Satellite Design Course

Spacecraft Configuration
Structural Design
Preliminary Design Methods

This material was developed by Rodger Farley (NASA GSFC) for ENAE 691 (Satellite Design)
Source of Launch Vehicle Loads

- MECO: POGO Transient Sine
- Pre-MECO: Max acceleration
- Max Q: Fairing clearance
- Transonic: Transient Sine, Acoustics, Random
- Steady Acceleration
- Separation of boosters
- Main engine and strap-on boosters: Acoustics, Random, Transient Sine
- Spin, Steady Acceleration
- Shock
- SECO: Transient Sine
- Fairing separation
- Spacecraft separation
- Deployments: Transient Impact
Typical Loads Time-Line Profile

- Max Q
- H2 vehicle to geo-transfer orbit

- Dynamic Pressure
- Altitude
- Relative Velocity
- Acceleration

- MECO
- SECO

- 1st stage
- 2nd stage
- 3rd stage

Time from Lift-off (sec)
Design Requirements, Loads

Material processing, assembling, handling
Testing
Transportation
Launch handling
Launch loads

• Steady state accelerations (max Q, end of burn stages)
• Sinusoidal vibrations, transient (lift off, transonic, MECO, SECO) and steady (resonant burn from solid rocket boosters)
• Acoustic and mechanical random accelerations (lift off, max Q)
• Shock vibrations (payload separation from upper stage adapter)
• Depressurization

Orbit Loads spin-up, de-spin, thermal, deployments, maneuvers
Configuration Types LEO stellar pointing

Low inclination  XTE, HST

High inclination  WIRE

Articulated solar array

Fixed solar array sun-sync orbit
Configuration Types GEO nadir pointing

Articulated s/a spin once per 24 hours

TDRSS A

Intelsat V
Configuration Types HEO

AXAF-1 CHANDRA required to stare uninterrupted

250m long wire booms, ¼ rpm
Configuration types misc

MAP at L2

Magellan to Venus

Cylindrical, body-mounted solar array

Sun shield

High-gain antenna

Lunar Prospector

Body-mounted s/a
Structural Configuration Examples

- Cylinders and Cones  high buckling resistance
- Box structures  hanging electronics and equipment
- Triangle structures  sometimes needed
- Rings  transition structures and interfaces
- Trusses  extending a ‘hard point’ (picking up point loads)
- As much as possible, payload connections should be kinematic
- Skin Frame, Honeycomb Panels, Machined Panels, Extrusions

NOTE: ALWAYS PROVIDE A STIFF AND DIRECT LOAD PATH! AVOID BENDING!

STRUCTURAL JOINTS ARE BEST IN SHEAR!
Structural examples

MAGELLAN  box, ring, truss
Structural examples

HESSI  ring, truss and deck

OAO  cylinder/stringer and decks
Structural examples

TRMM

Rain Radar

Reaction wheel ‘pyramid’

Instrument support truss and panel structure

Bus structure, cylinder, machined and honeycomb panels
Structural examples

Hybrid ‘one of everything’

Archetypical cylinder and box
"Egg crate" composite panel bus structure

Hard points created at intersections

EOS aqua, bus
Structural examples

COBE
STS version
5000 kg !

COBE
Delta II version
2171 kg !
Modular Assembly

Instrument Module – optics, detector
Bus Module (house keeping)
Propulsion Module

Modules allow separate organizations, procurements, building and testing schedules. It all comes together at observatory integration and test (I&T)

Interface control between modules is very important: structural, electrical, thermal
Anatomy of s/c

Axial viewing, telescope type

- Interface structure
- Reaction wheels
- Station-keeping hydrazine, polar mount
- Solid kick motor, equatorial mount
- Kinematic Flexure mounts to remove enforced displacement loads
- Deck-mounted or wall-mounted boxes
- Launch Vehicle adapter

Instrument module

Bus module

Propulsion module
Anatomy of s/c

Transverse viewing, Earth observing type

Heat of boxes radiating outward

'Egg-crate' extension

Cylinder-in-box

Instruments

FOV

Drag make-up

Propulsion module

Hydrazine tank equatorial mount on a skirt with a parallel load path! The primary structure barely knows it's there.
Attaching distortion-sensitive components

Ball-in-cone, Ball-in-vee, Ball on flat
Breathes from point ‘3’

Note: vee points towards cone

Bi-pod legs, tangential flexures
Breathes from center point

Rod flexures arranged in 3,2,1
Breathes from point ‘3’
General Arrangement Drawings

- Stowed configuration in Launch Vehicle
- Transition orbit configuration
- Final on-station deployed configuration
- Include Field of View (FOV) for instruments and thermal radiators, and communication antennas, and attitude control sensors
3-view layout, on-orbit configuration

Overall dimensions and field of views (FOV)

Antennas showing field of regard (FOR)
Make note of protrusions into payload envelope

Omni antennas

Solar array panels

Star trackers

Torquer bars

High gain antenna, stowed

PAF
Spacecraft Drawing in Launch Vehicle

- Fairing access port for the batteries
- Pre-launch electrical access “red-tag” item
- XTE in Delta II 10’ fairing
- Launch Vehicle electrical interface
Drawing of deployment phase

Pantograph deployment mechanism

EOS aqua
Payload Envelope

For many vehicles, if the spacecraft meets minimum lateral frequency requirements, then the envelope accounts for payload and fairing dynamic motions.

