# Systems Engineering for Space Missions


**Alejandro Cardesín Moinelo**

ESA Science Operations


(slides courtesy of Christian Erd & Peter Falkner, ESA-ESTEC)

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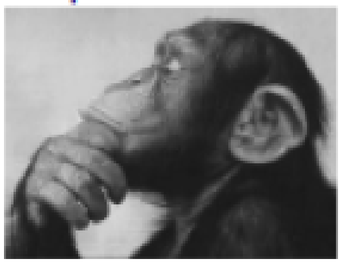
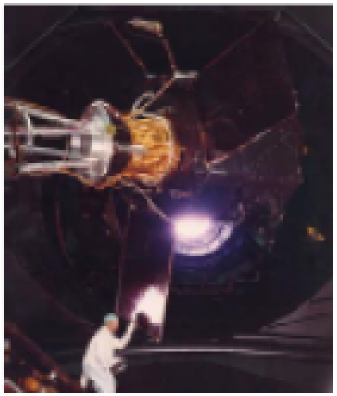
# Mission Flow Diagram

1. Conceptual design, feasibility and requirements definition



2. Design

3. Qualification (*verification that the system design fulfils the specified requirements with a margin)*



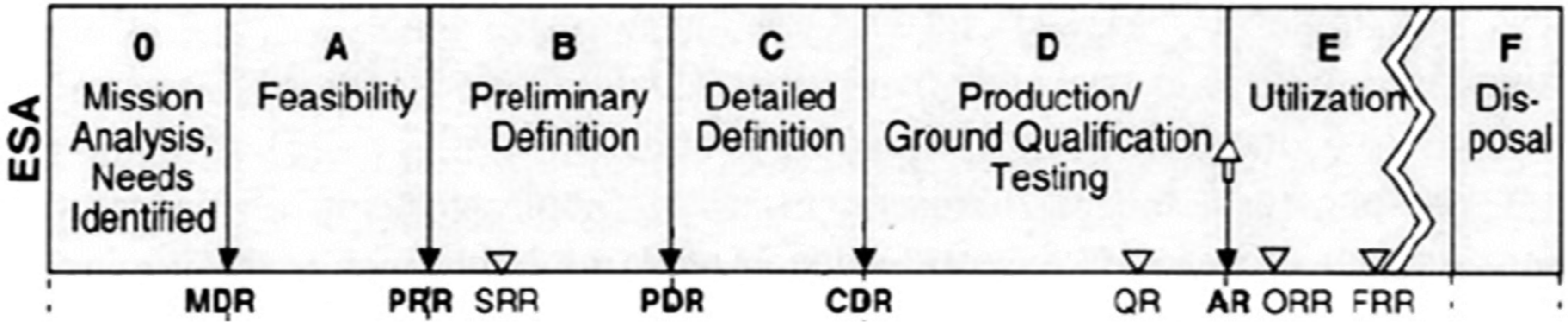
4. Production

5. Customer Acceptance (*check that the product is in agreement with the qualified design, is free from workmanship defects and acceptable for use)*

6. Utilisation/Operation

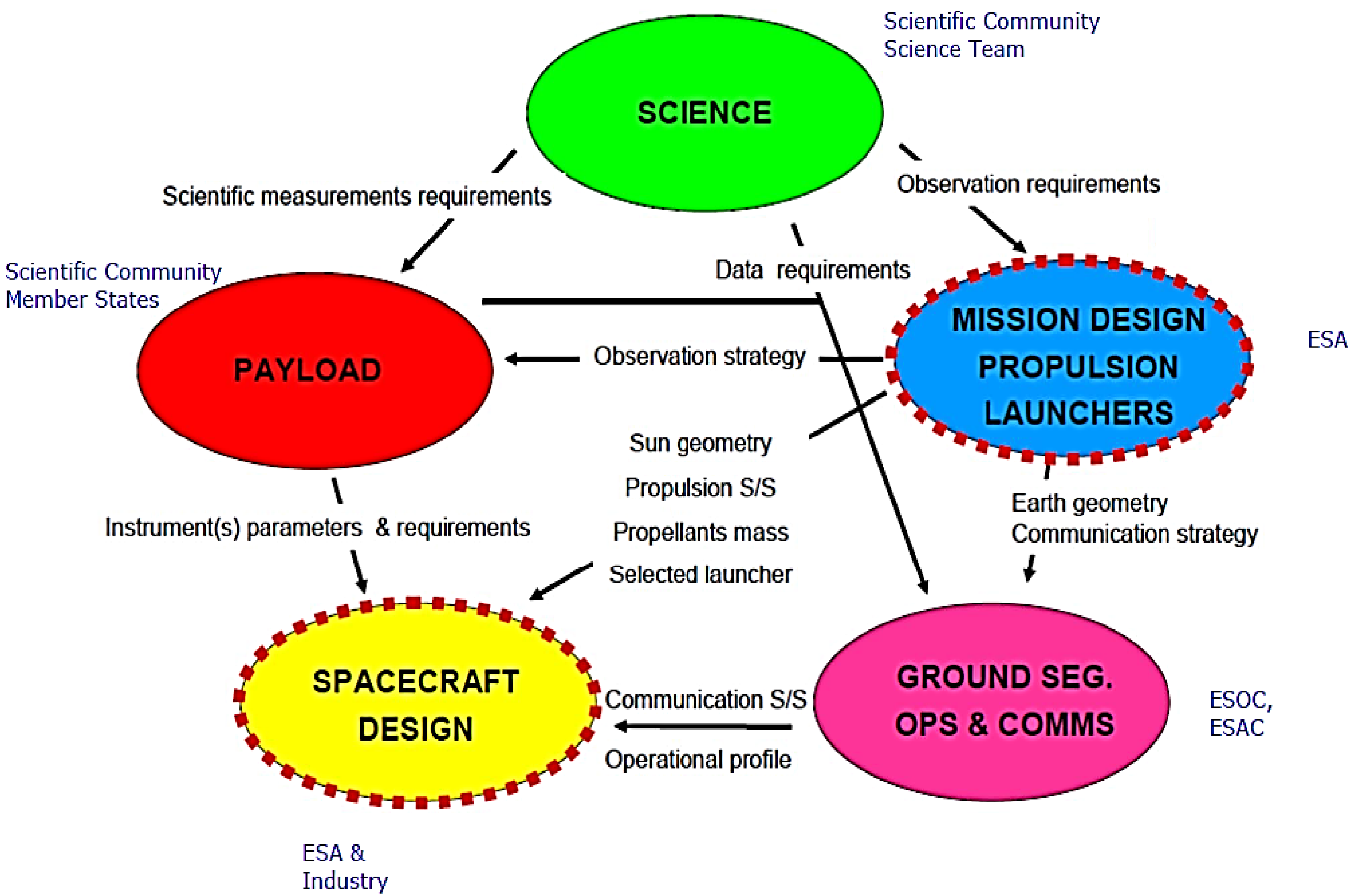
7.Disposal

# Space Mission Timeline: Phases



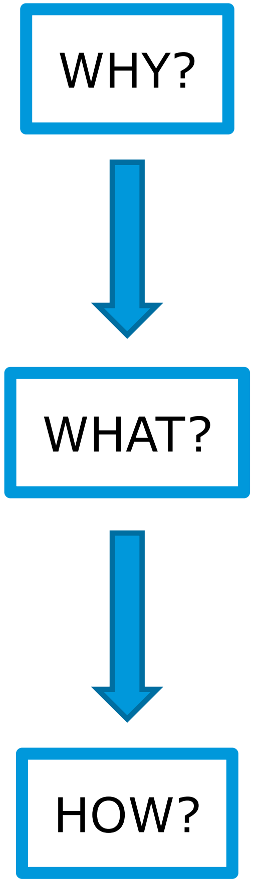

| Activity | Approximate Duration |
|---|---|
| ESA Internal Assessment Phase 0 | 1.25 yrs |
| Industrial Assessment Phase A | 2.25 yrs |
| Definition Phase B1 | 0.5 yrs |
| Preparation of Implementation Phase | 1 yr |
| Implementation Phase B2/C/D | 5 – 7 yrs |

# Science Mission Design Process



# Science Objectives – Requirements - Solutions

- **<u>Science</u> Objective** is the high level motivation
  - Which scientific question/application purpose shall the project address and what answer is sought
- **Requirement** is the translation of this objective into verifiable statements of what is needed to achieve the objective
  - With detailed quantities (unambiguous)
  - Several levels of detail
  - Traceable, all the way back to the top level
  - <u>Careful with conflicting requirements</u>
- **Solution** is the response to the all requirements
  - There can be several solutions meeting requirements
  - Non-compliance needs to be negotiated



# Conflicting Requirements

**Farm "Optimal" Animal :**

***Eierlegende Wollmilchsau***

*(famous Austrian animal)*

# Trade-off

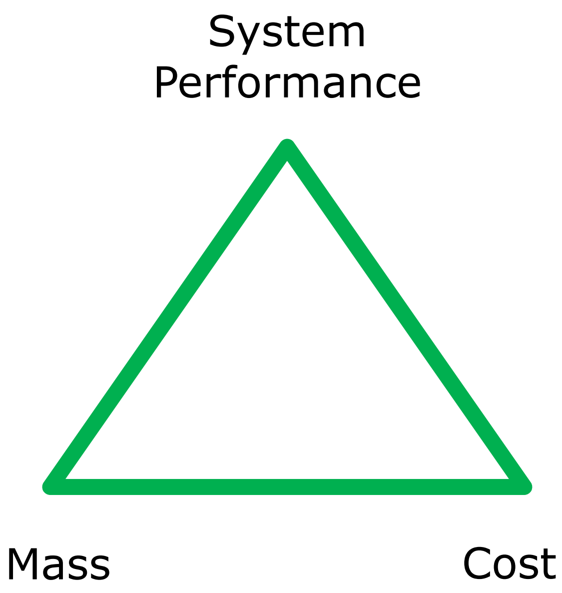
Trade-off allows exploring alternative solutions to a baseline

Most common criteria: **mass, cost budget**; several system properties can be translated into them

