EE-585 – Space Mission Design and Operations

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Ecole Polytechnique Fédérale de Lausanne

Week 11 – 29 Nov 2024

Today's outline

Spacecraft attitude control

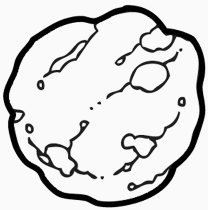
Spacecraft electrical power system (EPS)

Ascent to space

Re-entries

Spacecraft attitude control

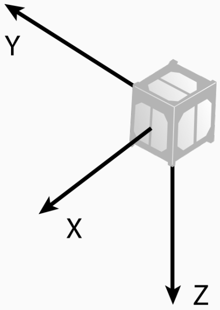
Orientation of an object in space



Any free object in space – like an asteroid or an uncontrolled spacecraft – is generally in a slow rotation (with respect to an inertial reference frame).

Orientation and rotation rates influenced by:

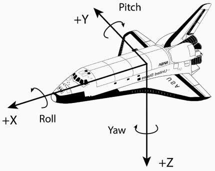
- Gravity gradient
- Solar radiation and solar wind
- Atmosphere
- Magnetic field
- ...



This is unacceptable for most applications → need for a spacecraft attitude determination and control system (ADCS).

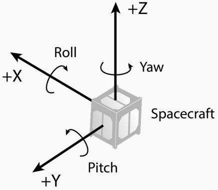
Attitude determination and control system – ADCS

The ADCS consists in measuring and maintaining, or changing in a controlled manner, the orientation of a coordinate system attached to the spacecraft with respect to an inertial or any other reference system. The attitude is maintained or controlled within a specified deadband.



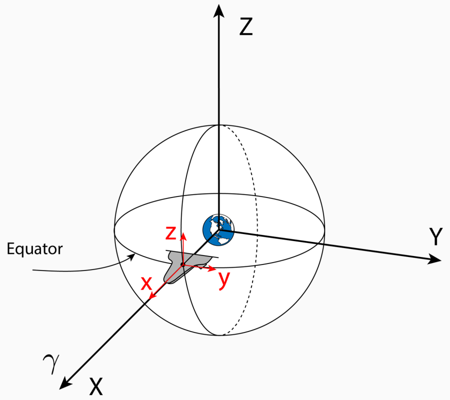
ADCS is a common abbreviation in Europe. In the US, you may also hear Attitude Measurement and Control System - AMCS.

ADCS is a subsystem of the larger Guidance, Navigation & Control which includes determination of the trajectory to follow (guidance), determination of state vector and attitude (navigation) and actuators (control).



J2000 geocentric inertial coordinate system

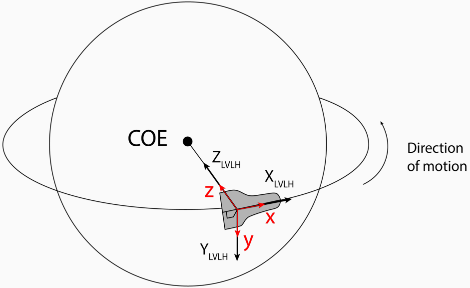
Example: Space shuttle at $(P, Y, R) = (0, 0, 0)_{\text{intertial}}$



J2000 means that the X-axis of this coordinate system is oriented to the vernal equinox ♈ of the year 2000

Local Vertical Loval Horizontal (LVLH) coordinate system

Example: Space shuttle at $(P, Y, R) = (0, 0, 0)_{\text{LVLH}}$

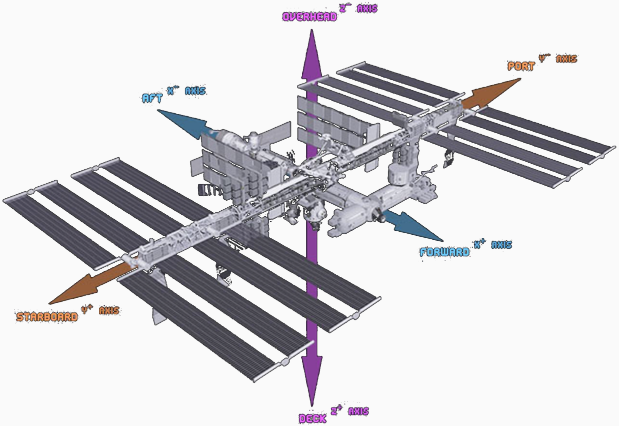


Frame centered at the Center of Mass of a spacecraft orbiting the Earth, +Z to the COE, +X in the direction of the velocity vector for a circular orbit.

ISS coordinate system

Usually the LVLH attitude of ISS is close to LVLH (0,0,0) with the X axis toward the velocity vector and Z down to the Center of the Earth.

The bias to LVLH (0,0,0) is to orient ISS close to the TEA or Torque Equilibrium Attitude. TEA is the attitude at which the gravitational torques and atmospheric torques best cancel each other over the orbit of about 93 minutes duration.



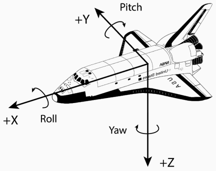
Credits: NASA

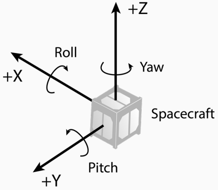
Euler angle rotation sequence

Rotations around X, Y, and Z axes are non commutative. An order should be chosen. This is the Euler sequence.

Usually the sequence is **YPR**. I.e. you change first the yaw, then the pitch and finally the roll angle.

For the ISS, its robotic systems (like grappling arm) and the space shuttle, the sequence is **PYR**.





Spacecraft attitude control methods

Passive or active control method?

Passive: gravity gradient.

Active: thrusters, spinning spacecraft, spinning wheels, or magnetic torquers.

Magnetic torquers can only be used on orbit around a body that has a magnetic field.

Attitude control systems (ACS):

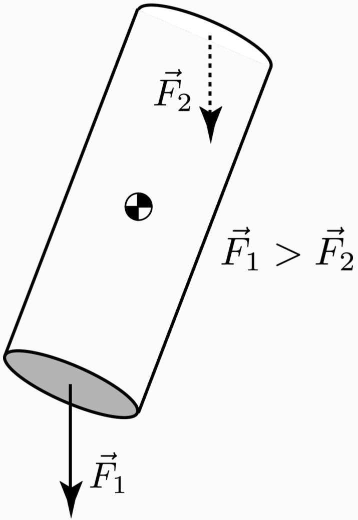
1. Gravity gradient
2. Magnetic torquers
3. Spinning spacecraft
4. Thrusters
5. Momentum devices: reaction wheels (RW) or Control Moment Gyros (CMG)

Gravity gradient

An elongated object in orbit around the Earth will take an orientation such that its long axis will be along the local vertical.

There could be oscillations (in plane and out of plane) around the Center of Gravity (CG), normally close to the Center of Mass (CM).

GC is passive will not be a precise attitude control system.