If frequency requirements are not met, or for protrusions outside the designated envelope, Coupled Loads Analysis (CLA) are required to qualify the design, in cooperation with the LV engineers.
Launch Vehicle Payload Adapters

6019  3-point adapter

6915  4-point adapter

“V-band” clamp, or “Marmon” clamp-band

Clamp-band

Shear Lip
### Structural Materials

<table>
<thead>
<tr>
<th>Material</th>
<th>ρ (kg/m³)</th>
<th>E (GPa)</th>
<th>Fty (MPa)</th>
<th>E/ρ</th>
<th>Fty/ρ</th>
<th>α (µm/m K˚)</th>
<th>κ (W/m K˚)</th>
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<tbody>
<tr>
<td><strong>Aluminum</strong></td>
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<td><strong>Ferrous</strong></td>
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<td>INVAR 36</td>
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<td><strong>Heat resistant</strong></td>
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<td><strong>Non-magnetic</strong></td>
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<tr>
<td>Inconel 718</td>
<td>8220</td>
<td>203</td>
<td>1034</td>
<td>25</td>
<td>125.7</td>
<td>23.0</td>
<td>12</td>
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</tbody>
</table>
Materials Guide Definitions

\[ \rho \quad \text{mass density} \]
\[ E \quad \text{Young’s modulus} \]
\[ F_{\text{ty}} \quad \text{material allowable yield strength} \quad \text{Note: Pa = Pascal = N/m}^2 \]
\[ \frac{E}{\rho} \quad \text{specific stiffness, the ratio of stiffness to density} \]
\[ \frac{F_{\text{ty}}}{\rho} \quad \text{specific strength, the ratio of strength to density} \]
\[ \alpha \quad \text{coefficient of thermal expansion CTE} \]
\[ \kappa \quad \text{coefficient of thermal conductivity} \]

Material Usage Conclusions: USE ALUMINUM WHEN YOU CAN!!!

Aluminum 7075 and Titanium 6Al-4V have the greatest strength to mass ratio

Beryllium has the greatest stiffness to mass ratio and high damping

4130 Steel has the greatest yield strength \hspace{1cm} \text{(expensive, toxic to machine, brittle)}

INVAR has the lowest coefficient of thermal expansion, but difficult to process

Titanium has the lowest thermal conductivity, good for metallic isolators
## Metallic Materials Usage Guide

<table>
<thead>
<tr>
<th>Material</th>
<th>Advantages</th>
<th>Disadvantages</th>
<th>Applications</th>
</tr>
</thead>
<tbody>
<tr>
<td>Aluminum</td>
<td>High strength to weight, good machining, low cost and available</td>
<td>Poor galling resistance, high CTE</td>
<td>Truss structure, skins, stringers, brackets, face sheets</td>
</tr>
<tr>
<td>Titanium</td>
<td>High strength to weight, low CTE, low thermal conductivity, good at high temperatures</td>
<td>Expensive, difficult to machine</td>
<td>Attach fittings for composites, thermal isolators, flexures</td>
</tr>
<tr>
<td>Steel</td>
<td>High stiffness, strength, low cost, weldable</td>
<td>Heavy, magnetic, oxidizes if not stainless steel. Stainless galls easily</td>
<td>Fastenbers, threaded parts, bearings and gears</td>
</tr>
<tr>
<td>Heat-resistant Steel</td>
<td>High stiffness, strength at high temperatures, oxidation resistance and non-magnetic</td>
<td>Heavy, difficult to machine</td>
<td>Fastenbers, high temperature parts</td>
</tr>
<tr>
<td>Beryllium</td>
<td>Very high stiffness to weight, low CTE</td>
<td>Expensive, brittle, toxic to machine</td>
<td>Mirrors, stiffness critical parts</td>
</tr>
</tbody>
</table>
## Composite Materials

<table>
<thead>
<tr>
<th><strong>Pros</strong></th>
<th><strong>Cons</strong></th>
</tr>
</thead>
<tbody>
<tr>
<td>Lightweight (Strength to Weight ratio)</td>
<td>Costly</td>
</tr>
<tr>
<td>Ability to tailor CTE</td>
<td>Tooling more exotic, expensive/specialized tooling (higher rpms, diamond tipped)</td>
</tr>
<tr>
<td>High Strength</td>
<td>Electrical bonding a problem</td>
</tr>
<tr>
<td>Good conductivity in plane</td>
<td>Some types of joints are more difficult to produce/design</td>
</tr>
<tr>
<td>Thermal property variation possible (K1100)</td>
<td>Fiber print through (whiskers)</td>
</tr>
<tr>
<td>Low distortion due to zero CTE possible</td>
<td>Upper temperature limit (Gel temperature)</td>
</tr>
<tr>
<td>Ability to coat with substances (SiO)</td>
<td>Moisture absorption / desorption / distortion</td>
</tr>
</tbody>
</table>

provided by Jeff Stewart
Composite Material Properties

Graphite Fiber Reinforced Plastic (GFRP) density ~ 1800 kg/m³

If aluminum foil layers are added to create a quasi-isotropic zero coefficient of thermal expansion (CTE < 0.1 x10⁻⁶ per C₀) then density ~ 2225 kg/m³

