Satellite Technology (20EC81)

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Unit-V-B
Satellite Structures & Thermal Control

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Satellite Mechanical Subsystem

- The mechanical subsystem is responsible for designing and manufacturing the satellite structure while ensuring that the satellite can bear all the structural and thermal loads.
- Fully functional satellite for the whole period of mission life is the end goal, the mechanical subsystem contributes by making sure that the journey to the orbit and the harsh environment of space doesn't pose a problem to the other subsystems.

Satellite Mechanical Subsystem

- This makes three major subdivisions in the mechanical subsystem:
- **Structures:** Ensures the structural stability of the satellite while launch and during the operations of the satellite.
- **Thermals:** Ensures that the satellite functions in varied thermal cycles presented by the harsh space conditions in the orbit.

Satellite Mechanical Subsystem

- **Integration:** Manufacturing and the integration of the various components of the satellite to ensure proper functioning of the satellite within the mass budget.

Satellite Mechanical Subsystem

- The Structures division provides the basic structure of the satellite known as the bus of the satellite.
- Also, the various mechanisms such as the deployable solar panels are designed by the structures team.
- The Thermals team makes sure that all the components of the satellite are within their operational temperature throughout the mission life.

Satellite Mechanical Subsystem

- Finally, the Integration team gets to do the coolest part of actually ‘building’ the satellite.
- They put together the whole satellite ensuring that there are no components on board which interfere with each other.

Requirements imposed by Mechanical Subsystem

- **Volume and Mass requirements**
  - The satellite's mass and volume are constrained by the maximum payload the launch vehicle can carry. To ensure that these constraints set for the satellite are met, mechanical subsystem poses mass and volume requirements on all the subsystems.

Requirements imposed by Mechanical Subsystem

- **Stiffness Requirements**
  – Mechanical subsystem poses stiffness requirements on the other subsystems. The other subsystems need to ensure that PCBs.

Requirements imposed by Mechanical Subsystem

- **Softwares required**
  - **Modeling and FEM Softwares**
    - ANSYS, Abaqus, NASTRAN, COMSOL, HYPERMESH, SOLIDWORKS
  - **Dynamic Softwares**
    - ADAMS, ANSYS

Satellite Structures

- The design of satellite structure is dependent on many factors.
  - Some of the most important factors are placement of components

Satellite Structures

- The design approach that is followed generally is the following:
  - first system engineering team prepares a Configuration Layout and Mass Budget and
  - structures team analyses it and decides the parameters like the material of satellite body and thickness of the material.

Satellite Structures

- This iterative process is followed till a design satisfying both system engineering and structural requirements is obtained.
- The various parameters that can be changed by structural subsystem are material properties of structure, geometric parameters like thickness of the structure and joining mechanisms.

Satellite Structures

- The structures team also designs the various mechanisms of the satellite and ensures reliability of the mechanisms in space conditions.
- The analysis of various loads encountered is done through simulations performed in Finite Element Modelling Software.
- There are many such software available in the market. To validate the results of the software, theoretical calculations are done and compared.

Requirements on Structures

- **Launch Vehicle Placement Requirements**
  - The satellite is to be launched on a launch vehicle.
  - Each launch vehicle has some specifications for different classes of satellites.
  - The satellite structure should be able to interface with the launch vehicle and hence meet the specifications of the launch vehicle.

Requirements on Structures

- **Launch Loading Requirements**
  - The satellite is carried to its orbit by a launch vehicle in a flight lasting about 17 minutes for Low Earth Orbits.
  - During this period, the vehicle experiences high levels of acceleration, vibrations and shocks which are transmitted to the payloads attached to the flight decks of the vehicle.

Requirements on Structures

– Launch loads experienced include static loads, vibration loads, acoustic loads and shocks and impose certain strict requirements on the structure of the satellite.

– Satellite structure should be able to withstand these loads during launch. All the components should be safe and working after the launch.

– The loading specification for which the launch vehicle interface is tested is assumed to be the loading data for the satellite during launch.

