABAQUS/CAE
- ...

The objective of the analysis defines the detail of the finite element model; in terms of the number of nodes and finite elements, and therefore the number of degrees of freedom. The detail of the finite element model is also defined by the availability of geometic information, applied materials, mass distribution and loads.

8.5.1 Finite Element Model

If the geometry and applied materials and associated material properties are known with the aid of pre- and post processors, a finite element or mathematical model can be generated. This finite element model, in general, consists of the following:

- Nodes or nodal points, scalar points (one DOF).
- Finite elements, 0-D, 1-D, 2-D and 3-D
- Material properties (Young's modulus, Poisson's ratio, shear modulus, density, coefficient of thermal expansion (CTE), structural damping,...)
- Mass distribution (material density, non structural mass, discrete masses)
- Boundary conditions (clamped, simply supported, pinned, etc.)
- Constraint equations to relate DOFs with each other
- Applied loads
- Damping
- ...

The quality of the finite element model will be checked by performing dedicated finite element model checks later in the programme by tests (static, dynamic and modal survey).

8.5.2 Finite Element Model Verification

Finite element models shall be checked for:

- Rigid body strain energy to determine for hidden constraints in the finite element model. Theoretically the rigid body strain energy must be zero.
- Free-free modal analysis to determine for unwanted mechanisms in the finite model. A correct finite element will show six zero rigid body natural frequencies, three translations and three rotations.
- Stress free thermal-expansion to determine for bad elements in the finite element model. Bad aspect ratio elements, and warped plate elements, for example, will show non-zero stresses.

- In the case of pressure loads the normal vectors of the faces on which the pressures are applied must point in the same direction, otherwise forces due to pressures may be cancelled out.
- ...

8.5.3 Finite Element Analyses

Generally, the finite element models are used to perform the following types of analysis:

- Strength/stiffness
- Thermo-elastic
- Dynamic (e.g. modal response)
- Spacecraft / launch vehicle coupled load analysis
- Vibroacoustic
- ...

Strength analysis
The strength properties of the spacecraft structure (structural elements) shall be verified by finite element analysis and later in the programme by dedicated tests. The stress/load distribution in structural elements must be verified and compared with allowable stresses/loads (associated with identified failure modes) showing margins of safety greater than zero. A dedicated buckling analysis will give allowable buckling stresses/loads. Typical applied loads are the quasi-static inertia loads and combinations of them. If only dynamic loads factors are applicable, the stresses/load due dynamic load application must verified against allowable stresses and loads (in general static). For random loads 3-sigma values of stresses must be compared with the allowable stresses (yield, ultimate).

Thermo-elastic Analysis
The thermal deformation and stress due to temperature gradients in the structure must be calculated to check alignment requirements. In spacecraft, thermal stresses in the structure are, in general, not important. One of the major tasks is to depict the temperature distribution on nodes in a finite element applied to structural analyses. The temperature distribution is mostly obtained from thermal analysers based on the lumped parameter method.

Dynamic Analysis
Dynamic analysis is done to check if the specifications for natural frequencies and dynamic responses are met. To check the requirements about minimum natural frequencies a modal analysis will be done. The outcomes of such analysis are primarily: the natural frequencies and the associated mode shapes, generalised masses and stiffnesses and the modal effective mass. The damping is mostly ignored during the eigenvalue extraction process because damping in spacecraft structures is low (2–10% damping ratio). The deviation between real and complex modes is small.

A spacecraft industry standard modal damping ratio of $\zeta = 0.015$ or $g = 0.03$ structural damping is used [Foist 2004]. The associated amplification factor is $Q \approx \frac{1}{2\zeta} = \frac{1}{g} = 33.33$.

Later in the programme, the modal properties are checked with modal analysis or with a low input frequency sweep with a low sweep rate on a shaker table.

After the calculation of the modal properties of the spacecraft, the response characteristics due to mechanical deterministic and random loads are calculated. This is mostly done in the frequency domain. The application of the finite element model to specified frequency ranges must be checked.

The statistical energy analysis (SEA) method can be applied in frequency ranges out of the scope of the finite element application [Wijker 2004].

Coupled Load Analysis

Early in the programme a spacecraft/launch vehicle coupled dynamic load analysis is done to analyse the dynamic loads during the launch of the spacecraft. This analysis is done to verify the preliminary design loads from the launch vehicle user's manual (which are conservative).

A complete or reduced finite element model, in combination with load transformation matrices, must be delivered to launcher authority. The outcome of the coupled load analysis may be used during vibration tests.

Vibroacoustic Analysis

Lightweight, large area structures, i.e. solar arrays and antenna reflectors, are very sensitive to acoustic loads (sound pressures) and are mounted outside the spacecraft. A combined finite element method and boundary element method analysis is needed to simulate the fluid structure interaction (FSI). The structural behaviour will be covered by the finite element method (modal base: natural frequencies, vibration modes, stress modes,...) and the influence of the fluid (added mass, radiation damping) and the acoustic loads are dealt with by the boundary element method.

8.6 Manufacturing of the spacecraft structure

The structural parts of the spacecraft structures may be produced by the responsible company or sub-contracted to other companies. The subcontracting of structural parts may either be based on the 'build to print' principle or risk sharing principle (which means that the design and production of a structural part is done under the responsibility of the subcontractor).

The assembly of the structure is mostly done at the premises of the company responsible for that spacecraft structure. The complete spacecraft is generally assembled at the premises of the prime contractor.

Mechanical ground support equipment (MGSE) must be produced to assemble mechanical parts to a complete structure and to transport the spacecraft or parts of the spacecraft.

8.7 Testing

Tests are applied to verify requirements posed on a spacecraft design. The general applied tests and the associated requirements are shown in Table 8.10.

Table 8.10 Test verification (Continued)

Type of test	Test verification
Static test and centrifuge test.	• Check/qualification of structural strength in particularly the primary structure and critical interfaces • Verify (partially) the stiffness matrix
Modal survey test	• identify the natural frequencies ω_i, vibration modes $\{\phi_i\}$ and the modal damping ratios ζ_i to support the verification of the mathematical model which is used in loads cycles, and the CDLA
Shaker sine vibration test	• Support the verification of the spacecraft mathematical model (amplification from launcher spacecraft interface input to various spacecraft parts) • Qualification of secondary structures • Qualification of the spacecraft system by performing functional tests after the shaker vibration at qualification test input • Flight acceptance of the spacecraft system by performing functional tests often shaker vibration at flight test input
Acoustic test	• Verification and qualification of spacecraft system to acoustic environment which might be experienced by the spacecraft during flight • Qualification of the spacecraft system by performing functional tests after the acoustic test at qualification level. In case units are represented with dummies the random level is measured at unit dummy interfaces. Subsequently this input is employed for unit qualification at subsystem and unit level. • Flight acceptance of the spacecraft system by performing functional test after acoustic tests at flight level.
Shaker random vibration test	• Qualification of electronic units subjected to random (acoustically generated) flight environment
Shock test	• Spacecraft verification and qualification due to shock type of loads (pyrotechnic and mechanically induced shocks)