Aluminum foil layers are used to reduce mechanical shrinkage due to desorption / outgassing of water from the fibers and matrix (adhesive, ie. epoxy)

NOTE! A single pin hole in the aluminum foil will allow water desorption and shrinkage. This strategy is not one to trust…

Cynate esters are less hydroscopic than epoxies

Shrinkage of a graphite-epoxy optical metering structure due to desorption may be described as an asymptotic exponential (HST data):

\[
\text{Shrinkage} \sim 27 \left(1 - e^{-0.00113D}\right) \text{ microns per meter of length, where } D \text{ is the number of days in orbit.}
\]
Subsystem Mass Estimation Techniques

Preliminary Design Estimates for Instrument Mass

Approximate instrument mass densities, kg / m\(^3\)

- Spectrometers ~ 250
- Mass spectrometers ~ 800
- Synthetic aperture radar ~ 32
- Rain radars ~ 150 thickness / diameter ~ 0.2
- Cameras ~ 500
- Small instruments ~ 1000

Scaling Laws: If a smaller instrument exists as a model, then if SF is the linear dimension scale factor…

- Area proportional to SF\(^2\)
- Mass proportional to SF\(^3\)
- Area inertia proportional to SF\(^4\)
- Mass inertia proportional to SF\(^5\)
- Frequency proportional to 1/ SF
- Stress proportional to SF

BEWARE THE SQUARE-CUBE LAW! STRESS WILL INCREASE WITH SF!
## Typical List of Boxes, Bus Components

### ACS
- Reaction wheels
- Torquer bars
- Nutation damper
- Star trackers
- Inertial reference unit
- Earth scanner
- Digital sun sensor
- Coarse sun sensor
- Magnetometer
- ACE electrical box

### Communication
- S-band omni antenna
- S-band transponder
- X-band omni antenna
- X-band transmitter
- Parabolic dish reflector
- 2-axis gimbal
- Gimbal electronics
- Diplexers, RF switches
- Band reject filters
- Coaxial cable

### Power
- Batteries
- Solar array panels
- Articulation mechanisms
- Articulation electronics
- Array diode box
- Shunt dissipaters
- Power Supply Elec.
- Battery a/c ducting

### Mechanical
- Primary structure
- Deployment mechanisms
- Fittings, brackets, struts, equipment decks, cowling, hardware
- Payload Adapter Fitting

### Thermal
- Radiators
- Louvers
- Heat pipes
- Blankets
- Heaters
- Heat straps
- Sun shield
- Cryogenic pumps
- Cryostats

### Propulsion
- Propulsion tanks
- Pressurant tanks
- Thrusters
- Pressure sensors
- Filters
- Fill / drain valve
- Isolator valves
- Tubing

### Electrical
- C&DH box
- Wire harness
- Instrument electronics
- Instrument harness
Mass Estimation, mass fractions

Some Typical Mass Fractions for Preliminary Design

<table>
<thead>
<tr>
<th></th>
<th>Payload</th>
<th>Structure</th>
<th>Power</th>
<th>Electrical harness</th>
<th>ACS</th>
<th>Thermal</th>
<th>C&amp;DH</th>
<th>Comm</th>
<th>Propulsion, dry</th>
</tr>
</thead>
<tbody>
<tr>
<td>LEO nadir (GPM)</td>
<td>37 %</td>
<td>24 %</td>
<td>13 %</td>
<td>Fixed arrays</td>
<td>7 %</td>
<td>6 %</td>
<td>4 %</td>
<td>1 %</td>
<td>3 %</td>
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<td>5 %</td>
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<td></td>
<td></td>
<td>26 % w/fuel</td>
</tr>
<tr>
<td>LEO stellar (COBE)</td>
<td>52 %</td>
<td>14 %</td>
<td>12 %</td>
<td>spins at 1 rpm</td>
<td>8 %</td>
<td>8 %</td>
<td>2 %</td>
<td>3 %</td>
<td>1 %</td>
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<td>0 %</td>
</tr>
<tr>
<td>GEO nadir (DSP 15)</td>
<td>37 %</td>
<td>22 %</td>
<td>20 %</td>
<td>spins at 6 rpm</td>
<td>7 %</td>
<td>6 %</td>
<td>0.5 %</td>
<td>2 %</td>
<td>2 %</td>
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</table>

Mass fractions as percentage of total spacecraft observatory mass, **less fuel**

COBE with 52% payload fraction is unusually high and not representative

ACS is Attitude Control System, C&DH is Command and Data Handling, Comm is Communication subsystem
Mass Estimation, Refinements

The largest contributors to mass are (*neglecting the instrument payload*)

- Structural subsystem *(primary, secondary)*
- Propulsion fuel mass, if required
- Power subsystem

Remember to target a mass margin of ~20% when compared to the throw weight of the launch vehicle and payload adapter capability. There must be room for growth, because evolving from the cartoon to the hardware, it *always* grows!