Requirements on Structures

- **Deployment Requirements**
  – Many satellites have various deployable components such as antennas.
  – The structures team has to ensure that the components are deployed when required.

Requirements on Structures

- **Transportation and Handling Requirements**
  - The satellite, once integrated, needs to be handled and transported to the launch site.
  - For this purpose the structures team needs to design a handle for the satellite and a satellite box for transportation.
  - Many factors such as electrostatic charge, vibration, contamination from humidity, pressure management, thermal control needs to be looked at.

Configuration Considerations

- Configuration Layout is a map explaining where each component is placed within as well as on the exterior of the satellite.
- While preparing the configuration layout of the satellite, many considerations have to be taken into account.

Configuration Considerations

- **Centre of Mass and Moment of Inertia**
  - The controls subsystem have requirements pertaining to the Centre of Mass and Moment of Inertia Matrix for the satellite.
- **Shadow on Solar Panels / Sun Sensors**
  - The components placed on the exterior of the satellite should not create shadows on the sun devices. The solar panels will produce less energy and the sensors will give wrong readings leading to power shortage or misalignment of the orientation of the satellite respectively.

Configuration Considerations

- **Communication Considerations**
  - If the antennas are fixed with respect to the satellite the antennas need to be placed such that the maximum gain is in the direction of earth vector.
  - Also, to establish link with the GPS satellites in GEO orbit, the antenna for the GPS receiver should be placed such that maximum gain is in the direction of the GEO satellites.

Configuration Considerations

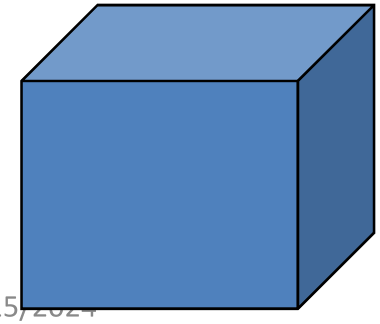
- **Integration Considerations**
  - To reduce the mass of the wires between the different electrical boards and sensors, the components should be placed as near as possible.
- **Electromagnetic Considerations**
  - Some components are sensitive to the electromagnetic inductions produced due to other components, special care has to taken while placing these components.

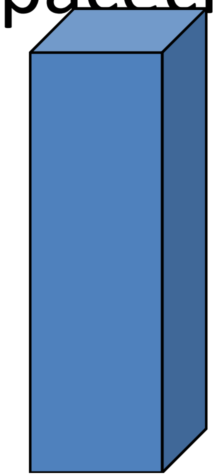
Design and analysis of satellite structures

- Structural Loads
  - Static (constant with time)
    - External
      - Weight of supported components during integration
    - Internal
      - Pressurized tanks
      - Mechanical preloads
      - Thermo-elastic loads
  - Dynamic (time varying)
    - External
      - Transport to launch site
      - Launch vehicle
      - Wind
      - Attitude control actuators
    - Internal
      - Thermal cycling
      - Mechanism operation

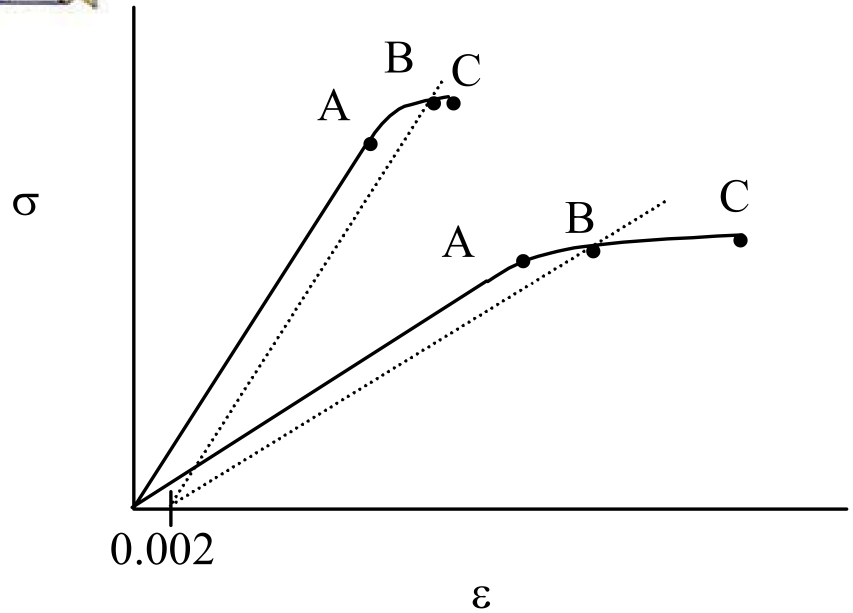
Design and analysis of satellite structures