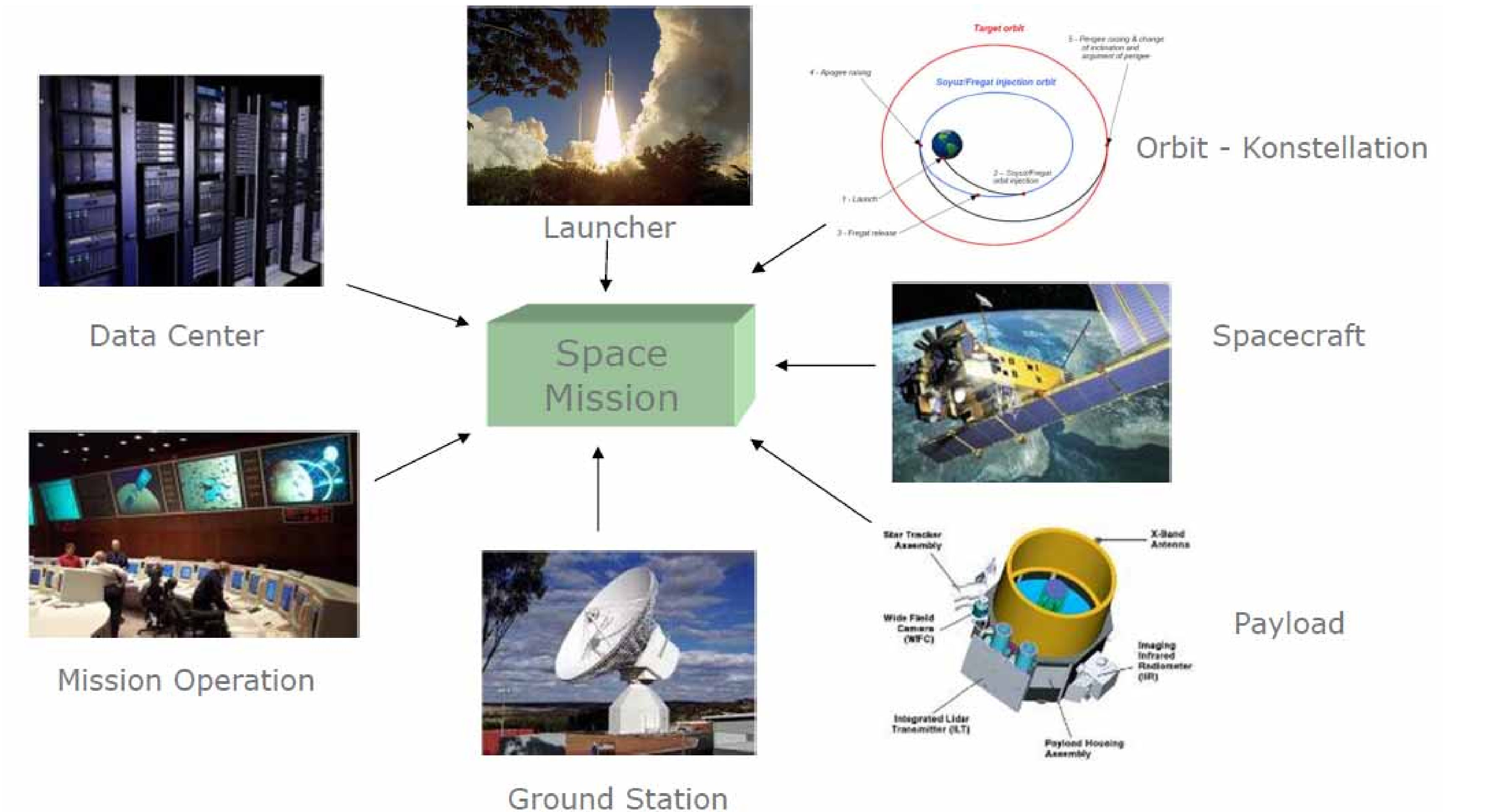
Power consumption → generation of more power → solar array size → mass

Higher telemetry volume → larger HGA, more power for TM&C → mass

High performance → complex solutions → more effort for verification → longer integration time → cost

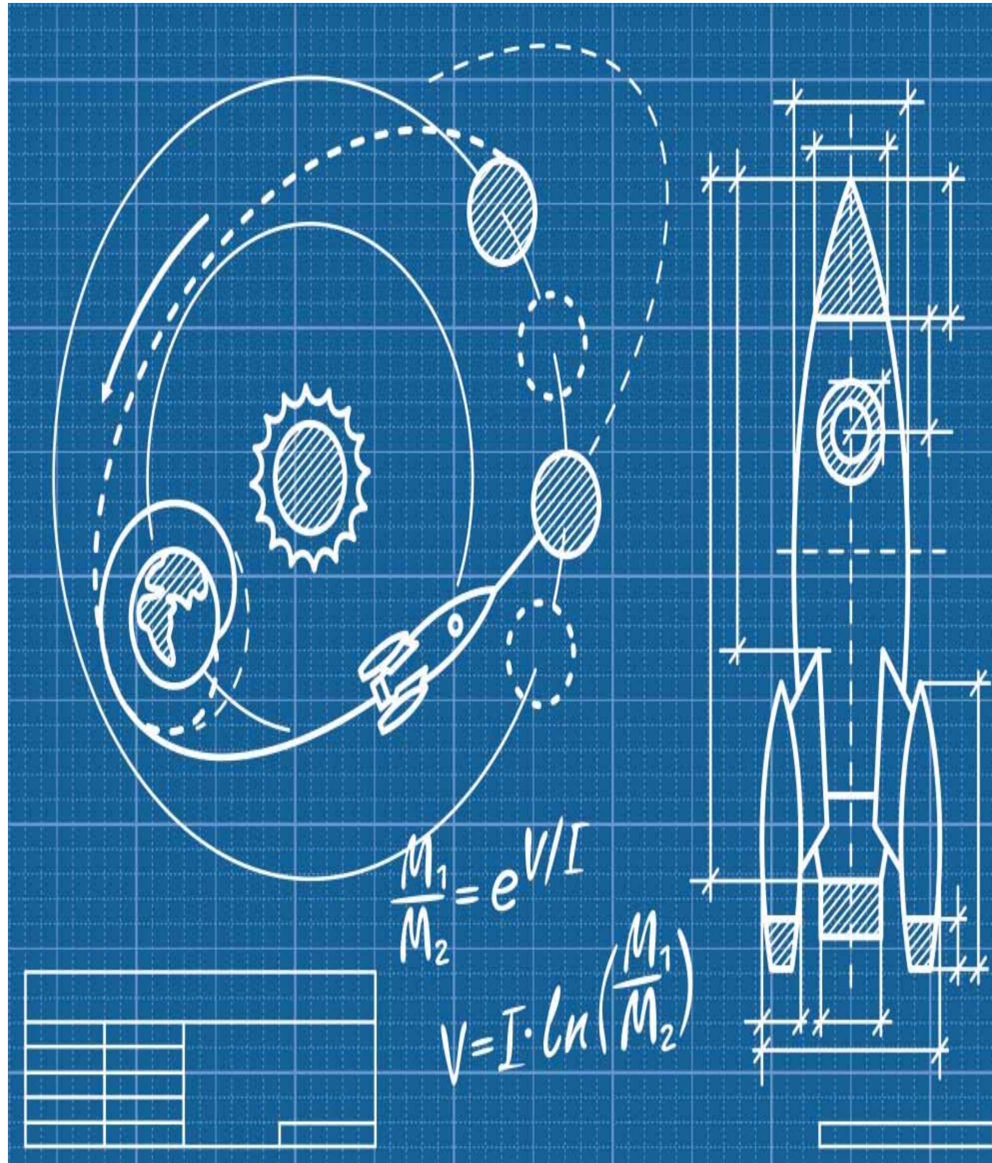


# Mission Segments, Systems & Subsystems



# Mission Analysis

- Launch
- Transfer trajectory
- Insertion into target orbit
- Orbit and Maintenance
- End-of-Life disposal



# Launchers

**Rocket launcher gives initial impulse in order to**:

Compensate gravity and atmospheric drag

Insertion in terrestrial orbit (Low/Medium/Geostationary)

Earth Escape Velocity (11km/s, 40000km/h)

Insertion into interplanetary transfer orbit

**3 types of ESA launchers:**

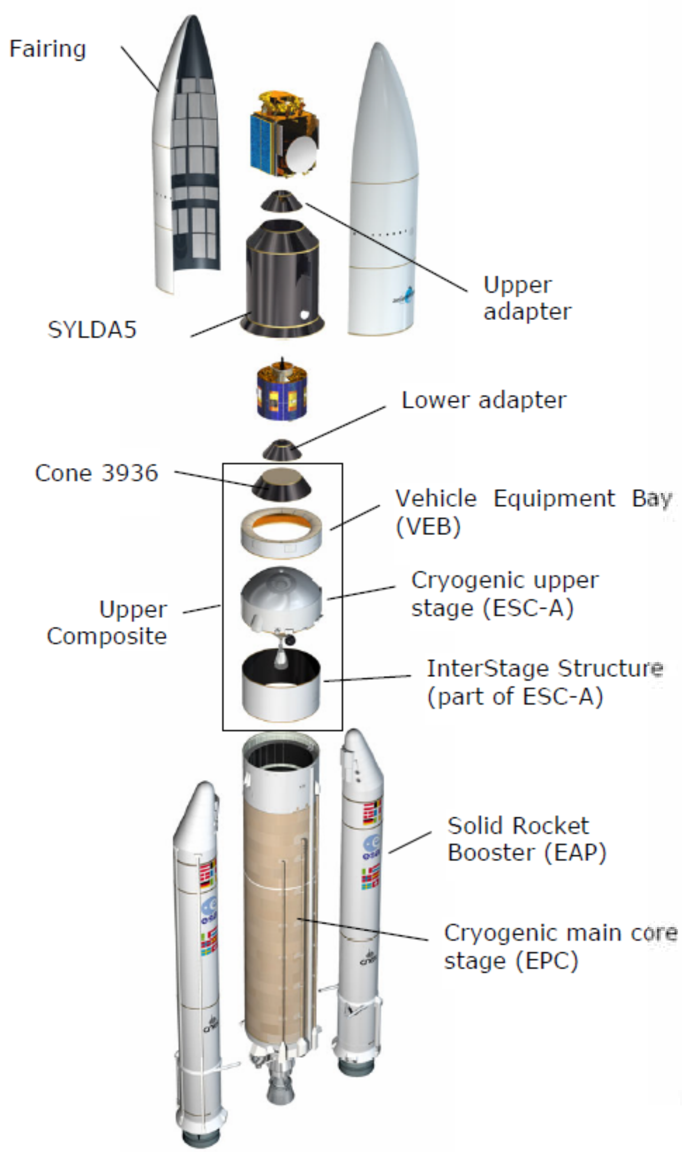
<u>Vega</u> (35M€) : 1500kg Low Earth Orbit