The following slides will show the ‘cheat-sheet’ for making preliminary estimates on some of these subsystems.
Mass Estimation with Mass Ratios

\[ \frac{M_{\text{payload}}}{M_{\text{Dry}}} \sim 0.4 \text{ for a first guess} \]

\[ M_{\text{Dry}} = \frac{M_{\text{payload}}}{M_{\text{payload}} / M_{\text{Dry}}} \]

First cut estimate on total dry mass

Calculate total orbital average power required \( P_{\text{oa}} \) as function of bus mass and payload requirements

Calculate \( M_{\text{Power}} \), and solar array area leading to a calculation on drag area

Calculate \( M_{\text{Fuel}} \), and \( M_{\text{PropDry}} \) with original \( M_{\text{Dry}} \) and drag area as guide

New Estimate of total dry mass:

\[ M_{\text{Dry}} = M_{\text{payload}} + M_{\text{Power}} + M_{\text{PropDry}} + M_{\text{Dry}} \left( \frac{M_{\text{Struc}}}{M_{\text{Dry}}} + \frac{M_{\text{Elec}}}{M_{\text{Dry}}} + \frac{M_{\text{CDH}}}{M_{\text{Dry}}} + \frac{M_{\text{ACS}}}{M_{\text{Dry}}} + \frac{M_{\text{Comm}}}{M_{\text{Dry}}} \right) \]

Iterate several times to achieve Total Dry Mass

Taking typical values:

\[ M_{\text{Dry}} = M_{\text{payload}} + M_{\text{Power}} + M_{\text{PropDry}} + 0.45 M_{\text{Dry}} \]

\[ M_{\text{Wet}} = M_{\text{Dry}} + M_{\text{Fuel}} \]

Total wet mass, observatory launch mass
Mass Estimation, Power Subsystem

Givens:

ALT circular orbit altitude, km

$P_{oa}$ required orbital average power, watts

AF area factor, if articulated arrays AF = 1, if omni-directional in one plane, AF = 3.14, spherical coverage

AF = 4

$\eta_{cell}$ standard cell efficiency, 0.145 silicon, 0.18 gallium, 0.25 multijunction

If the required orbital average power is not settled, then estimate with:

$P_{oa} \sim P_{REQpayload} + 0.5 (M_{Dry} - M_{payload})$ watts, mass in kg

Power for payload instruments and associated electronics

Dry bus mass, kg
Mass Estimation, Power Subsystem

\[ \text{EtoDratio} = 3.2 \cdot ( \text{ALT} + 50 )^{-0.2} - 0.3 \]

\[ A_{psa} = \frac{P_{oa}}{997 \cdot \eta_{cell}} \cdot \left( \frac{1}{0.9} + \frac{\text{EtoDratio}}{0.54} \right) \]

Maximum eclipse-to-daylight time ratio, estimate good for 300<ALT<1000

Required solar array projected area normal to sun, m²

\[ A_{sa} = A_F \cdot A_{psa} \]

Physical panel area, m²

Total solar array panel area considering energy balance going thru eclipse and geometric area factors, m²
Mass Estimation, Power Subsystem

Mass of cells, insulation, wires, terminal boards:

\[ M_{\text{elec}} \sim 3.8 \times A_{\text{SA}} \quad \text{for multi-junction cells} \quad \text{kg} \]

\[ M_{\text{elec}} \sim 3.4 \times A_{\text{SA}} \quad \text{for silicon cells} \quad \text{kg} \]

Mass of honeycomb substrates:

\[ M_{\text{substrates}} \sim 2.5 \times A_{\text{SA}} \quad \text{for aluminum panels} \quad \text{kg} \]

\[ M_{\text{substrates}} \sim 2.0 \times A_{\text{SA}} \quad \text{for composite panels} \quad \text{kg} \]

Since multi-junction cells usually go on composite substrates, and silicon cells went on aluminum substrates, it all comes out in the wash:

\[ M_{\text{PANELS}} \sim 5.85 \times A_{\text{SA}} \quad \text{kg} \]

Mass of solar array panels, electrical and structure:

\[ M_{\text{PANELS}} = M_{\text{elec}} + M_{\text{substrates}} \quad \text{kg} \]
Mass Estimation, Power Subsystem

\[ M_{\text{battery}} = 0.06 \ P_{\text{oa}} \]

Battery mass, kg  
Ni-Cads

\[ M_{\text{PSE}} = 0.04 \ P_{\text{oa}} \]

Power systems electronics, kg

\[ M_{\text{SAD}} = 0.33 \ M_{\text{panels}} \]

Solar array drives and electronics, kg

\[ M_{\text{SAdeploy}} = 0.27 \ M_{\text{panels}} \]

Solar array retention and deployment mechanisms, kg

\[ M_{\text{Power}} = M_{\text{panels}} + M_{\text{SAD}} + M_{\text{SAdeploy}} + M_{\text{battery}} + M_{\text{PSE}} \]

Total power subsystem mass estimate, kg

(this can vary greatly due to the peak power input and charging profile allowed. ‘High noon’ power input can be enormous sometimes, depending on orbit inclination and drag reduction techniques)
Mass Estimation, **Propulsion Subsystem**

**Givens:**
- \( \text{Isp} \): fuel specific thrust, seconds (227 for hydrazine, 307 bi-prop)
- \( \Delta V \): deltaV required for maneuvers during the mission, m/s
- \( T_M \): mission time in orbit, years
- \( \text{YSM} \): years since last solar maximum, 0 to 11 years
- \( A_P \): projected area in the velocity direction, orbital average, \( m^2 \)
- \( \text{ALT} \): circular orbit flight altitude, km
- \( M_{\text{dry}} \): total spacecraft observatory mass, dry of fuel, kg

\[
\text{Area}_{\text{SAoa}} = \frac{\text{Area}_{\text{SA}}}{\pi} \left( 1 - \cos\left( \frac{\pi}{\text{EtoD} + 1} \right) \right)
\]