Magnetic torquers

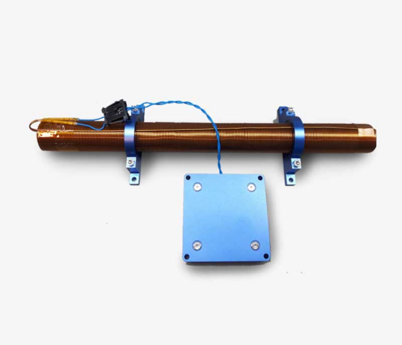
A magnetic torquer is an elongated bar with a wire coil wrapped around it and an external protection.

A current through the coil will produce a magnetic field which will try to align itself along the geo-magnetic field (or any strong magnetic field) with a torque T,

$$T = NBAI \sin\theta$$

where

T is the torque (Nm)
N is the number of loops in the coil
B is the magnetic field (Tesla)
$B(R_\oplus) = 3.1 \cdot 10^{-5}$ Tesla is B at Earth's surface



Magnetic Torquer MTQ800, AAC Clyde Space

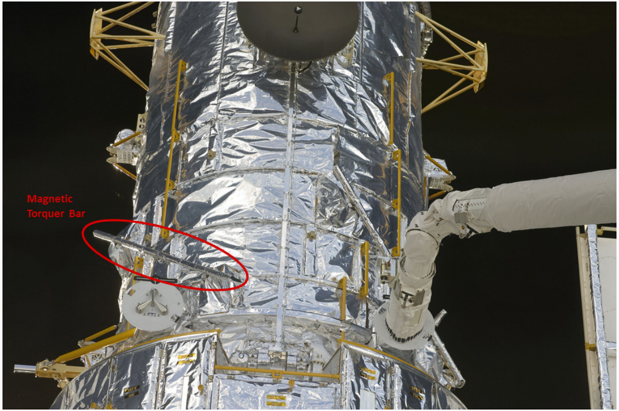
A is the area of the coil (m^2)
I is the current (A)
θ is the angle between the magnetic field and the bar

Magnetic torquers on spacecraft

Advantages: magnetic torquers are reliable (no moving part!), energy efficient and lightweight.

Disadvantages: need a strong $\vec{B}$-field, works only in LEO, not very precise.

Magnetic torquers are used if the orientation to spacecraft does not need to be extremely precise or as a system to desaturate Reaction Wheels (right: Hubble Space Telescope example).



Credits: NASA

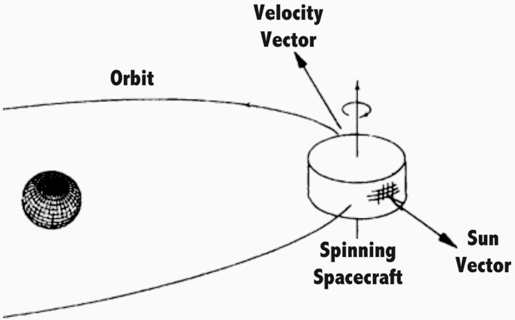
They can also be used as a de-tumbling mechanism.

Spinning spacecraft

Whole spacecraft spins along an axis, like a top.

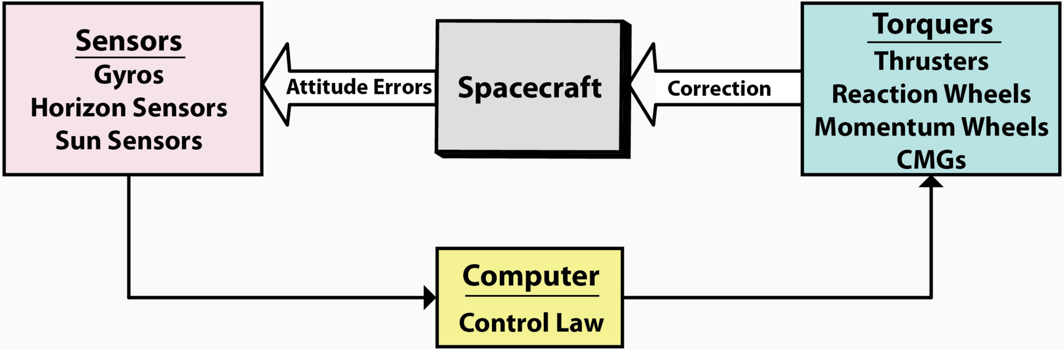
Advantages: cheap, propellant flow from tanks provided by inertial forces.

Disadvantages: low accuracy in controlled attitude ($0.3 - 1°$), translations only possible along rotation axis, pointing of antennas and other devices impossible except in the direction of spin.



Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

Closed loop attitude control: attitude control loop



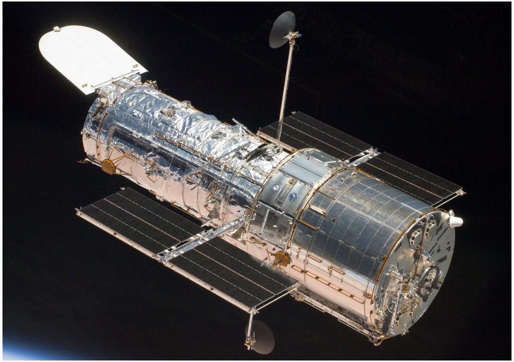
Required accuracy of attitude control systems

Better than 10° for solar arrays

$0.1 - 0.5^\circ$ for high gain antennas

$10^{-4} - 0.1^\circ$ for optical systems

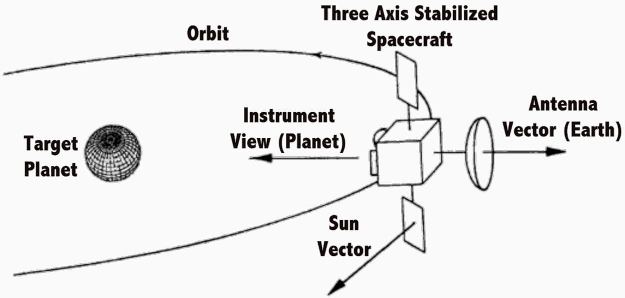
Hubble: 0.007 arcsec ($\approx 2 \cdot 10^{-6}$ deg) !



Credits: NASA

Three axis stabilised spacecraft

Spacecraft can be oriented in any direction and maintaining pointing.



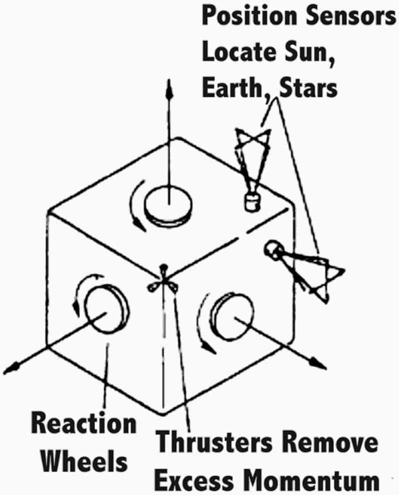
Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

Attitude controlled by thrusters, Reaction Wheels, or Control Moment Gyros (CMGs)

Three axis stabilisation

Advantages: any orientation possible, high pointing accuracy possible.