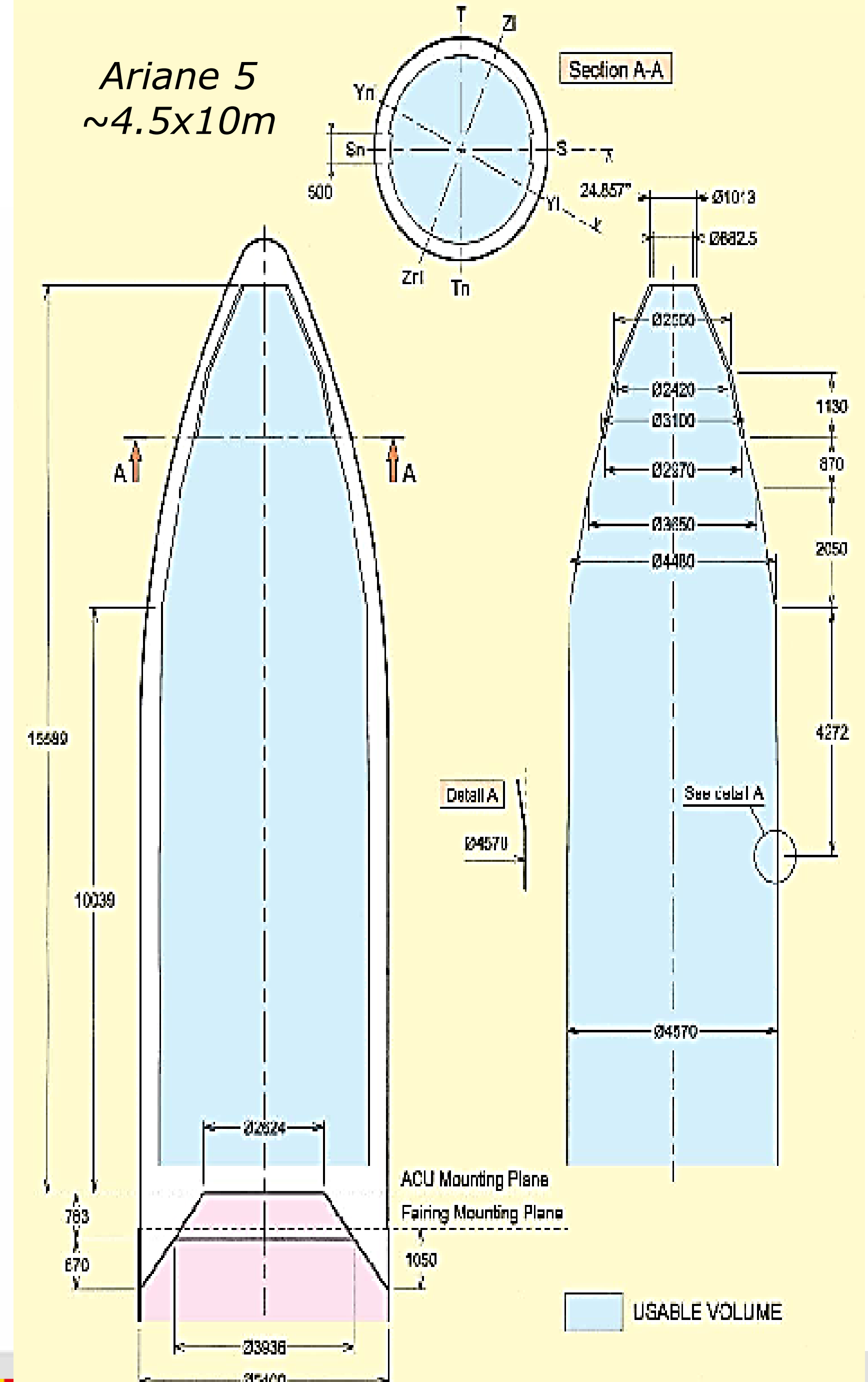
<u>Soyuz-Fregat </u>(70M€) : 3000kg GEO transfer

<u>Ariane 5</u> (150M€) : 6000-10000kg GEO transfer

*Mars Express on Soyuz-Fregat*
*Baikonur 2003*

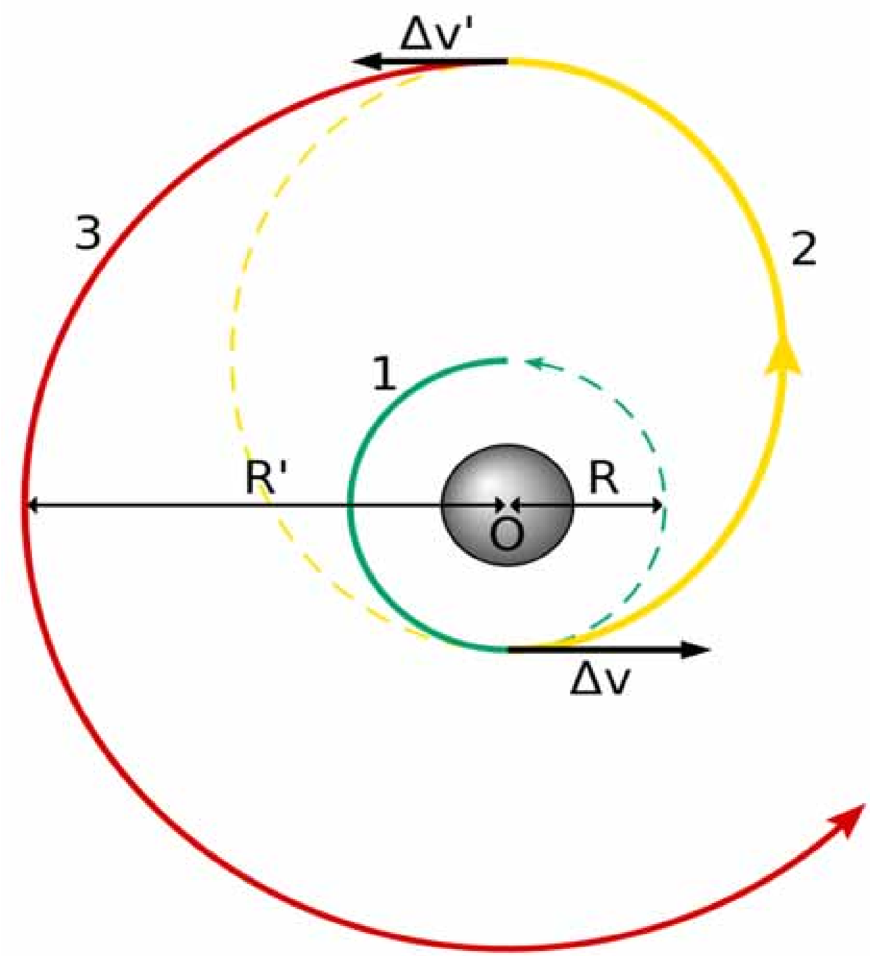
# SC size needs to fit the Launcher!



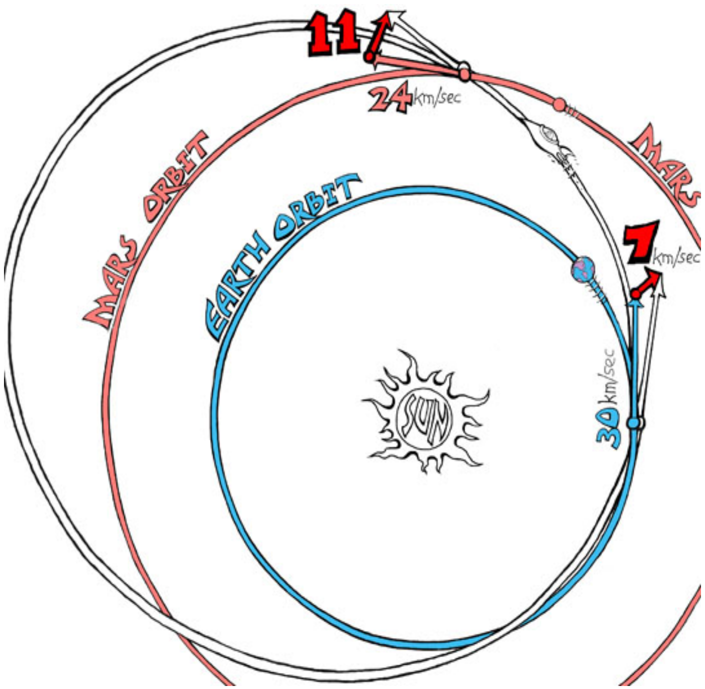



# Interplanetary Transfer Orbit:
## Simple Hohmann transfer trajectory

Cheapest transfer ellipse between two circular co-planar orbits:
minimum acceleration: least fuel




Other trajectories may be faster, but more expensive!



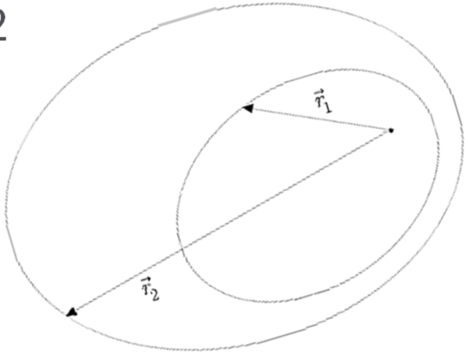
# Lambert Problem - Cost function

Lambert Problem:

In the real world the orbits of the planets are neither coplanar nor circular

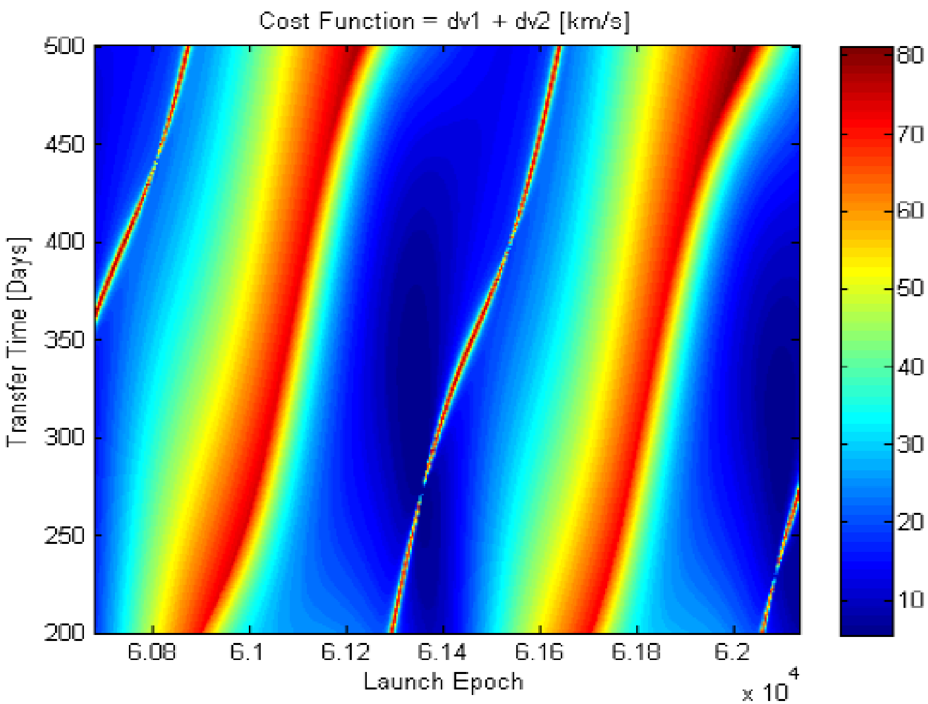
We are looking for the ellipse or hyperbola which connects r1 to r2