Solar array orbital average projected drag area for a tracking solar array that feathers during eclipse
Mass Estimation, Propulsion Subsystem

Density\textsubscript{\text{max}} = 4.18 \times 10^{-9} \times e^{(-0.0136 \text{ ALT})}

Approximate maximum atmospheric density at the altitude ALT (km), kg / m\textsuperscript{3}

good for 300 < ALT < 700 km

DF = 1 - 0.9 \sin [\pi \times \text{YSM} / 11]

Density factor, influenced by the 11 year solar maximum cycle (\sin() argument in radians)

Density\textsubscript{\text{atm}} = \text{DF} \times \text{Density\textsubscript{\text{max}}}

Corrected atmospheric density, kg / m\textsuperscript{3}

\[ V_{\text{circular}} = \sqrt{\frac{3.986 \times 10^{11}}{\sqrt{(6371 + \text{ALT})}}} \]

Circular orbit velocity, m/s

\[ C_D = 2.2 \quad \text{Assumed Drag coefficient} \]

Drag = \frac{1}{2} \cdot \text{Density\textsubscript{\text{atm}}} \cdot V_{\text{circular}}^2 \cdot C_D \cdot A_P

Drag force, Newtons

\[ M_{\text{fuelDRAG}} = \frac{\text{Drag} \cdot T_M \cdot 31.536 \times 10^6}{\text{Isp} \cdot g} \]

Fuel mass for drag make-up, kg

If the mission requires altitude control, this is the approximate fuel mass for drag over the mission life

Note: \( g = 9.81 \text{ m/s}^2 \)
Mass Estimation, Propulsion Subsystem

Fuel mass for maneuvering

If maneuvers are conducted at the beginning of the mission (attaining proper orbit):

$$M_{\text{fuelMANV}} = \left( M_{\text{dry}} + M_{\text{fuelDRAG}} \right) \left[ e^{\frac{\Delta V}{(I_{\text{sp}} \cdot g)}} - 1 \right]$$

Maneuvering fuel mass, kg beginning of mission case

If the fuel is to be saved for a de-orbit maneuver:

$$M_{\text{fuelMANV}} = M_{\text{dry}} \left[ e^{\frac{\Delta V}{(I_{\text{sp}} \cdot g)}} - 1 \right]$$

Maneuvering fuel mass, kg end of mission case (de-orbit)

Or expressed as a ratio:

$$\frac{M_{\text{fuelMANV}}}{M_{\text{dry}}} = \left[ e^{\frac{\Delta V}{(I_{\text{sp}} \cdot g)}} - 1 \right]$$
Mass Estimation, Propulsion Subsystem

Approximate $\Delta V$ to de-orbit to a 50km perigee:

- From 400km circular $\sim 100$ m/s
- From 500km circular $\sim 130$ m/s
- From 600km circular $\sim 160$ m/s
- From 700km circular $\sim 185$ m/s

If the thrusters are too small for one large $\Delta V$ de-orbit maneuver, then 2-3 times the calculated fuel maybe required (TRMM experience)

Add 10% for ‘ullage’

$$M_{\text{fuel}} = M_{\text{fuel,DRAG}} + M_{\text{fuel,MANV}}$$

Total fuel mass, kg

$$M_{\text{PropDry}} = 0.25 \cdot M_{\text{fuel}}$$

Propulsion system mass, dry, kg

tank, thrusters, lines, etc...

$$M_{\text{wet}} = M_{\text{dry}} + M_{\text{fuel}}$$

Total spacecraft observatory mass with fuel, kg

Remember, the dry mass estimate now includes the mass of the dry prop system components. This means iterating with a better dry mass estimate for a better fuel calculation.
Vibration Primer

• A vibrating structure can be thought of as the superposition of many ‘mode’ shapes
• Each mode has a particular natural resonant frequency and distortion shape
• If the resonant frequencies are sufficiently separated, then the structural response can be estimated by treating each mode as a single-degree-of-freedom (sdoF) system

3 usual flavors of vibration:
Harmonic Random Impulsive
Harmonic (sinusoidal) Vibration, sdof

\[ f_{\text{natural}} = \frac{1}{2\pi} \sqrt{\frac{K}{M}} \text{ Hz} \]

For low values of damping

Natural frequency of vibration

Recall:

\[ 2\pi f = \omega \] ‘Circular frequency’, radians / s

Critical Damping \( C_c \)

\[ C_c = 2\sqrt{M \cdot K} \]

Damping Ratio \( \xi \)

\[ \xi = \frac{C}{C_c} \]

0.05 typical
Harmonic Vibration, sdoF con’t

Transmissibility, or transfer function; the ratio of output over input

\[ T = \frac{1}{\sqrt{\left(1 - \left(\frac{f}{f_n}\right)^2\right)^2 + \left(2\xi\left(\frac{f}{f_n}\right)^2\right)^2}} \]

Harmonic output acceleration = Harmonic input acceleration \times T

Amplification at Resonance

\[ Q = \frac{1}{2\xi} \]

Typical structural values for Q: 10 to 20

For small amplitude (jitter), or cryo temperatures: \( Q > 100 \)