- The 3 over-riding design criteria for space vehicle structures are:
  - Strength: ability to support a static load
  - Stiffness (Rigidity): measure of flexibility
  - Stability (Buckling): resistance to collapse under compression
- For simplicity, we will consider only spacecraft that resemble beams





Strength



A: Proportional Limit
B: Yield Point
(0.2% residual strain)
C: Ultimate Failure

Stress:

$$\sigma = \frac{Load}{Area} = \frac{P}{A}$$

Strain:

$$\varepsilon = \frac{\Delta L}{L}$$

Young's Modulus:

$$E = \frac{\sigma}{\varepsilon}$$

Poisson's Ratio: $\upsilon = \dfrac{\varepsilon_{lateral}}{\varepsilon_{axial}}$

Stiffness

- Natural frequency is the frequency at which an unforced, vibrating system will vibrate

$$f_n = \frac{1}{2\pi}\sqrt{\frac{k}{m}}$$

$$where$$

$$k = \text{stiffness(spring constant)}$$

$$m = \text{mass}$$

- When vibrating freely, a single degree of freedom system will always vibrate at the same frequency, regardless of amplitude

Stiffness

- Without energy dissapation, harmonic motion will go on forever. Of course, things do quit vibrating eventually.
- A damping force is one that resists vibration and dissipates energy, normally through heat (friction).
- A viscous damping force is proportional to velocity; we typically assume viscous damping to simplify analysis. Assuming a spring is linear-elastic,

$$m\ddot{x}(t) + c\dot{x}(t) + kx(t) = F(t)$$

$$where$$

$$m = \text{mass}$$

$$k = \text{stiffness}$$

$$c = \text{damping factor}$$

$$F(t) = \text{externally applied force as a function of time}$$

$$x(t), \dot{x}(t), \ddot{x}(t) = \text{position, velocity, acceleration}$$

Stability

- Theoretically, a linear-elastic column will buckle at a critical, or Euler Buckling Load, P_{cr}, given by