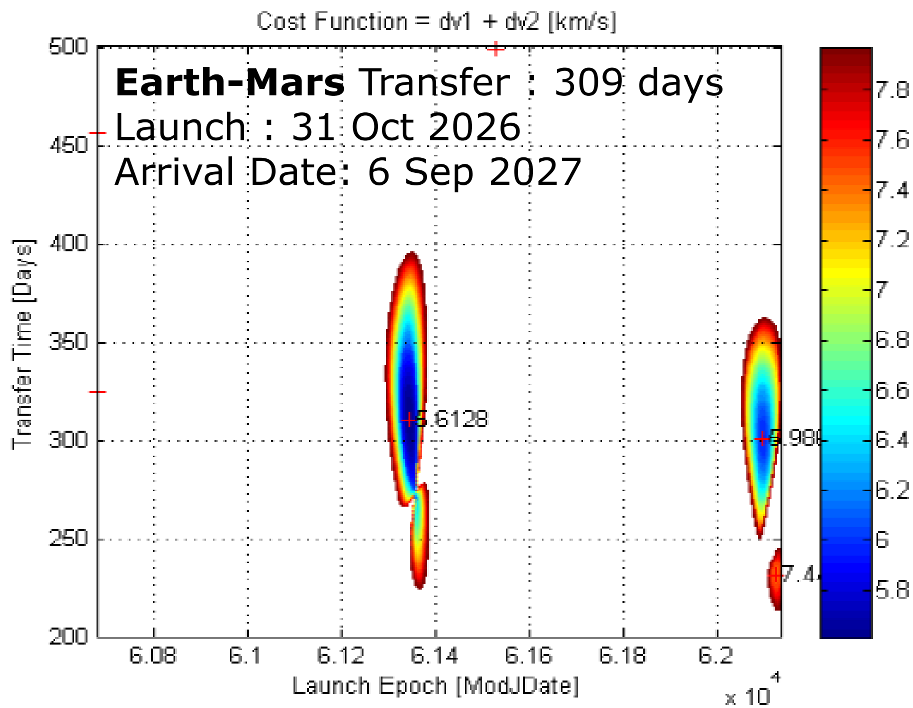
If we specify the time-of-flight($t$ -$t$ = D$t$), only one solution exists

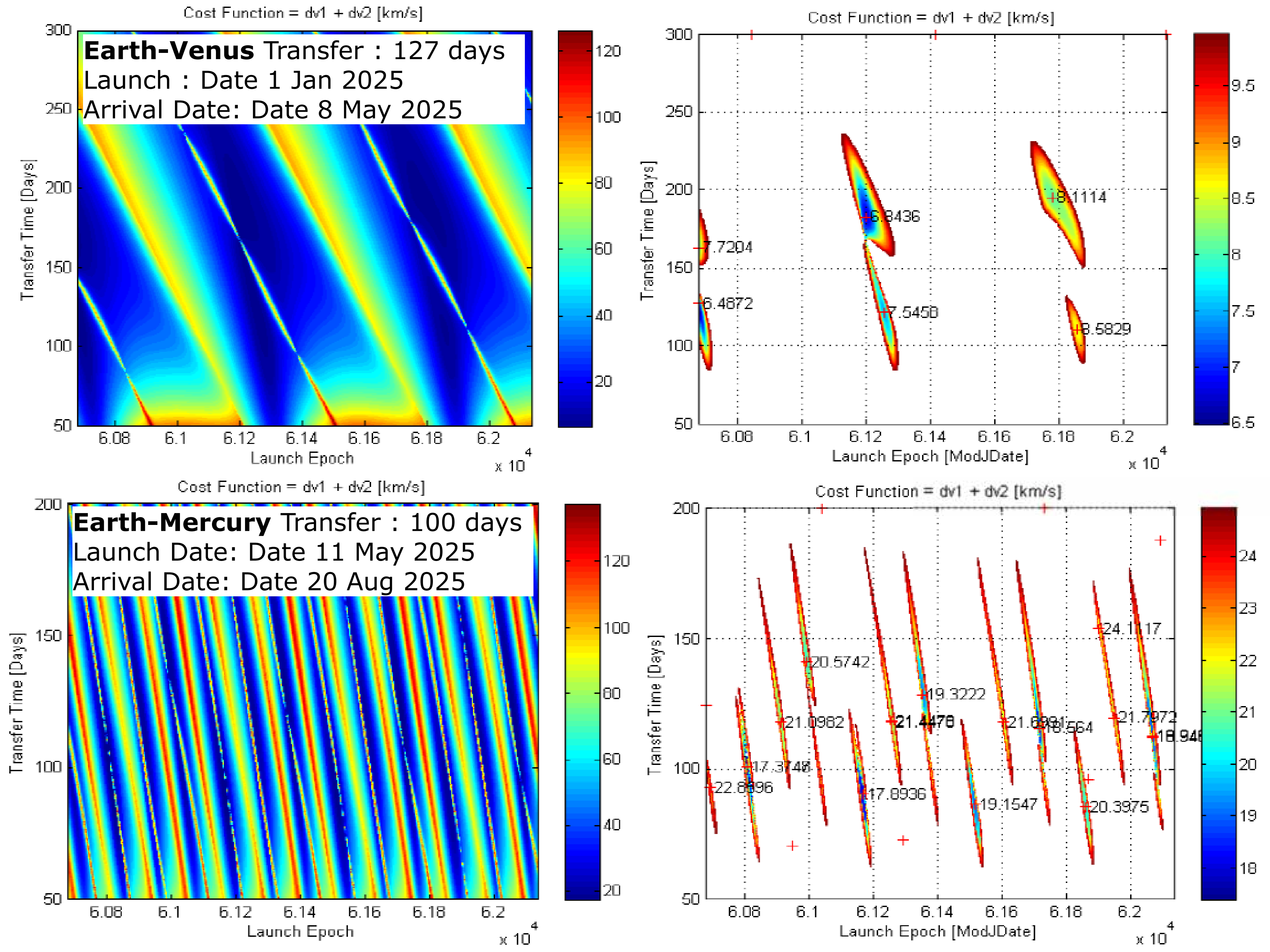



Cost function:

Minimize fuel consumption for departure and arrival maneuvers (dv1+dv2)