Disadvantages: complexity and price, complex redundancy architecture, need to ensure propellant supply from the tanks by other means than using the inertial forces.



Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

Main techniques for three axis stabilization

Thrusters: Examples are the Space Shuttle, Soyuz capsule, Crew Dragon, ISS Russian Orbital Segment (ROS).

Reaction (or Momentum) Wheels: Torques on the spacecraft induced by variations of the wheel's rotational speed. Requires thrusters or magnetic torquers to get out of wheels saturation. Hubble as an example.

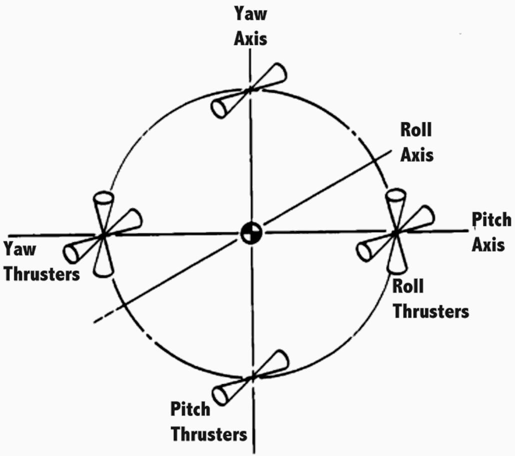
Control Moment Gyros or CMGs: Constant angular velocity. Mounted on gimbals. A torque generated along the input axis produces a corresponding torque reaction along the output axis. ISS in the USOS segment as an example.

Thruster-based attitude control: geometry

For an ACS based on thrusters and for 3-axis attitude controlled, 12 thrusters are required.

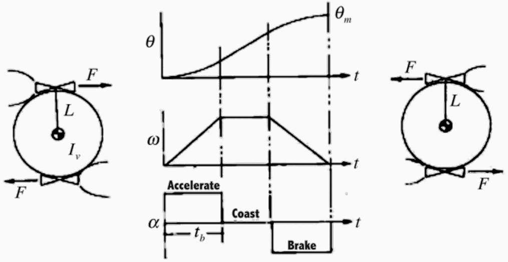
A minimum of **12 thrusters** is needed for **pure rotation** manoeuvres (without induced translation)
4 on each axis:

- 2 thrusters for acceleration impulse to initiate the rotation
- 2 opposite thrusters for the braking impulse.



Credits: NASA

Attitude manoeuvre around one axis using thrusters



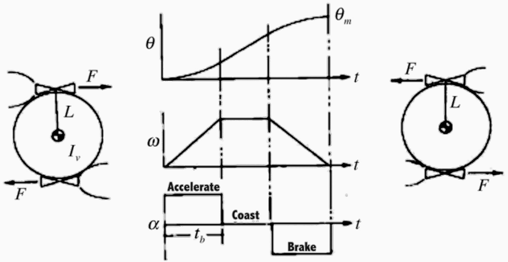
Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

F: thrust for each thruster (N)
T: torque (Nm)
n: number of thrusters used
L: perpendicular distance from the center of mass to the thrust vector (m)
θ: angle of rotation (rad)
ω: angular velocity (rad/s)
α: rotational acceleration (rad/s^2)
I_v: moment of inertia around the rotational axis (kg·m^2)
t_b: duration of the burn (s)

During the initial acceleration α of the angular velocity ω:

$$\alpha = \frac{T}{I_v} = \frac{nFL}{I_v}, \qquad \omega = \alpha t \implies \omega_{\max} = \alpha t_b = \frac{nFL}{I_v} t_b$$

Attitude manoeuvre around one axis using thrusters



Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

At burnout of the acceleration phase,

$$\theta = \frac{1}{2}\alpha t^2 = \frac{1}{2}\frac{nFL}{I_v}t_b^2$$

The braking phase is the same (including same t_b, and therefore):

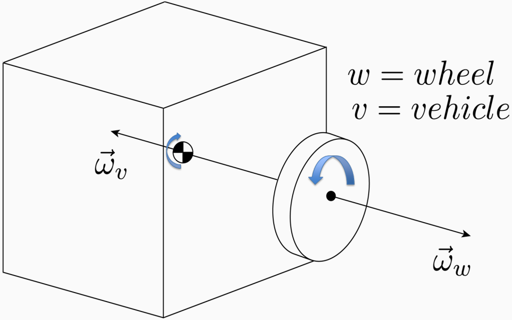
$$\begin{aligned}\theta_m &= \text{acceleration} + \text{coast} + \text{braking} \\ &= \frac{nFL}{I_v}t_b^2 + \frac{nFL}{I_v}t_b t_{\text{coast}}\end{aligned}$$

Determination of the propellant used in the manoeuvre:

$$I_{\text{sp}} = \frac{nF}{\dot{m}_p g_0} \implies m_p = 2t_b\dot{m}_p = \frac{2nF \cdot t_b}{I_{sp}g_0}$$

Reaction Wheel

Principle: If the angular rotation speed of the reaction wheel is increased, the angular rotation speed of the spacecraft will increase in the opposite direction.

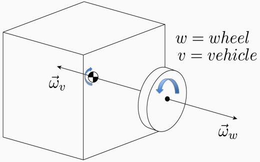


Angular drift of the spacecraft, around the reaction wheel axis, as a consequence of a manoeuvre of total time t_m duration:

$$\Delta\theta_v = \frac{1}{4}\frac{I_w}{I_v}\alpha_w t_m^2$$

t_m: total manoeuvre time on a triangular profile of ω_w rising linarly until time $t_m/2$, then decreasing linearly until time t_m.

Reaction Wheel – angular drift

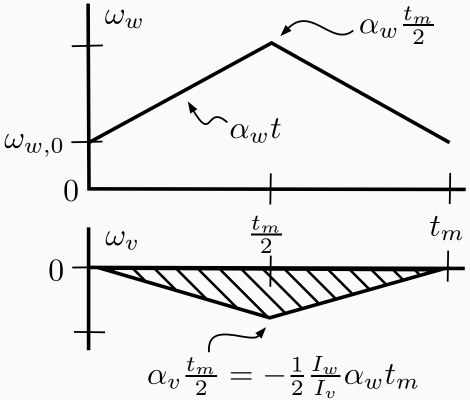


Initial conditions:

$$\omega_v = 0 \qquad \omega_w = \omega_{w,0}$$

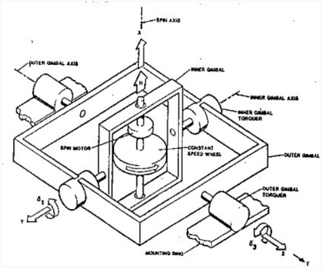
Conservation of angular momentum:

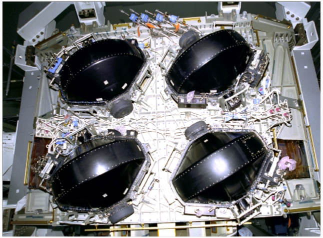
$$I_v\omega_v + I_w\omega_w = \text{cst}$$
$$I_v\alpha_v + I_w\alpha_w = 0$$
$$\Longrightarrow \quad \alpha_v = -\frac{I_w}{I_v}\alpha_w$$



$$\boxed{\Delta\theta_v = \text{surface of } \blacktriangledown = \frac{1}{4}\frac{I_w}{I_v}\alpha_w t_m^2}$$

Cluster of double gimbal Control Moment Gyroscopes (CMGs) on ISS



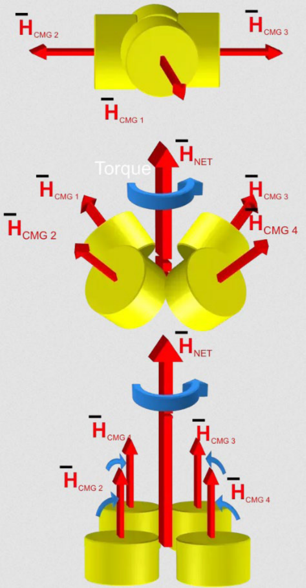


Credits: Wikipedia

ISS attitude control is provided by a cluster of 4 double gimbal CMGs. Each of these CMGs has a 100 kg wheel spinning at a constant 6600 rpm.

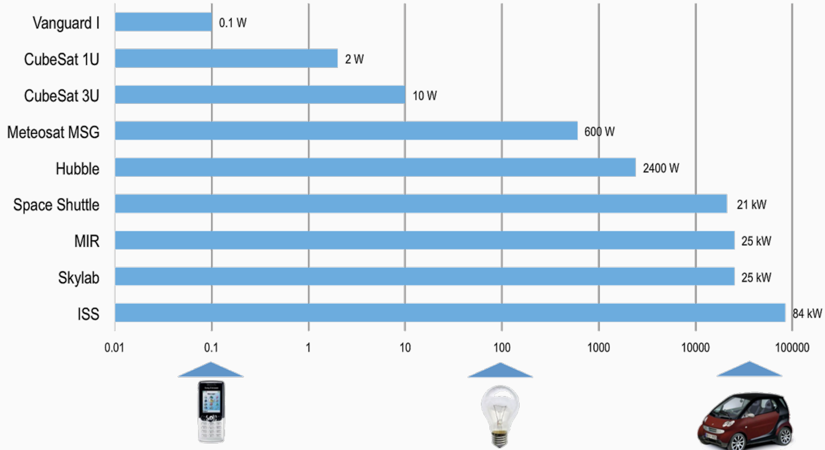
For each CMG, torques on the outer and inner gimbals reorient the wheels which modifies the cluster vector angular momentum which is the sum of the individual CMGs angular momenta.

The ISS is reoriented to satisfy the conservation of the overall angular momentum (ISS plus cluster).



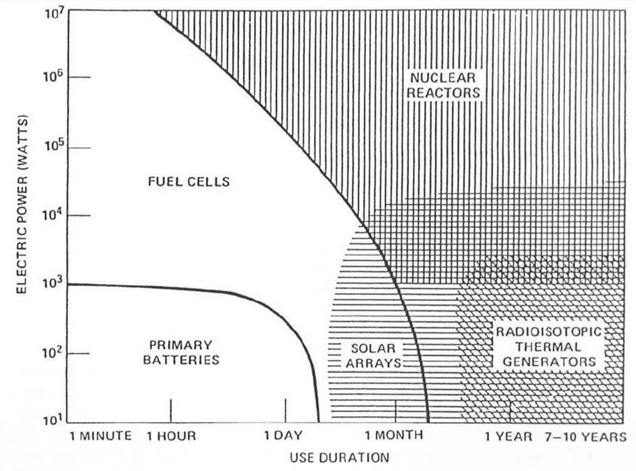
Spacecraft electrical power system (EPS)

Electrical power for various spacecraft



Most of a spacecraft subsystems require electrical power (computers, payload(s), antennas, com, ...). **Heat is also generated** (and incoming from Sun and Earth) $\rightarrow$ need for thermal design!

Operating regimes of spacecraft power sources

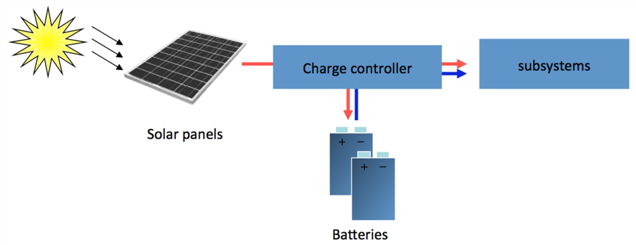


Energy source must be:

- Reliable
- Low mass
- Tolerant to space conditions

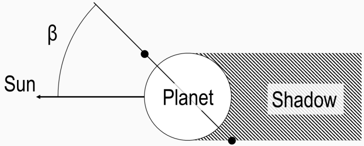
Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA

Operating regimes of spacecraft power sources



When panels are illuminated by the Sun, they provide power to subsystems and charge batteries.

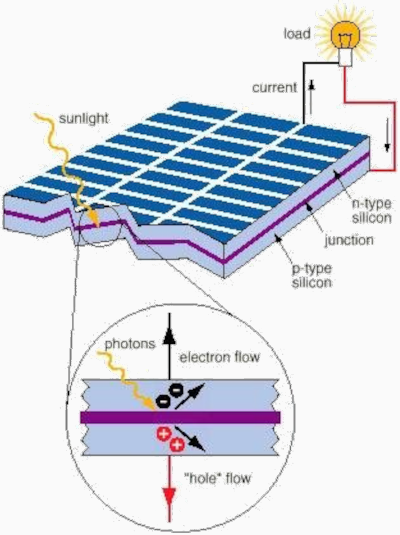
When panels are in Earth's shadow, batteries provide enough power to subsystems for a long enough time at the end of life. A dedicated charge controller manages these cycles.