Resonant Load Acceleration = input acceleration \times Q

\( f \) is the applied input sinusoidal frequency, Hz

\( f_n \) is the natural resonant frequency
Natural Frequency Estimation

evenly distributed mass and stiffness

Length \( L \) \( \text{m} \)

Mass \( M \) \( \text{kg} \)

Cross-sectional area \( A \) \( \text{m}^2 \)

Area inertia \( I \) \( \text{m}^4 \)

\[ f_{\text{axial}} = \frac{1}{2\pi} \sqrt{\frac{E \cdot A}{0.34 \cdot M \cdot L}} \]

\( f \) frequency in Hz

\[ f_{\text{lateral}} = \frac{1}{2\pi} \sqrt{\frac{3 \cdot E \cdot I}{0.24 \cdot M \cdot L^3}} \]

\( E \) is the Young's Modulus of Elasticity of the structural material
Natural Frequency Estimation
discrete and distributed mass and stiffness

Length $L$ m
Tip Mass $M_{\text{tip}}$ kg
Beam Mass $M$ kg
Cross-sectional area $A$ m$^2$
Area inertia $I$ m$^4$

$E$ is the Young’s Modulus of Elasticity of the structural material

Natural Frequency, axial

$$f_{\text{axial}} = \frac{1}{2\pi} \sqrt{\frac{E \cdot A}{(M_{\text{tip}} + 0.34M) \cdot L}}$$

frequency in Hz

Natural Frequency, lateral

$$f_{\text{lateral}} = \frac{1}{2\pi} \sqrt{\frac{3 \cdot E \cdot I}{(M_{\text{tip}} + 0.24M) \cdot L^3}}$$
Base-Driven Random Vibration

• Random loads affect mostly smaller, high frequency components, not the overall spacecraft structure.

Input spectrum ‘So’ described in power spectral density $g^2$ per Hz

Miles’ equation gives a statistical equivalent peak g load for a single degree of freedom system, 68% of the time ($1\sigma$). For 99.73% confidence ($3\sigma$), multiply by 3.

$$G_{RMS} = \sqrt{\frac{1}{2} f_n Q S_o}$$

fn is the natural frequency of the system, Q is the resonant amplification, and So is the input acceleration power spectral density, $g^2$/Hz.
Deployable Boom, equations for impact torque

\[ Q_{\text{max}} = I_o \cdot \dot{\theta} \cdot 2\pi \cdot f_N \]

- \( f_N \): Natural frequency at lock-in (Hz)
- \( C \): Damper (N-m per rad/s)
- \( T \): Applied spring torque (N-m)
- \( I_o \): Rotational mass inertia about point “o”, (kg-m^2)

\[ \dot{\theta} = \frac{T}{C} \]

Boom rotational velocity, rad/s

Dynamic system is critically damped

\( d\theta/dt \) can just be the velocity at time of impact if not critically damped

Maximum Impact torque at lock-in, N-m
Static Beam Deflections

For Quick Hand Calculations, these are the most common and useful

Cantilever beam length $L$ and properties $E, I$

\[
y = \frac{PL^3}{3EI} + \frac{ML^2}{2EI}
\]
\[
\theta = \frac{PL^2}{2EI} + \frac{ML}{EI}
\]

Point Load and Tip Moment Deflections

CTE = $\alpha$

Top surface at $T$, bottom at $T - \Delta T$

Linear gradient thru thickness $t$

\[
y = \frac{\alpha \Delta T L^2}{2t}
\]
\[
R = \frac{t}{(\alpha \Delta T)}
\]
\[
\theta = \frac{L}{R}
\]

Thermal Gradient Deflections
Developing Limit Loads for Structural Design

- **Quasi-Static loads**
  - Linear and rotational accelerations + dynamic
  Dynamic Loads included in Quasi-Static:
  - Harmonic vibration (sinusoidal)
  - Random vibration (mostly of acoustic origin)
  - Vibro-acoustic for light panels

- **Shock loads**

- **Thermal loads**

- **Jitter**

Don’t forget, there are payload mass vs. c.g. height limitations for the launch vehicle’s payload adapter fitting
## Launch Loads and Fundamental Frequency

<table>
<thead>
<tr>
<th>Launch Vehicle</th>
<th>Axial Load Factor $N_{\text{axial}}$, g’s</th>
<th>Lateral Load Factor $N_{\text{lateral}}$, g’s</th>
</tr>
</thead>
<tbody>
<tr>
<td>Delta IV</td>
<td>6.5 +/- 1.0 27Hz min</td>
<td>+/- 2.0 +/- 0.7 10Hz min</td>
</tr>
<tr>
<td>steady state</td>
<td></td>
<td></td>
</tr>
<tr>
<td>dynamic</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Atlas II</td>
<td>5.5 +/- 2.0 15Hz min</td>
<td>+/- 0.4 +/- 1.2 10Hz min</td>
</tr>
<tr>
<td>steady state</td>
<td></td>
<td></td>
</tr>
<tr>
<td>dynamic</td>
<td></td>
<td></td>
</tr>
<tr>
<td>H-II</td>
<td>5.2 +/- 5.0 30Hz min</td>
<td>2.8 +/- 2.0 10Hz min</td>
</tr>
<tr>
<td>steady state</td>
<td></td>
<td></td>
</tr>
<tr>
<td>dynamic</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Ariane V</td>
<td>4.6 +/- less than 1.0 18Hz min</td>
<td>+/- 0.25 +/- 0.8 10Hz min</td>
</tr>
<tr>
<td>steady state</td>
<td></td>
<td></td>
</tr>
<tr>
<td>dynamic</td>
<td></td>
<td></td>
</tr>
<tr>
<td>Delta II</td>
<td>6.7 for 2000kg +/- 1.0 35Hz min</td>
<td>+/- 3.0 +/- 0.7 15Hz min</td>
</tr>
<tr>
<td>steady state</td>
<td></td>
<td></td>
</tr>
<tr>
<td>dynamic</td>
<td></td>
<td></td>
</tr>
</tbody>
</table>