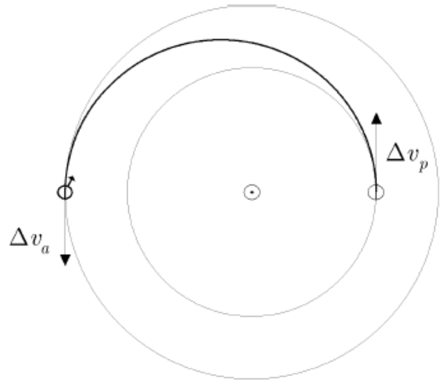


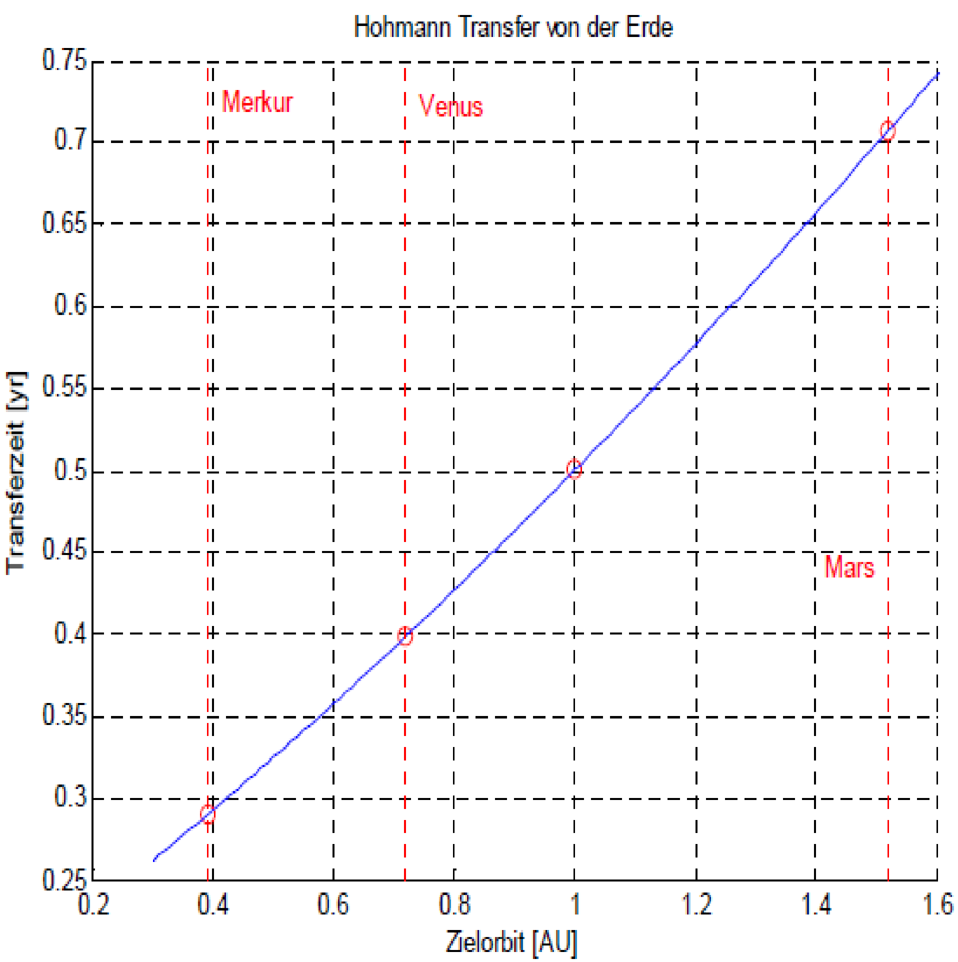



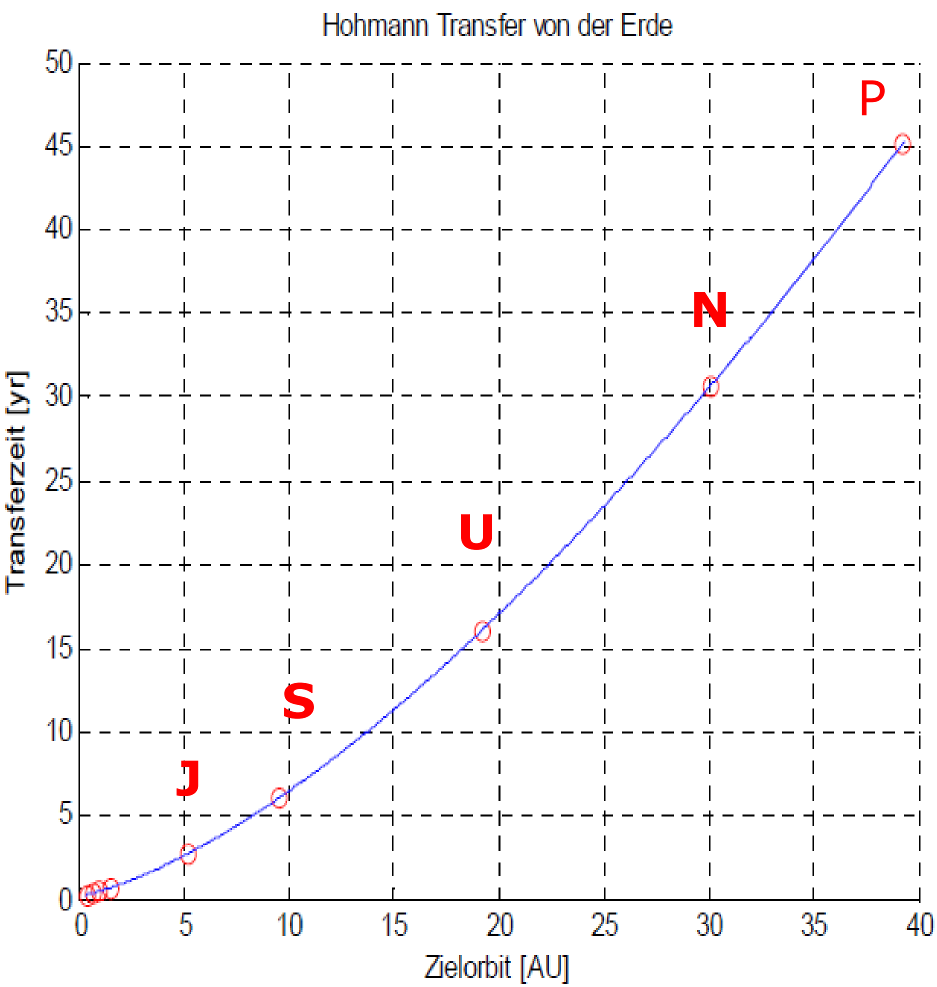


Cost Function = dv1 + dv2 [km/s]
Earth-Venus Transfer : 127 days
Launch : Date 1 Jan 2025
Arrival Date: Date 8 May 2025
Transfer Time [Days]
Launch Epoch
x 10^4
Cost Function = dv1 + dv2 [km/s]
Transfer Time [Days]
Launch Epoch [ModJDate]
x 10^4
7.7204
6.4872
6.8436
7.5458
8.1114
8.5829
Cost Function = dv1 + dv2 [km/s]
Earth-Mercury Transfer : 100 days
Launch Date: Date 11 May 2025
Arrival Date: Date 20 Aug 2025
Transfer Time [Days]
Launch Epoch
x 10^4
Cost Function = dv1 + dv2 [km/s]
Transfer Time [Days]
Launch Epoch [ModJDate]
x 10^4
20.5742
21.0982
19.3222
24.1117
22.8896
17.8936
19.1547
20.3975

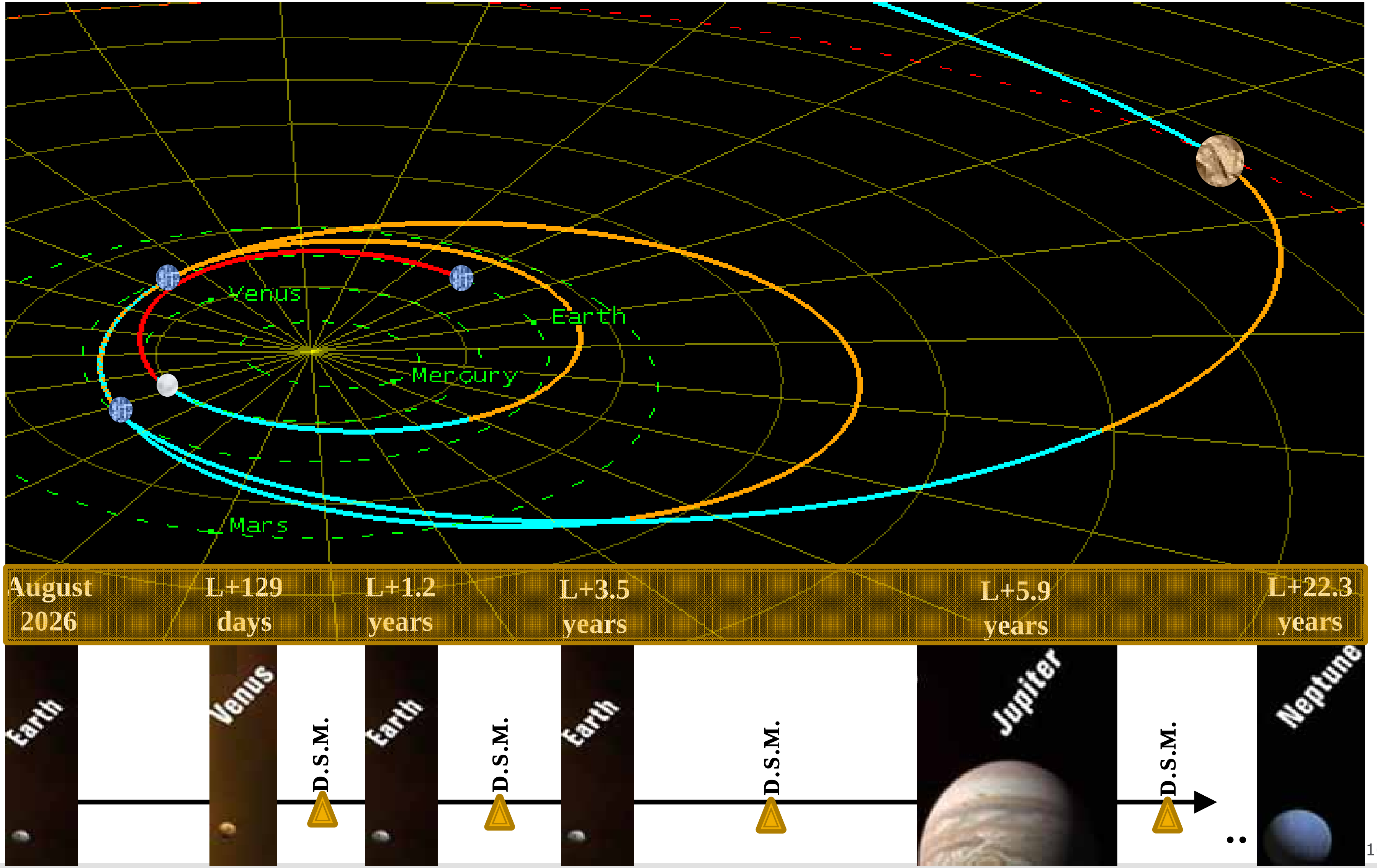
# Hohhman Transfer time durations from Earth



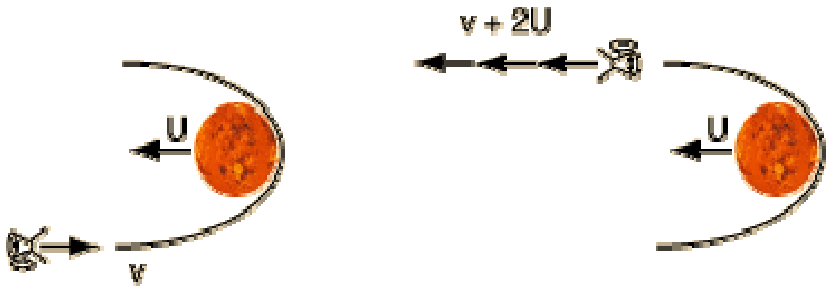



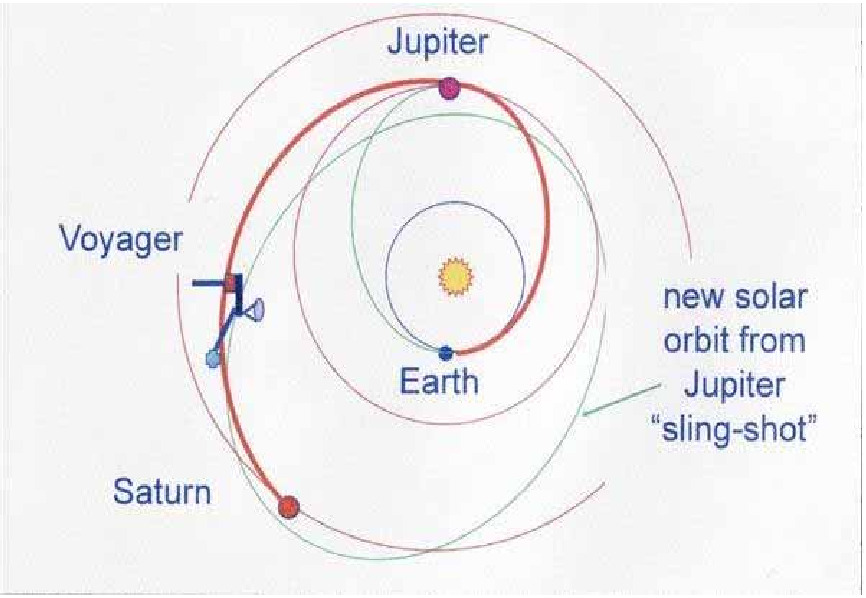



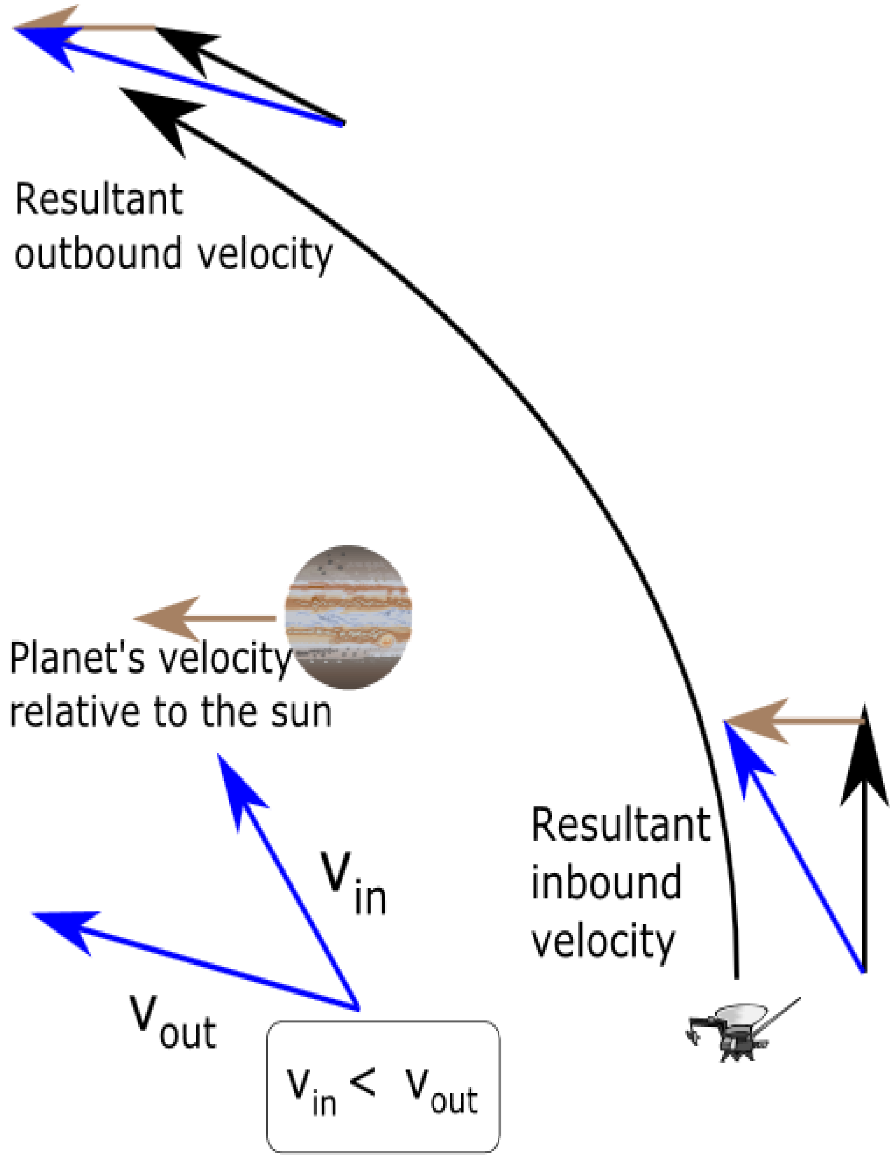
# Interplanetary Transfer Orbit:
## Other trajectories can be much more complex…