$$P_{cr} \equiv \frac{\pi^2 EI}{(L')^2}$$

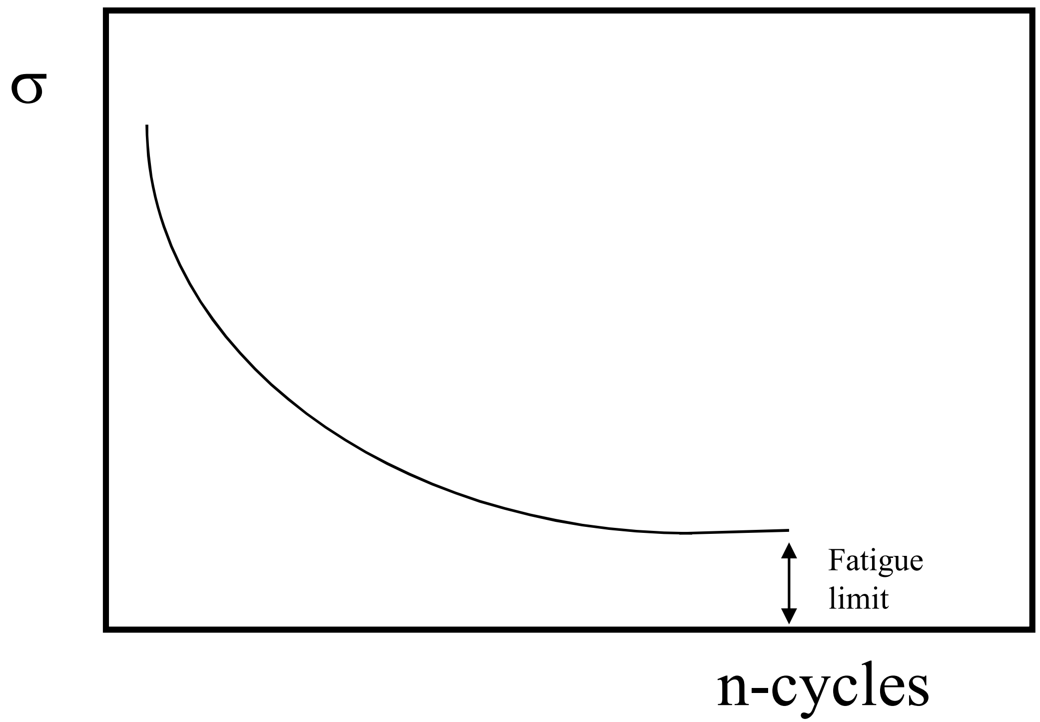
$$where$$

$$L' = \text{effective length} = \text{2L (for beam)}$$

- This equation applies only if the axial stress at buckling (P_{cr}/A) does not exceed the materials proportional limit.

Cyclic Failure

- Fatigue failure is caused by repeated, cyclical loading of a component at a load well below ultimate or yield



It is difficult (or impossible) to accurately predict the actual fatigue limit for a given part. The only sure way is to test to failure.

Structural Design

- Design Stress x Factor Safety < Allowable Stress
- Allowable Stress depends on
  - Type of stress
  - Material used

Option	Design Factors of Safety			
	Critical for Personnel Safety		Not Critical for Personnel Safety	
	Yield	Ultimate	Yield	Ultimate
1) Ultimate test of a dedicated qualification article	1.1	1.4	1.0	1.25
2) Proof test of all flight structures	1.1	1.4	1.1	1.25
3) Proof test of one flight unit of a fleet	1.25	1.4	1.25	1.4
4) No structural test	1.6	2.25	1.6	2.0

SSAM, Table 12.5 SMAD, Table 11.54

(Source: DOD-HDBK-343, MIL-HDBK-340 and MSFC-HDBK-505A offer similar options.)

Material Selection

Performance Characteristics
• Stiffness (Young's modulus and Poisson's ratio) • Rupture and yield strength (allowable stresses) • Ductility (elongation) • Fatigue resistance and fracture toughness • Mass density • Corrosion resistance • Creep resistance • Wear or galling resistance • Outgassing • Thermal conductivity, absorptivity, and emissivity • Coefficient of thermal expansion

Cost, Schedule, and Risk
• Availability • Cost of raw material • Cost of developing processes and tooling • Cost of processing (recurring) • Ease of controlling processes • Variability in key properties • Versatility of attachment options

Material Selection

Material	Advantages	Disadvantages
Aluminum	• High strength vs. weight • Ductile; tolerant of concentrated stresses • Easy to machine • Low density; efficient in compression	• Relatively low strength vs. volume • Low hardness • High coefficient of thermal expansion
Steel	• High strength • Wide range of strength, hardness, and ductility obtained by treatment	• Not efficient for stability (high density) • Most are hard to machine • Magnetic
Heat-resistant alloy	• High strength vs. volume • Strength retained at high temperatures • Ductile	• Not efficient for stability (high density) • Not as hard as some steels
Magnesium	• Low density – very efficient for stability	• Susceptible to corrosions • Low strength vs. volume
Titanium	• High strength vs. weight • Low coefficient of thermal expansion	• Hard to machine • Poor fracture toughness if solution treated and aged
Beryllium	• High stiffness vs. density	• Low ductility and fracture toughness • Low short transverse properties • Toxic
Composite	• Can be tailored for high stiffness, high strength, and extremely low coefficient of thermal expansion • Low density • Good in tension (e.g., pressurized tanks)	• Costly for low production volume; requires development program • Strength depends on workmanship; usually requires individual proof testing • Laminated composites are not as strong in compression • Brittle; can be hard to attach