Eclipse duration depends on:

- Orbital altitude
- The β angle between the orbital plane and the Sun's direction

Solar cells: principle and types



Credits: RESLAB

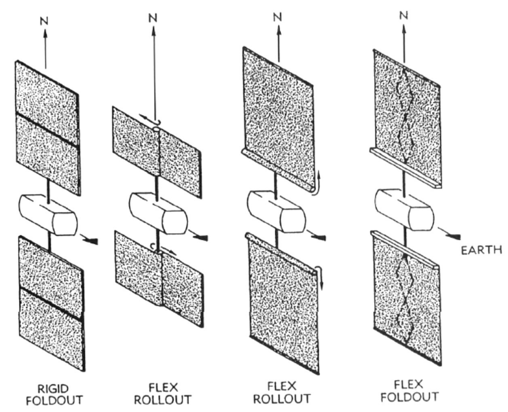
Principle:

- Upper layer: N doped semiconductor
- Lower layer: P doped semiconductor
- PN junction when irradiated by a photon flow displace electrons from N to P, holes from P to N
- Resulting voltage and current

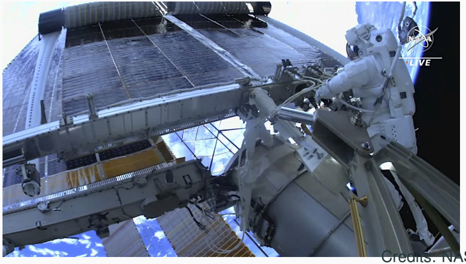
Cell types:

- Silicium:
  - Amorphous: Efficiency 5-10%, low cost
  - Poly-crystallines: Efficiency 10-13%.
  - Mono-crystallines: Efficiency 15-22%, high cost
- Gallium arsenide:
  - High efficiency around 30-40%, high cost.

Solar cells and solar panels

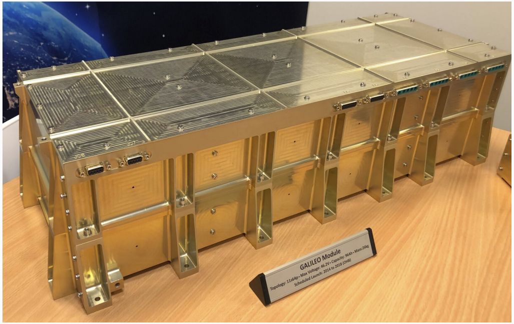


Credits: Charles D. Brown, *Elements of Spacecraft Design*, AIAA



Credits: NASA

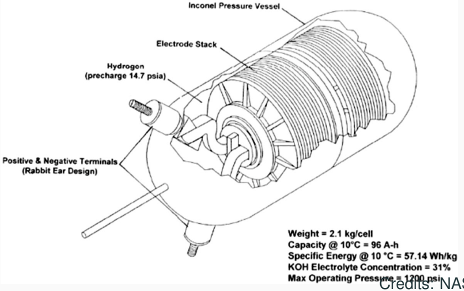
Batteries: types and characteristics



Credits: ESA

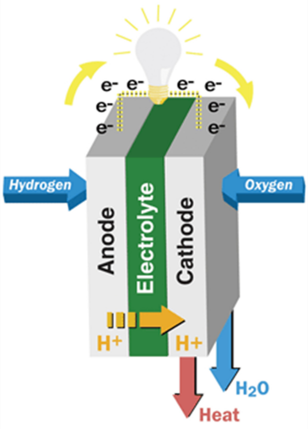
Lithium-Ion batteries, better specific energy ($\approx$ 150 Wh/kg)

Rechargeable batteries NiH_2 (Nickel Hydrogen), on HST and ISS, Long life and takes many charge/discharge cycles ($> 2 \times 10^4$)



Credits: NASA

Fuel Cells (FC)

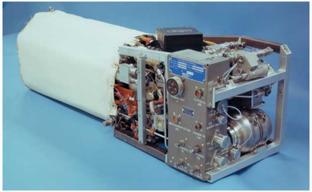


Credits: Wikipedia

Fuel cells convert chemical energy from reactants into electricity through a chemical reaction of positively charged hydrogen ions with oxygen (or other oxidizing agent).

They require a continuous source of reactants to sustain the chemical reaction. The by-products of this reaction are water and heat.

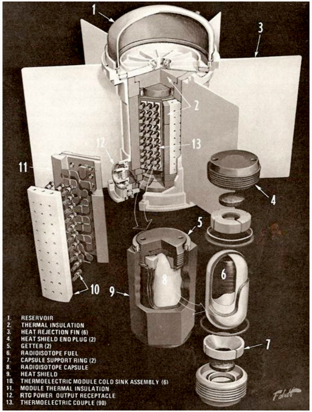
Fuel cells have been successfully used in the Gemini, Apollo and Shuttle programs. The three Shuttle fuel cells generated each around 7 kW.



Credits: NASA

One of three Shuttle Fuel Cell

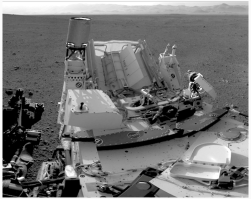
Radioisotope Thermoelectric Generators (RTG)

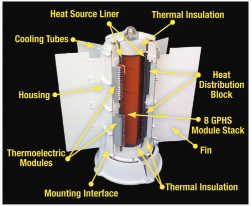


Credits: Wikipedia

A Radioisotope Thermoelectric Generator (RTG), uses the fact that radioactive materials (such as plutonium 238) to generate heat as they decay into non-radioactive materials. The heat is converted into electricity by an array of thermocouples.

RTGs are very reliable and long lasting, but of low efficiency (less than 10%). Used in the Apollo ALSEP program, Viking, Pioneer 10 and 11, Voyager 1 and 2, Cassini, New Horizons, Curiosity.





Credits: NASA

RTG installed in the Curiosity rover on the surface of Mars.

Want to work on satellite subsystems?

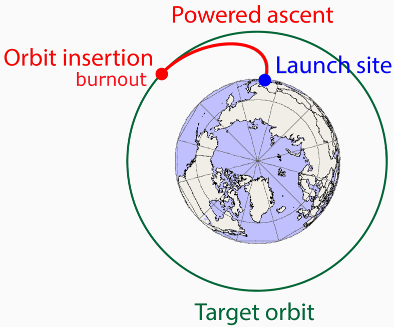
→ *Course ENG-411 / Concurrent engineering of space missions*
→ *Course ENG-510 / Space propulsion*
→ *Course ENG-580 / Introduction to the design of space mechanisms*
→ *Course EE-584 / Spacecraft design and system engineering*

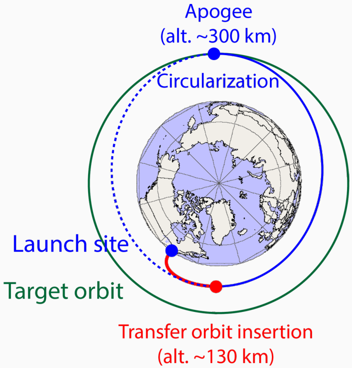
→ *Course EE-582 / Lessons learned from the space exploration*
→ *EPFL Minor in Space Technologies Course book*

Ascent to space

Orbit insertion

Orbit insertion consists in bringing a spacecraft to a desired stable orbit after a launch from the Earth surface.





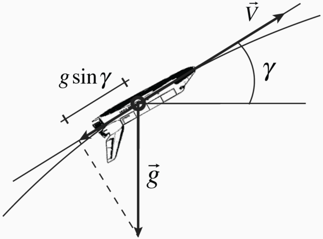
Direct insertion into orbit (left) or via a **transfer orbit** (right). The powered ascent uses either one, two or three stages until orbit insertion.