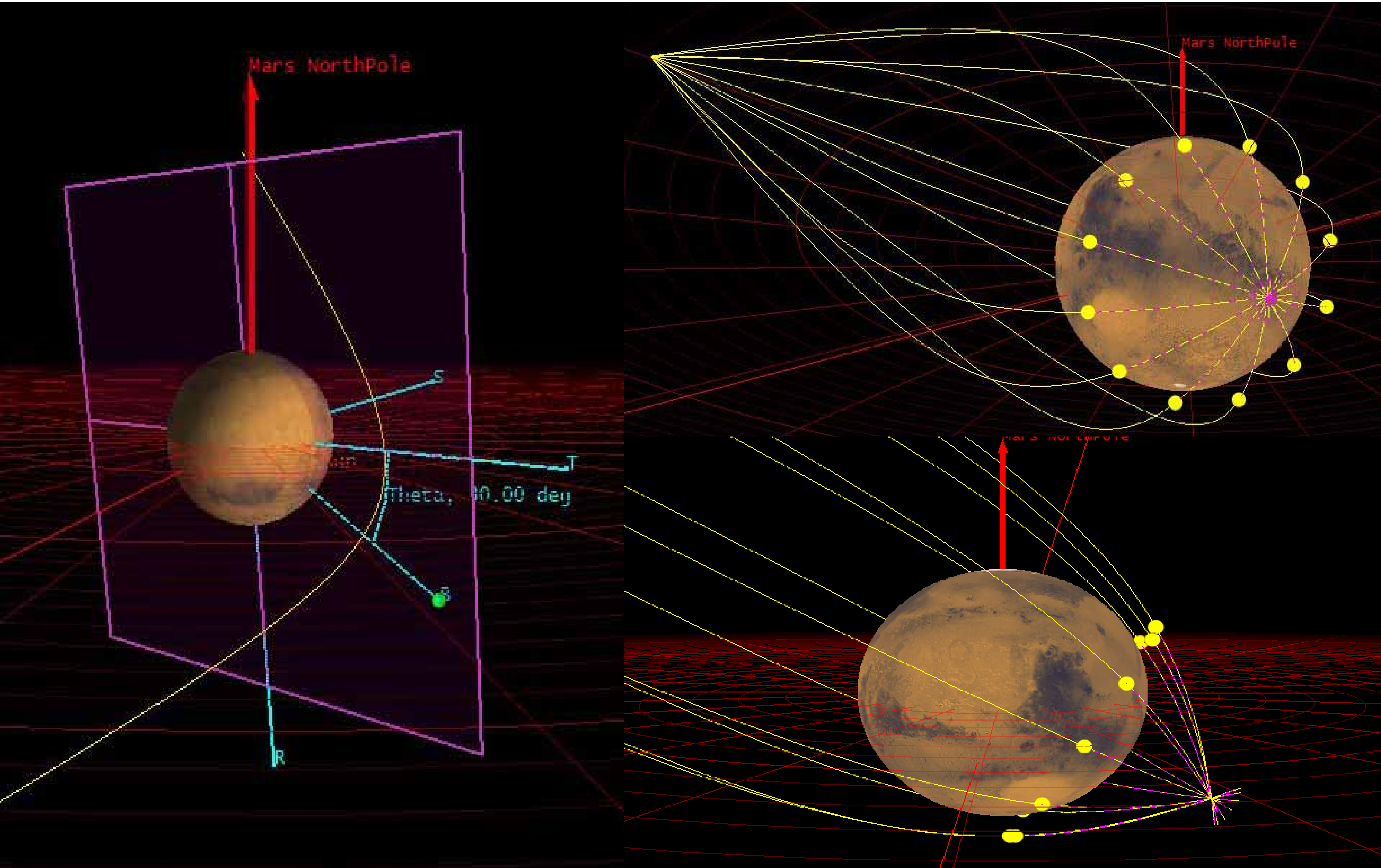
# Gravity Assist concept



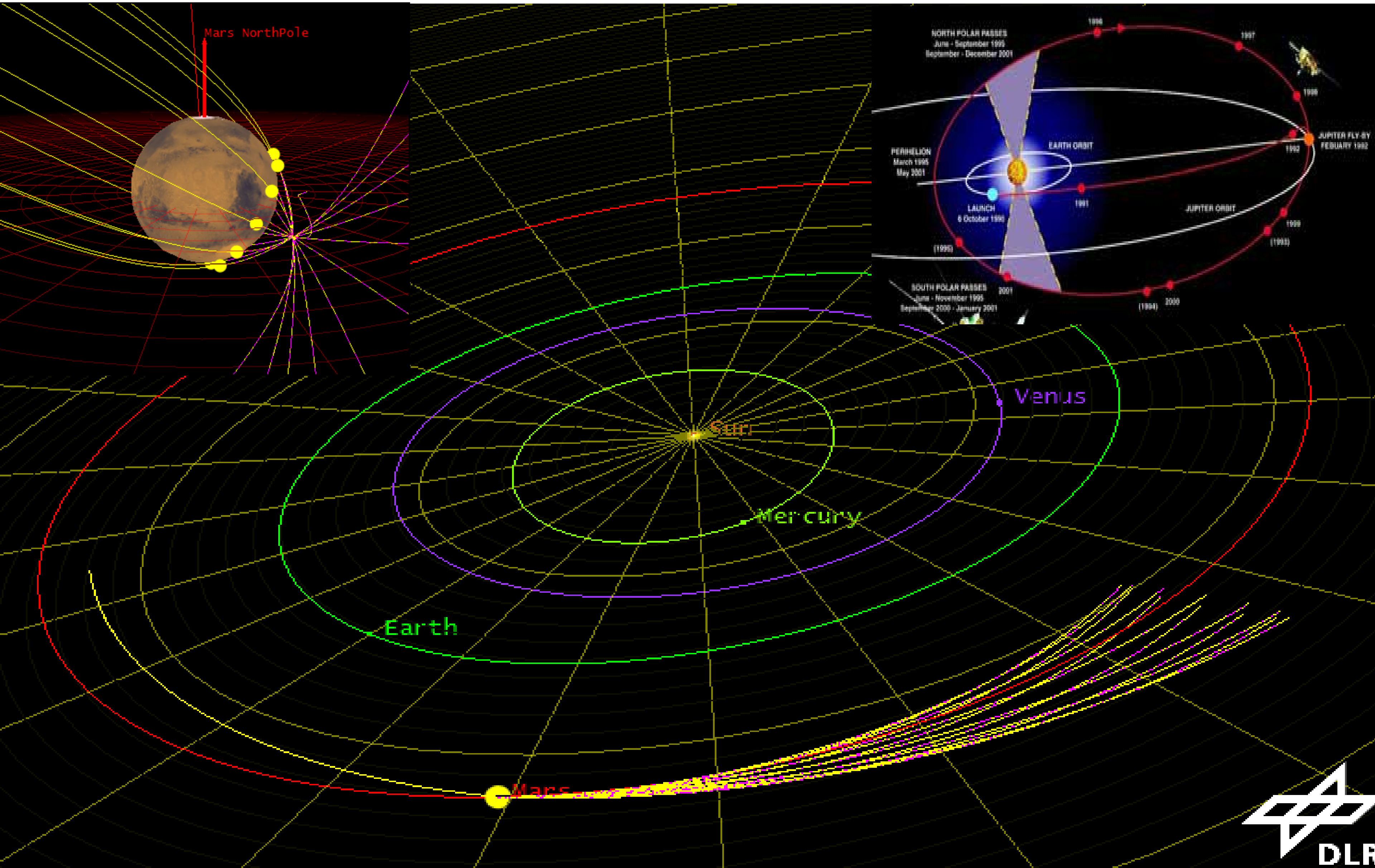



# B-Plane

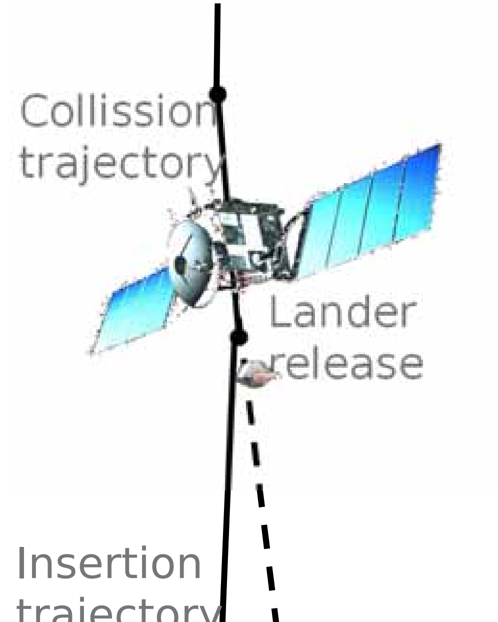


# Off-plane swing-bys

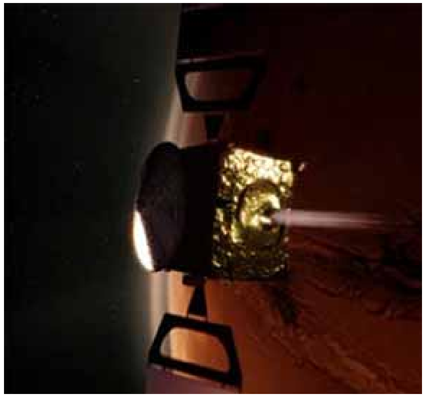
Ulysses Polar orbit

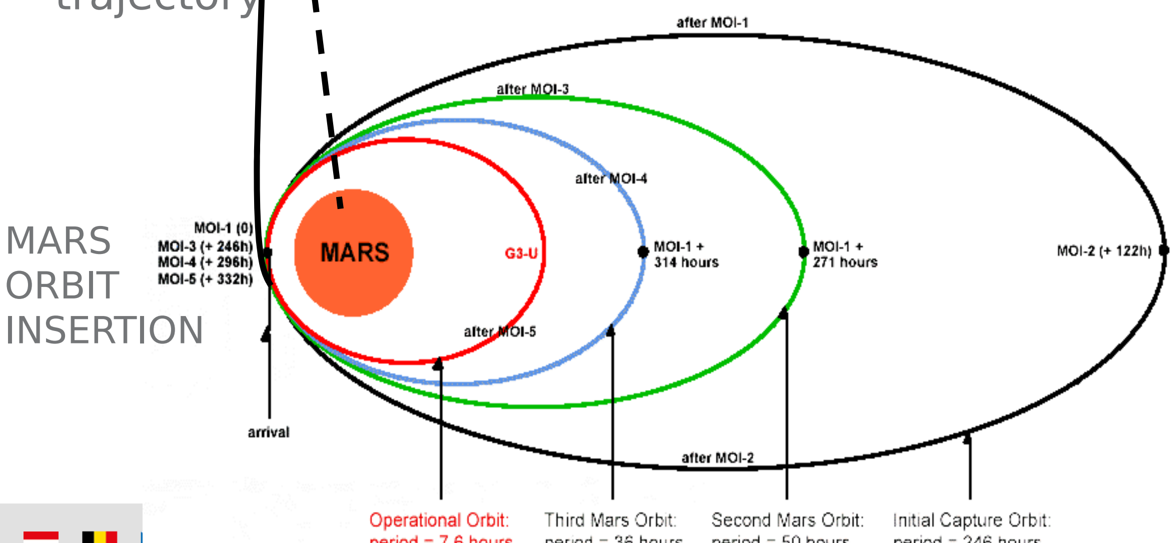


# Orbit Insertion

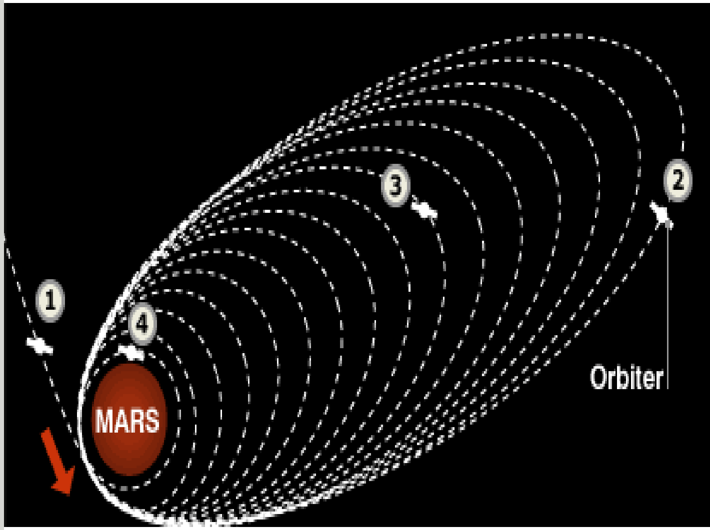



1300kg fuel (TGO)



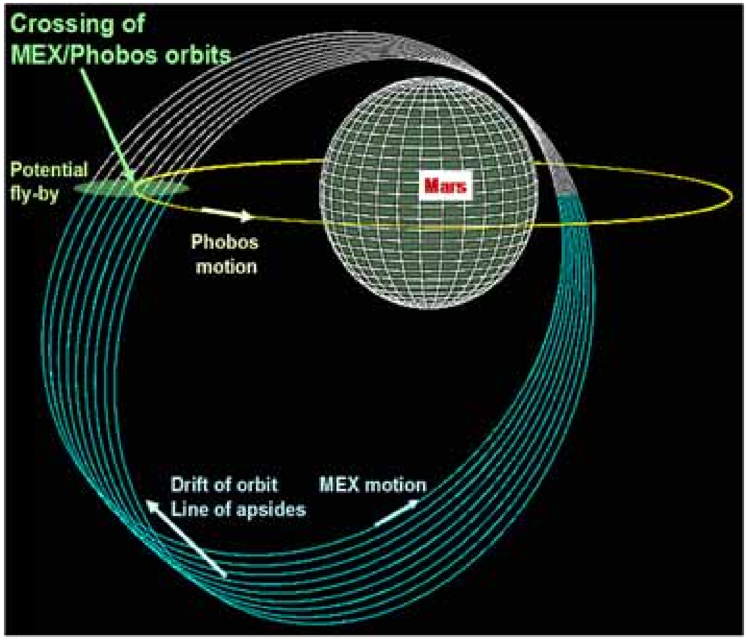


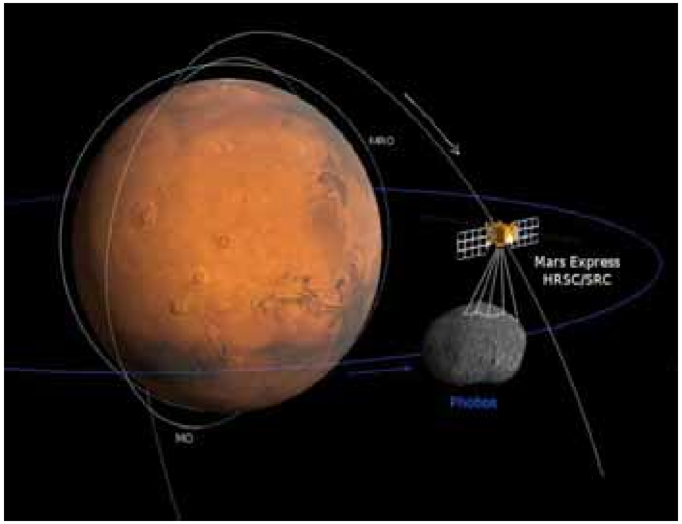

## "Aerobraking" for Orbit Circularizatión

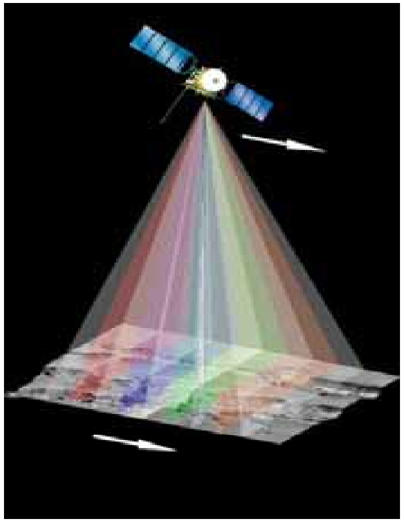


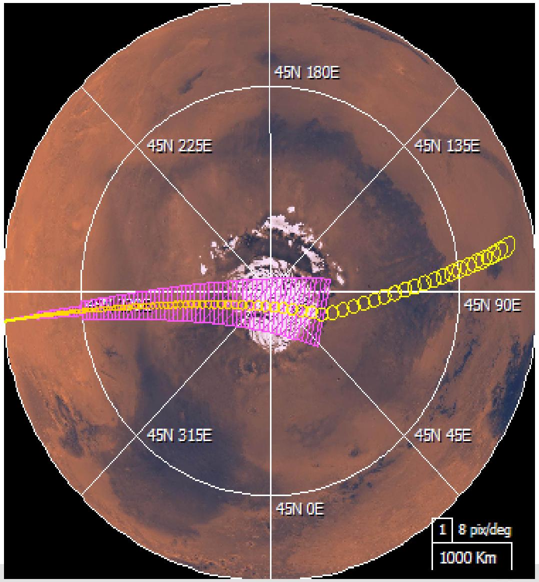