**Note:** the static and dynamic loads do not occur at the same time (usually)
Rigid-body accelerations

- **Linear force** \( F = m \times a = m \times g \times N_{\text{factor}} \)
  - \( N \) is the load factor in g’s
  - Low frequency sinusoidal below first natural frequency will produce ‘near-static’ acceleration
    \( a = A \times (2 \pi f)^2 \) where \( f \) is the driving frequency and \( A \) is the amplitude of sinusoidal motion

- **Rotational torque** \( Q = I \times \alpha \)

- **Centrifugal force** \( F_c = m \times r \times \Omega^2 \)

\[ x = A \sin(\omega t) \]
\[ v = A\omega \cos(\omega t) \]
\[ a = -A\omega^2 \sin(\omega t) \]
Combining the loads to form ‘quasi-static’ levels

Limit Load = Static + dynamic + resonant + random

(low frequency)

Note: The maximum values for each usually occur at different times in the launch environment, luckily.

The primary structure will have a different limit load than attached components.

Solar arrays and other low area-density exposed components will react to vibro-acoustic loads.
Loads for overall structure, example

If the structure meets minimum frequency requirements from the launch vehicle, then the low frequency sinusoidal environment is enveloped in the limit loads. Secondary structures with low natural frequencies may couple in, however, and should be analyzed separately.

The structural analyst will determine which load case produces the greatest combined axial-bending stress in the structure (I, A, mass and c.g. height)
Loads for a component, example

\[ M_{\text{payload}} = 500 \text{ kg} \]
\[ M_{\text{kickmotor}} = 50 \text{ kg} \quad \text{dry mass} \]
\[ T_{\text{static}} = 30000 \text{ N} \]
\[ T_{\text{dynamic}} = \pm 10\% \ T_{\text{static}} \text{ at } 150 \text{ Hz} \]
\[ m = 1 \text{ kg} \]
\[ f_{n_{\text{axial}}} = 145 \text{ Hz with } Q = 15 \]
\[ f_{n_{\text{lateral}}} = 50 \text{ Hz with } Q = 15 \]
\[ L = 0.5m \]
\[ R = 1m \]
\[ \Omega = 10.5 \text{ rad/s (100 rpm)} \]
Random input \( S_0 = 0.015 \text{ g}^2 \text{ per Hz} \)

The axial frequency is sufficiently close to the driven dynamic frequency that we can consider the axial mode to be in resonance.
Spinning upper stage example, con’t

Deflection $y$:

$$y = R \frac{(\Omega / \omega)^2}{[1 - (\Omega / \omega)^2]}$$

Axial-to-lateral coupling $AtoL = \frac{y}{L}$

Equivalent additional lateral load = $AtoL \times P_{axial}$

Axial-to-lateral coupling:

- Length $L$
- Radius $R$
- Deflection $y$

Lateral circular bending frequency, rad/s

$$\omega = 2 \pi f_{n_{\text{Lateral}}}$$
Loads for a component, con’t

Static axial acceleration \( G_{\text{static}}^A = \frac{T_{\text{static}}}{g} \left( M_{\text{payload}} + M_{\text{kickmotor}} \right) \) g’s

Dynamic axial acceleration \( G_{\text{dyn}}^A = \frac{T_{\text{dynamic}} \times Q}{g} \left( M_{\text{payload}} + M_{\text{kickmotor}} \right) \) g’s

Random axial acceleration \( G_{\text{rmd}}^A = 3 \sqrt{0.5 \pi f_{\text{axial}} \frac{Q}{S_0}} \) g’s

Axial Limit Load Factor, g’s, \( N_{\text{axial}} = G_{\text{static}}^A + G_{\text{dyn}}^A + G_{\text{rmd}}^A \) g’s

Axial Limit Load, Newtons, \( P_{\text{axial}} = g \times m \times N_{\text{axial}} \)

Static lateral acceleration \( G_{\text{static}}^L = \frac{R \Omega^2}{g} + A_{\text{toL}} \times N_{\text{axial}} \) g’s

Random lateral acceleration \( G_{\text{rmd}}^L = A_{\text{toL}} \times 3 \sqrt{0.5 \pi f_{\text{lateral}} \frac{Q}{S_0}} \) g’s

Lateral Limit Load Factor, g’s, \( N_{\text{lateral}} = G_{\text{static}}^L + G_{\text{dyn}}^L + G_{\text{rmd}}^L \) g’s

Lateral Limit Load, Newtons \( P_{\text{lateral}} = g \times m \times N_{\text{lateral}} \)