Material Selection

Property	6061-T62 Al plate .25-2”	A286 Bar steel ≤2.499”	Ti-6Al-4V Bar 2” (annealed)
Density (gm/cm^3)	2.71	7.95	4.43
Young’s Modulus, E (10^3 MPa)	69.0	201	110
Poisson’s ratio, ν	0.33	0.31	0.31
Allowable Tensile Ultimate stress, F_{tu} (MPa)	290	896	923
Allowable compressive yield stress, F_{cy} (MPa)	240	590	903
Allowable Shear Stress, F_{su} (MPa)	190	590	570
Thermal conductivity (W/mK)	150	12	7.3
Coefficient of thermal expansion, α, (10^{-6} m/m/°C)	22.9	23.0	8.5
Corrosion resistance	Good	Excellent	Excellent
Weld-ability	Good	Good	Fair
Machinability	Very Good	Good	Fair

Adapted from Sarafin, Spacecraft Structures and Mechanisms, 1998.

Need of thermal Control

- The temperature requirements of the instruments and equipment on board are the main factors in the design of the thermal control system.
- The goal of the TCS is to keep all the instruments working within their allowable temperature range.

Need of thermal Control

- All of the electronic instruments on board the spacecraft, such as cameras, data-collection devices, batteries, etc., have a fixed operating temperature range.
- Keeping these instruments in their optimal operational temperature range is crucial for every mission.

Examples

- Batteries, which have a very narrow operating range, typically between −5 and 20 °C.
- Propulsion components, which have a typical range of 5 to 40 °C for safety reasons, however, a wider range is acceptable.
- Cameras, which have a range of −30 to 40 °C.
- Solar arrays, which have a wide operating range of −150 to 100 °C.
- Infrared spectrometers, which have a range of −40 to 60 °C.

Externally induced thermal environment

- For a spacecraft the main environmental interactions are the energy coming from the Sun
- And the heat radiated to deep space by reflection from earth’s surface, atmosphere, clouds etc.
- In case of LEO, Earth's infrared emission, as well as the relatively short orbital period, less than 2 hours, and long eclipse duration causes temperature variations.
-

Externally induced thermal environment

- As a result, Small instruments or spacecraft appendages such as solar panels that have low thermal inertia can be seriously affected by this continuously changing environment and may require very specific thermal design solutions.
- In GEO, the Earth's influence is almost negligible, except for the shadowing during eclipses, which can vary in duration from zero at solstice to a maximum of 1.2 hours at equinox.

Externally induced thermal environment

- Long eclipses influence the design of both the spacecraft's insulation and heating systems.
- The seasonal variations in the direction and intensity of the solar input have a great impact on the design, complicating the heat transport by the need to convey most of the dissipated heat to the radiator in shadow, and the heat-rejection systems via the increased radiator area needed.

Externally induced thermal environment

- An interplanetary trajectory exposes spacecraft to a wide range of thermal environments more severe than those encountered around Earth's orbits.
- Interplanetary mission includes many different sub-scenarios depending on the particular celestial body.

Internally induced thermal environment

- Internal heat generated by the operation of the spacecraft itself.
- As neither convection nor conduction take place in space, the heat generated by electronic components, batteries and other elements has a great impact on the spacecraft’s temperatures.

Heat Transfer Mechanism

- **Internal to spacecraft**
  - Heat collection
    - Includes the removal of dissipated heat from the equipment in which it is created to avoid unwanted increases in the spacecraft's temperature.
  - Heat transport
    - Is taking the heat from where it is created to a radiating device.

Heat Transfer Mechanism

– Heat rejection
  - The heat collected and transported has to be rejected at an appropriate temperature to a heat sink, which is usually the surrounding space environment. The rejection temperature depends on the amount of heat involved, the temperature to be controlled and the temperature of the environment into which the device radiates the heat.