# Target orbit selection

- Driven by (contradicting) requirements:
- Resolution, revisit time, link budgets, ground station visibility, eclipse duration
- Cost of orbit acquisition and maintenance (e.g. drag, J-term perturbations, 3rd body perturbations etc…)
- Illumination conditions

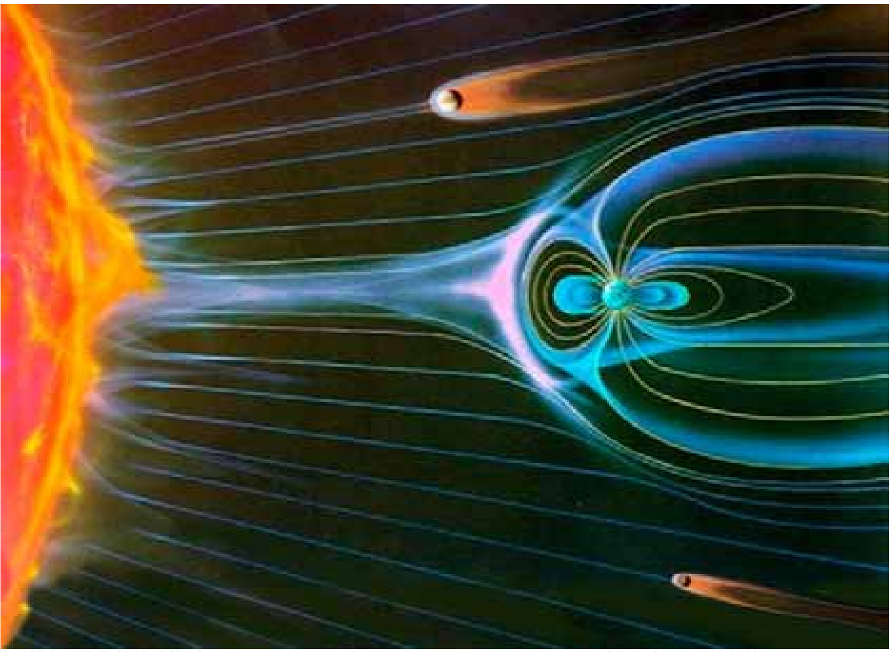




# Space Environment



**Radiation effects electronics, materials and increase noise in detectors**

Solar wind & flares: protons: 1 MeV to > 1 GeV

Cosmic Rays (protons, heavy nuclei)

Spacecraft charging (electric currents)