Moment at boom base \( M = L \times P_{\text{lateral}} \)
Component Loads, Mass-Acceleration Curve

JPL acceleration curve for component sizing

This curve envelopes limit loads for small components under 500 kg

Apply acceleration load separately in critical direction

Add static 2.5 g in launch vehicle thrust direction

Applicable Launch Vehicles:
- STS
- Titan
- Atlas
- Delta
- Ariane
- H2
- Proton
- Scout

Simplified design curve for components on 'appendage-like' structures under 80 Hz fundamental frequency
Sizing the Primary Structure
rigidity, strength and stability

Factors of safety  NASA / INDUSTRY, metallic structures

<table>
<thead>
<tr>
<th>Factors of safety</th>
<th>Verification by Test</th>
<th>Verification by Analysis</th>
</tr>
</thead>
<tbody>
<tr>
<td>FS yield</td>
<td>1.25 / 1.10</td>
<td>2.0 / 1.6</td>
</tr>
<tr>
<td>FS ultimate</td>
<td>1.4 / 1.25</td>
<td>2.6 / 2.0</td>
</tr>
</tbody>
</table>

Factors of safety for buckling (stability) elements ~ \( FS_{\text{buckling}} = 1.4 \)
(stability very dependent on boundary conditions….so watch out!)
For example, in many cases, the primary structure is some form of cylinder.
Sizing for rigidity (frequency)

Working the equations backwards….

When given frequency requirements from the launch vehicle users guide for 1\textsuperscript{st} major axial frequency and 1\textsuperscript{st} major lateral frequency, $f_{\text{axial}}$, $f_{\text{lateral}}$:

\begin{align*}
EA &= (2 \pi f_{\text{axial}})^2 (M_{\text{tip}} + 0.34 M) L \\
EI &= \frac{1}{3} (2 \pi f_{\text{lateral}})^2 (M_{\text{tip}} + 0.24 M) L^3
\end{align*}

Material modulus $E$ times cross sectional area $A$

Material modulus $E$ times bending inertia $I$

For a thin-walled cylinder:

$I = \pi R^3 t$, \quad or \quad t = I / (\pi R^3)$

$A = 2\pi R t$, \quad or \quad t = A / (2\pi R)$

Select material for $E$, usually aluminum

e.g.: 7075-T6 \quad E = 71 \times 10^9 \text{ N/m}^2

Determine the driving requirement resulting in the thickest wall $t$.

Recalculate $A$ and $I$ with the chosen $t$.

Calculate for a 10\% – 15\% frequency margin

Tapering thickness will drop frequency 5\% to 12\%, but greatly reduce structural mass
Sizing for Strength

Design Loads using limit loads and factors of safety

\[ P_{\text{Lateral}_{\text{des}}} = (M + M_{\text{tip}}) g N_{\text{limit}_L} \]  \hspace{1cm} \text{Lateral Design Load}

\[ P_{\text{Axial}_{\text{des}}} = (M + M_{\text{tip}}) g N_{\text{limit}_A} \]  \hspace{1cm} \text{Axial Design Load}

Recalling from mechanics of materials:

- axial stress = \( P/A \), bending stress = \( Mc/I \) (in a cylinder, the max shear and max compressive stress occur in different areas and so for preliminary design shear is not considered)

Max stress \( \sigma_{\text{max}} = P_{\text{Axial}_{\text{des}}} / A + (P_{\text{Lateral}_{\text{des}}} L_{\text{cg}} R) / I \)

Margin of safety \( MS = \left\{ \sigma_{\text{allowable}} / (FS \times \sigma_{\text{max}}) \right\} - 1 \quad 0 < MS \text{ acceptable} \)

For 7075-T6 aluminum, the yield allowable \( \sigma_{\text{allowable}} = 503 \times 10^6 \text{ N/m}^2 \)

With less stress the higher up, the more tapered the structure can be, saving mass
Sizing for Structural Stability

Determine the critical buckling stress for the cylinder

In the general case of a cone:

\[ \sigma_{CR} = 0.6 \gamma E \frac{t}{r} \]

\[ \gamma = 1.0 - 0.901 (1 - e^{-\phi}) \]

\[ \phi = \frac{1}{16} \sqrt{\frac{r}{t}} \]

Cone Angle \( \alpha \)
Thickness \( t \)

Margin of safety MS = \( \{\sigma_{CR} / (FS_{buckling} \times \sigma_{max})\} - 1 \)

0 < MS acceptable

Check top and bottom of cone: \( \sigma_{max} \), \( I \), moment arm will be different

Sheet and stringer construction will save ~ 25% mass
Structural Subsystem Mass

For all 3 cases of stiffness, strength, and stability, optimization calls for ‘tapering’ of the structure.

The frequency may drop between 5% to 12% with tapering

But the primary structural mass savings may be 25% to 35% - a good trade

Secondary structure (brackets, truss points, interfaces….) may equal or exceed the primary structure. An efficient structure, assume secondary structure = 1.0 x primary structure. A typical structure, assume 1.5.

So, if the mass calculated for the un-optimized constant-wall thickness cylinder (primary structure) is \( M_{\text{CYL}} \), then the typical structure (primary + secondary):

\[
M_{\text{STRUCTURE}} \sim (2.0 \text{ to } 3.5) \times M_{\text{CYL}}
\]

This number can vary significantly