– Heat provision and storage.
  - Is to maintain a desired temperature level where heat has to be provided and suitable heat storage capability has to be foreseen.

Heat Transfer Mechanism

- **External heat load variations**
  - Environment interaction
    - Includes the interaction of the external surfaces of the spacecraft with the environment. Either the surfaces need to be protected from the environment, or there has to be improved interaction. Two main goals of environment interaction are the reduction or increase of absorbed environmental fluxes and reduction or increase of heat losses to the environment.

Thermal control Systems

- In spacecraft design, the function of the **thermal control system** (**TCS**) is to keep all the spacecraft's component systems within acceptable temperature ranges during all mission phases.
- It must cope with the external environment, which can vary in a wide range as the spacecraft is exposed to deep space or to solar or planetary flux, and with ejecting to space the internal heat generated by the operation of the spacecraft itself.

Thermal control Systems

- Thermal control is essential to guarantee the optimal performance and success of the mission because if a component is subjected to temperatures which are too high or too low, it could be damaged or its performance could be severely affected.
- Thermal control is also necessary to keep specific components (such as optical sensors, atomic clocks, etc.) within a specified temperature stability requirement, to ensure that they perform as efficiently as possible.

Active or passive systems

- The thermal control subsystem can be composed of both passive and active items and works in two ways:
  - Protects the equipment from overheating, either by thermal insulation from external heat fluxes (such as the Sun or the planetary infrared and albedo flux), or by proper heat removal from internal sources (such as the heat emitted by the internal electronic equipment).

Active or passive systems

– Protects the equipment from temperatures that are too low, by thermal insulation from external sinks, by enhanced heat absorption from external sources, or by heat release from internal sources.

Passive thermal control system (PTCS)

- Multi-layer insulation (MLI), which protects the spacecraft from excessive solar or planetary heating, as well as from excessive cooling when exposed to deep space.
- Coatings that change the thermo-optical properties of external surfaces.
- Thermal fillers to improve the thermal coupling at selected interfaces (for instance, on the thermal path between an electronic unit and its radiator).

Passive thermal control system (PTCS)

- Thermal washers to reduce the thermal coupling at selected interfaces.
- Thermal doublers to spread on the radiator surface the heat dissipated by equipment.
- Mirrors (secondary surface mirrors, SSM, or optical solar reflectors, OSR) to improve the heat rejection capability of the external radiators and at the same time to reduce the absorption of external solar fluxes.

Passive thermal control system (PTCS)

- Radioisotope heater units (RHU), used by some planetary and exploratory missions to produce heat for TCS purposes.

Active thermal control system (ATCS)

- Thermostatically controlled resistive electric heaters to keep the equipment temperature above its lower limit during the mission's cold phases.
- Louvers (which change the heat rejection capability to space as a function of temperature).
- Thermoelectric coolers.

Active thermal control system (ATCS)

- Fluid loops to transfer the heat emitted by equipment to the radiators. They can be:
  – single-phase loops, controlled by a pump;
  – two-phase loops, composed of heat pipes (HP), loop heat pipes (LHP) or capillary pumped loops (CPL).

Integration

- The final Flight Model (FM) is the one which is launched into the space and undergoes the mission of the satellite.
- Integration team’s work comes into the picture once all the subsystems are ready with their designs and the individual components are manufactured.
- Then the critical part of integrating the components comes up.

Integration

- Integration team makes an Integration Sequence, making sure than integration can be done in a fast and effective manner.
- Apart from the Flight Model (FM), the team also integrates a Qualification Model (QM) which is used for qualifying the satellite design, before the flight model is made.
- It is generally a replica of the flight model, which is passed through more rigorous tests than the flight model.

Integration

- It is generally mandated by space agencies to ensure that the satellite design is fit to go into their launch vehicle, along with giving a confidence to the team.
- The Flight Model is also tested in similar conditions as Qualification Model but the loads applied in qualified model are higher.

Thank You