Losses during ascent to orbit

The real Δv achieved to reach orbit must be larger than the $\Delta \vec{v} = \vec{v}_{\text{circ}} - \vec{v}_{ground}$

$$\Delta v = g_0 I_{\text{sp}} \ln\left(\frac{m_i}{m_f}\right) - \left(\int_{t_0}^{t_f} g \sin\gamma \, \mathrm{d}t + \int_{t_0}^{t_f} \frac{D}{m} \mathrm{d}t\right)$$

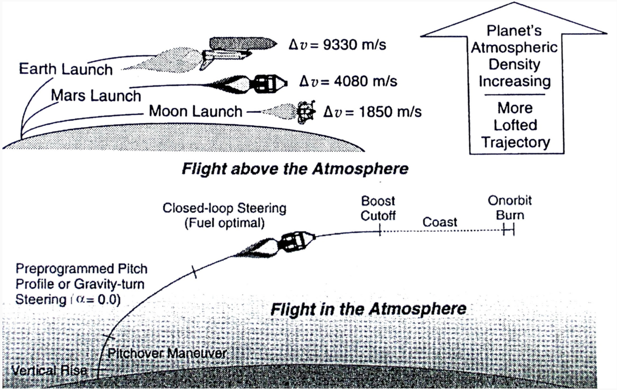
where D is the drag force (in N) and γ is the flight path angle.



Losses during ascent to orbit: **gravity loss** and **drag loss.**

The planned and actual ascent trajectory is shaped to minimize these losses.

Different cases of orbit insertion



Credits: Documention of the training division for NASA astronauts in the 90's.

For Earth launch, the ascent trajectory is significantly lofted because of the atmosphere.

On a planet with thinner atmosphere like Mars, loft is less necessary.

The case of the Moon: no atmosphere, only gravity loss during ascent to orbit. After a very short period of vertical launch the spacecraft tilts toward the desired direction.

Initial rotational velocity

A launch due East (towards 90°) takes the most advantage of the Earth's rotational velocity as $\vec{v}_{\mathrm{LV}} = \vec{v}_{\mathrm{rotational}} + \Delta\vec{v}_{\mathrm{LV}}$ (LV = launch vehicle).

At the equator,

$$v_{\mathrm{equator}} = \frac{2\pi R_{\oplus}}{T_s} \approx 0.465\,\mathrm{km/s}$$

This rotational velocity depends on the latitude ϕ of the launch base:

$$v_{\mathrm{rotational}} = v_{\mathrm{equator}} \cos\phi$$

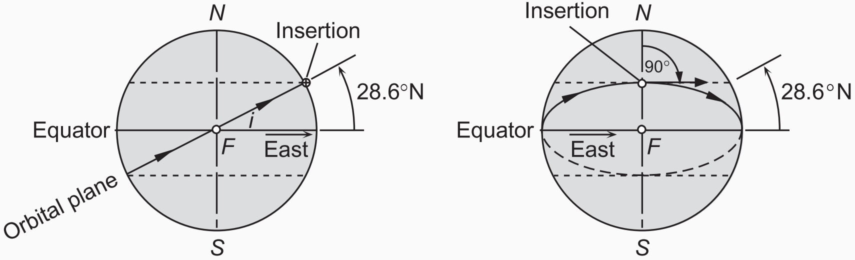
$\rightarrow$ launching eastwards from sites at low latitudes is the best option.

Minimal orbit's inclination

The inclination i of the insertion orbit is the latitude of the launch site ϕ if the launch is due East because the orbital plane must intersect the centre of the Earth:

$$\boxed{i \geq \phi}$$

Example for a launch due East from Kennedy Space Center (KSC) in Florida at $\phi \approx 28.6°$N.



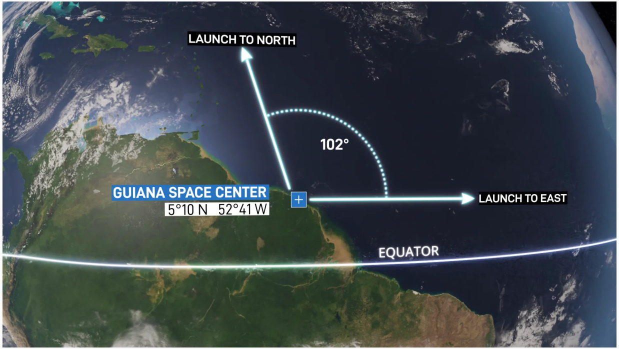
Credits: Curtis, *Orb. Mech. for Eng. Students*

Launch site restrictions

Some azimuths can be restricted to avoid flying over populated regions. This restricts the options for launch.

The proximity of the Guiana Space Centre (GSC, a.k.a. Kourou) to the equator give a 15% payload advantage over Cape Canaveral for 90 degree azimuth launches.

Inclinations from 5.2 degrees to 100.5 degrees are achievable.



Credits: Arianespace

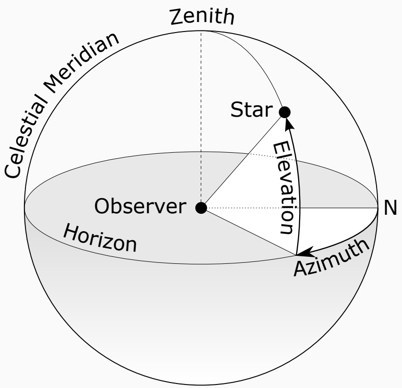
Launch azimuth A

Launch azimuth A is the flight direction at launch.

$A = 90^\circ$ is due East. If the launch direction is not directly eastward, the orbit will have an inclination $i > \phi$, launch latitude.

if $0^\circ < A < 90^\circ$ (northeast) or $90^\circ < A < 180^\circ$ (southeast) launches take only partial advantage of the Earth's rotational speed.
The orbit's $\phi < i < 90^\circ \rightarrow$ prograde orbits.

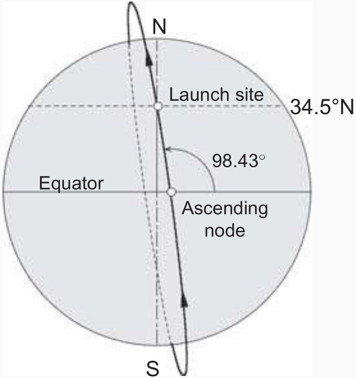
Launches to the West: retrograde orbits with
$i = 90^\circ < i < 180^\circ$.



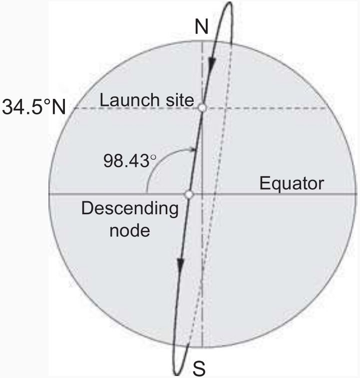
Credits: Adapted from Wikipedia, TWCarlson

Launches with different *A* but same *i*

There can be two choices of launch azimuth A for the same inclination i. Here is an example for a Sun-Synchronous Orbit. A determines the position of the ascending node with respect to the launch site.



Launch azimuth A = 349.8°



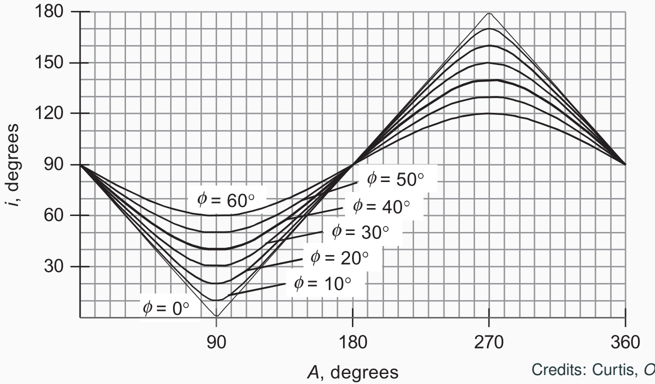
Launch azimuth A = 190.2°

Credits: Curtis, *Orb. Mech. for Eng. Students*

Achievable inclinations i

The inclination of the orbit i is linked to the latitude ϕ and the azimuth A:

$$\cos i = \cos\phi \sin A$$



Credits: Curtis, *Orb. Mech. for Eng. Students*

$\rightarrow$ launching from sites at low latitudes allow to reach a wider inclination range.

Launch vehicle staging

In free-space (i.e. no gravity or drag losses), the produced Δv is

$$\Delta v = I_{sp} g_0 \ln\left(\frac{m_i}{m_f}\right)$$

To maximise Δv, the final mass m_f must be minimised.

$$m_i = m_{\text{payload}} + \sum m_{\text{structures}} + m_p$$

$$m_f = m_{\text{payload}} + m_{\text{structure,final}}$$

If all of the structure of the launch vehicle is carried to orbit, $m_{\text{structure,i}} = m_{\text{structure,f}}$. To limit the negative impact of the structure on the performance, unnecessary structures are jettisoned. This allows to use different engines that are better adapted to sea-level or vacuum conditions. This is staging (*More on this in the exercises*).

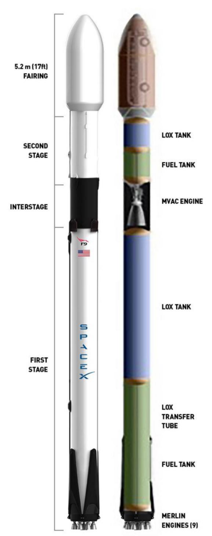


Figure 2-1: Falcon 9 overview

SpaceX

Effect of launch on the payloads

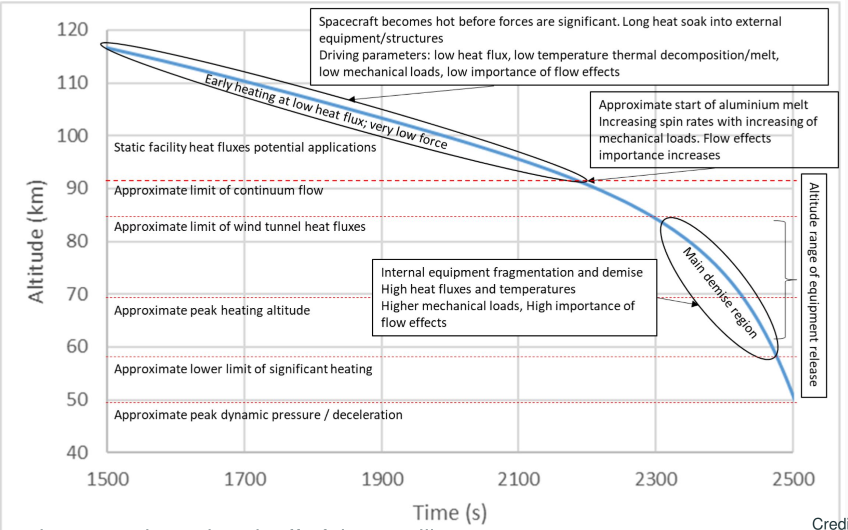
During launch payloads must undergo very high levels of vibrations and acoustic noise. This is one of the most risky phase of a spacecraft's mission.

Satellites must be extensively tested on the ground to ensure that they will survive the launch phase.

Credits: ESA/Airbus/RUAG

Re-entries

Main events during a strandard re-entry trajectory



Credits: *DIVE*, ESA

“equipment release” = pieces break off of the satellite structure

ATV Jules Verne re-entry on 29 Sep 2008

Credits: ESA/NASA

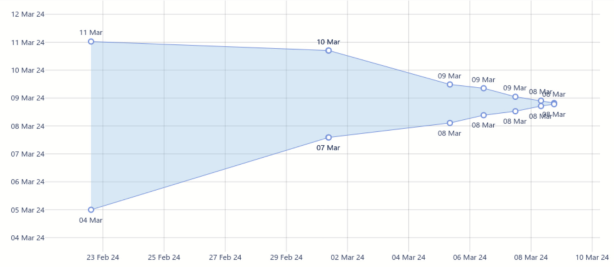
Photographed from aircraft. Breakup over the Pacific Ocean. It broke up at an altitude of 75 km.

Re-entry predictions

SpaceTrack and other entities (like EU SST) issue 60-day decay warnings on objects on their way to decay, Tracking and Impact Predictions messages ("TIP") ahead of re-entries and final decay message with approximate re-entry locations (or similar names and content).

TIPs contain a re-entry time prediction, but it has very large uncertainties:

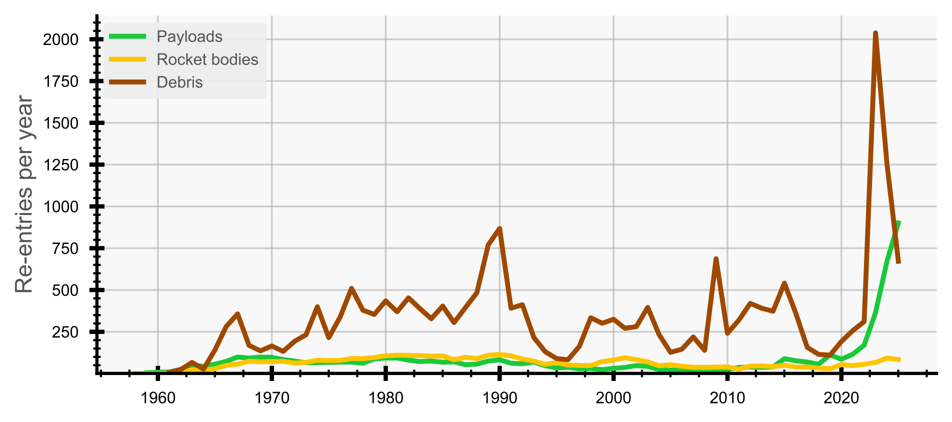
- 4 days ahead of re-entry, $\pm$20 hours (!)
- 3 days ahead, $\pm$15 hours (!)
- a few orbits ahead, $\pm$¼ orbit (!)



Re-entry window evolution, Credits: EU SST

Causes of uncertainties: mostly poor knowledge of the atmosphere and secondly state vector. Risk of causalities: extremely low.

Number of re-entries



The number of satellite re-entries is increasing rapidly from typically 50-75 per year to $\sim$ 700 because of the short lifetime of the satellites in the constellations and mega-constellations. Data as of 03 Nov 2024.

Consequences of satellite break-up in the atmosphere

RESEARCH ARTICLE | EARTH, ATMOSPHERIC, AND PLANETARY SCIENCES |

Metals from spacecraft reentry in stratospheric aerosol particles

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$\sim$ 10% of the aerosol particles in the stratosphere contain aluminum and other metals that originated from the "burn-up" of satellites and rocket stages during re-entry.

These measurements have broad implications for the stratosphere and higher altitudes. With many more launches planned in the coming decades, metals from spacecraft re-entries could induce changes in the stratospheric aerosol layer.

A typical 250-kg satellite can generate $\sim$ 30 kg of aluminium oxide nanoparticles.

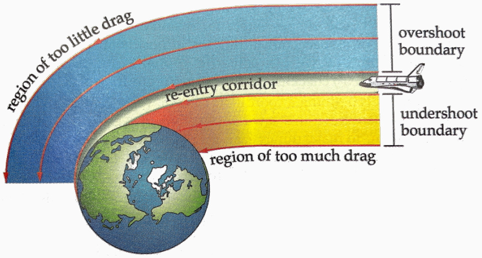
$\rightarrow$ More data needed, but this is a potential issue for the $\gtrsim 50,000$ satellites in a decade.

Re-entry corridor to a splash down or landing

Entry interface at around $\sim$ 80 km altitude. After, drag becomes the dominant force. The speed entry at interface v, the ballistic coefficient $BC = \frac{m}{C_D A_n}$ and the flight-path angle γ are driving parameters.

There are 3 competing requirements:

1. Deceleration: Human limit is about 12 g's for short duration ($\gtrsim$ 100 g's for structures).
2. Heating: Must withstand both total heat load and peak heating rate.
3. Accuracy of landing or impact: Function primarily of trajectory and vehicle design.



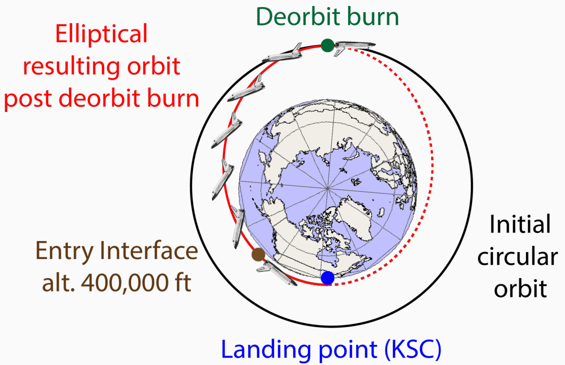
Credits: Sellers J., *Understanding Space*, 3rd Ed.

Size of the entry corridor: The size of the corridor depends on three constraints (deceleration, heating and accuracy).

Re-entry of the space shuttle

The deorbit burn was a braking manoeuvre, using the OMS (Orbital Maneuvering System) engines.

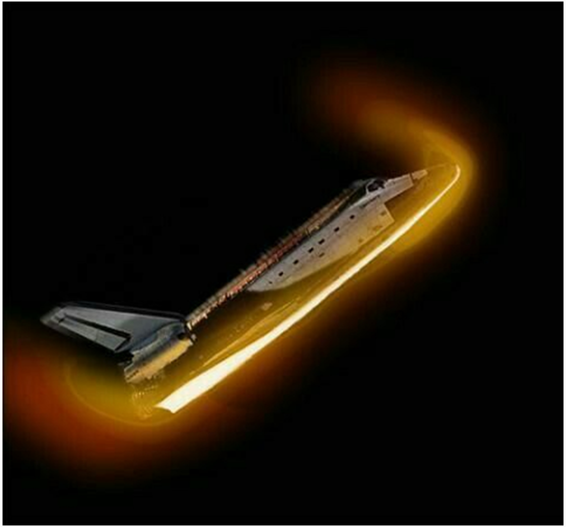
Goal was to reduce the velocity and come to an elliptical orbit with the perigee at some height above the vicinity of the landing point.



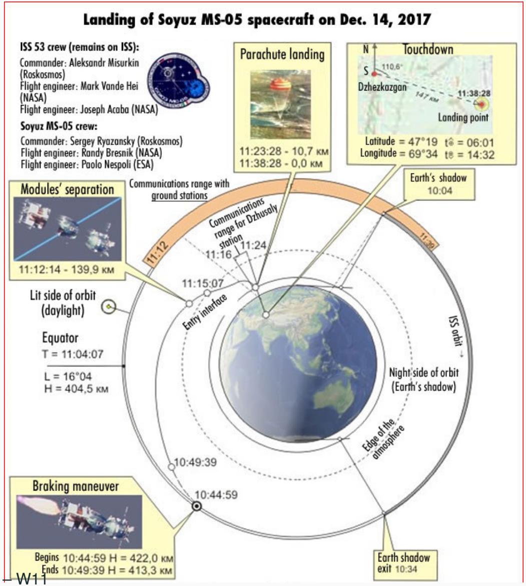
400,000 ft = 122 km

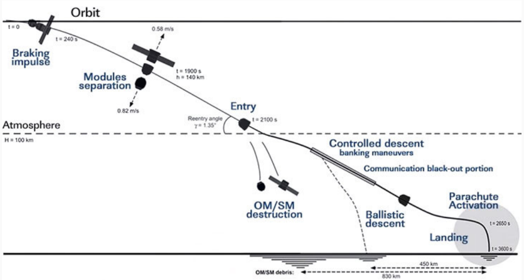
During the early part of re-entry, the orbiter's angle of attack α was maintained at 40° in order to cause enough braking high in the atmosphere, and reduce the drag and deceleration when reaching the lower layers of the atmosphere.

Re-entry of the space shuttle



The Shuttle's angle of attack α was precisely maintained at 40 degrees (+/- 1 degree) during the early part of re-entry, normally by the digital autopilot (DAP). It was then reduced when deceleration of 1.5 g was reached, with a gradual transition to aerodynamic flying.





Credits: RussianSpaceWeb.com

Interactive quiz

$\rightarrow$ EchoPoll platform

- You can scan a QR code or go to the link
- EchoPoll is the EPFL-recommended solution
- You do not have to register, just skip entering a username and/or email address