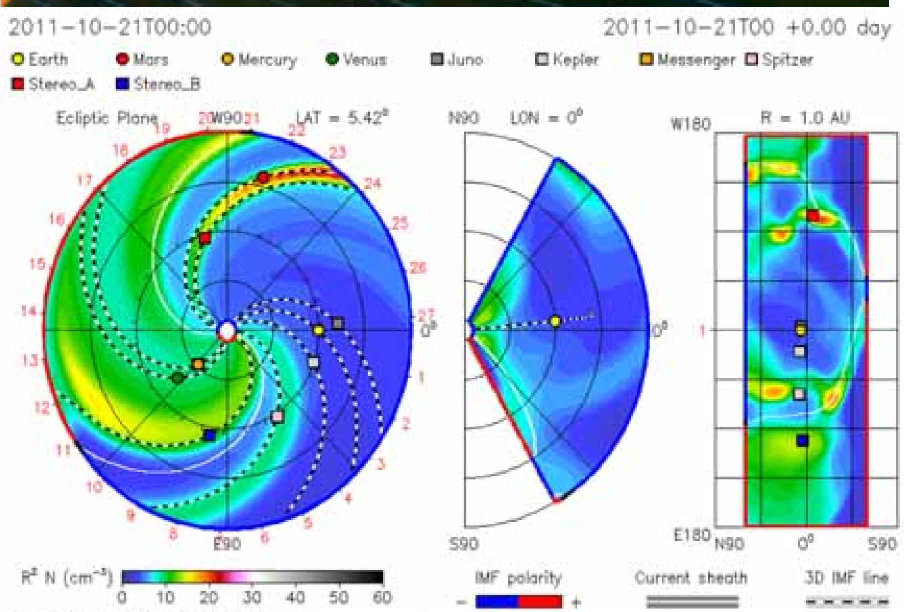
Magnetic Field

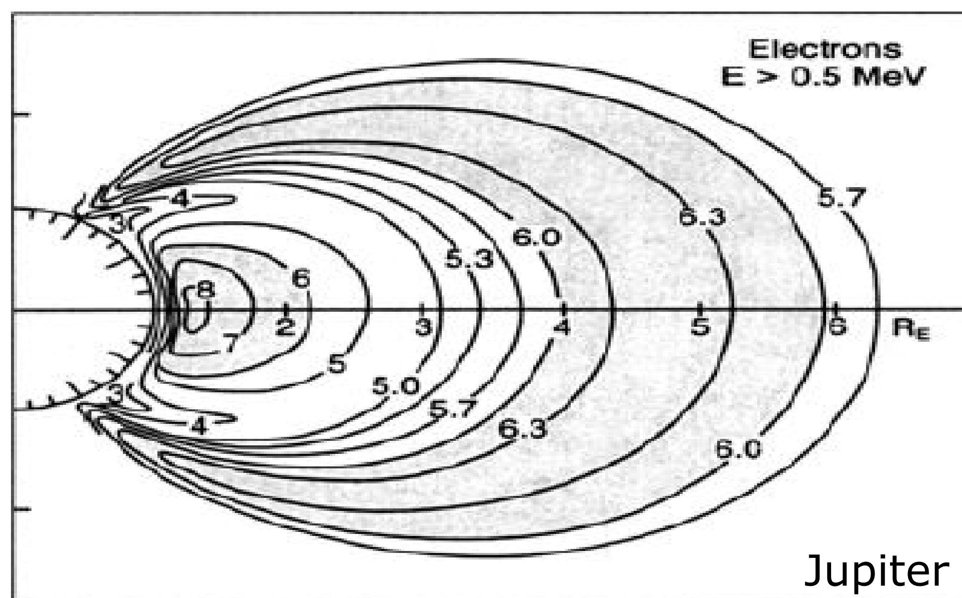
Solar Radiation Pressure

Thermal environment

Vacuum: Atomic Oxygen

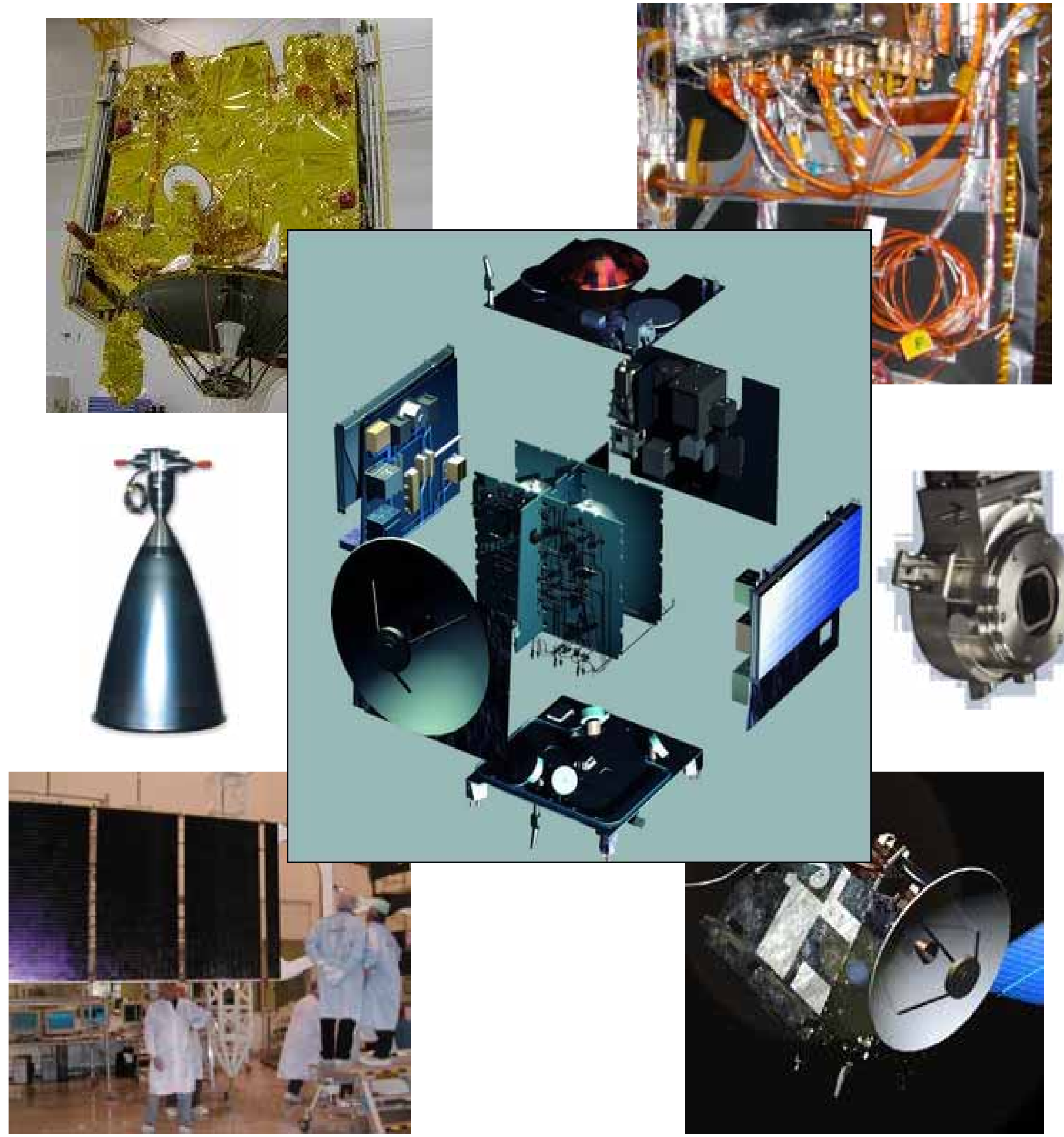
Radiation belts of Earth, Jupiter,… electrons, protons



# Spacecraft Sub-systems

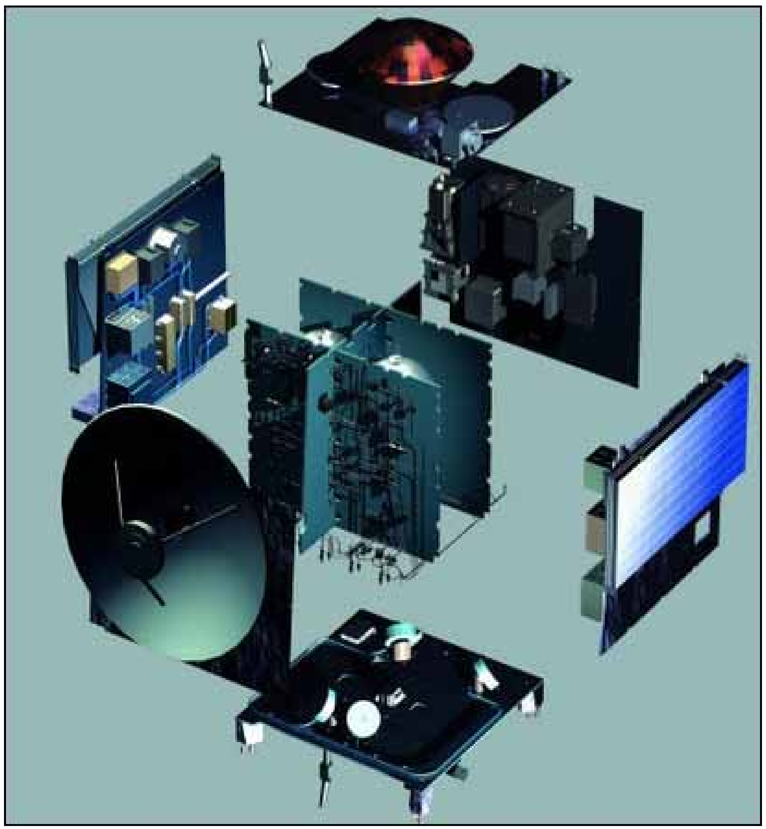
- Structure
- Propulsion
- Orientation
- Power
- On-board Computer
- Communications
- Thermal Control
- Payload



# Structures

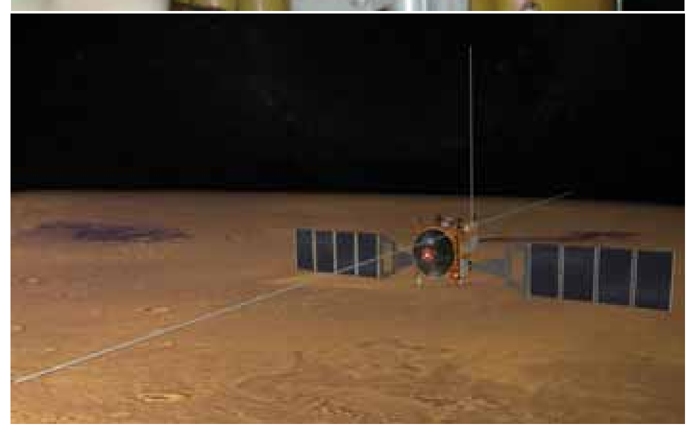
## Primary structure

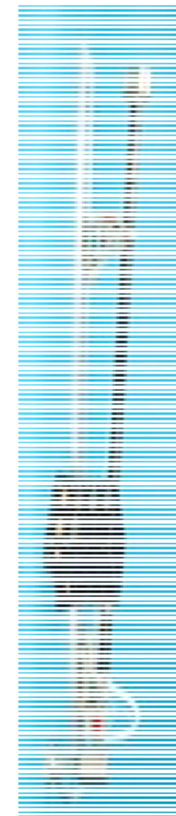
(platform harness)



## Secondary Structure

(equipment + mechanisms)







**Structure can be a large fraction of the total satellite mass: ~30 %**

# Structures

**Distinguish:**

- **Primary structures** (carrying s/c major loads)
- **Secondary structures** (carrying equipment) & appendages

- Structure need to provide stiffness in all mission modes (most driving launch, main propulsion manoeuvres, separation of stages, pyros firing etc.) <u>at lowest possible mass</u>
- **Sizing parameters**: acceleration, shock, vibration, acoustic noise (large surfaces!)
- **Critical parameter:** Strength, stiffness, density, thermal characteristics (expansion, conductivity), handling (machining), cost
- Thermal deformations / co-alignment requirements
- Eigenfrequencies > launcher induced frequencies = driving stiffness

## Mechanisms

**Moving parts**:

- Reliability is critical (lubrication in space, long storage, thermal range)
- Introduce vibrations, shock

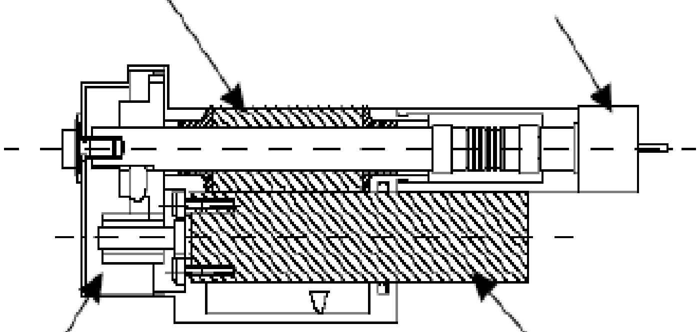
**Examples:**

- Launch lock mechanism (hold down release mechanism HDRM)
- Deployment of structure, appendages and booms
  (e.g. solar panel, sun shield, antennae, ...)
- Separation mechanism (separation of stages, multiple s/c,..)
- Pointing mechanism (e.g. payload, HG-antenna, panels)
- Reaction wheels
- Deployable instrument covers

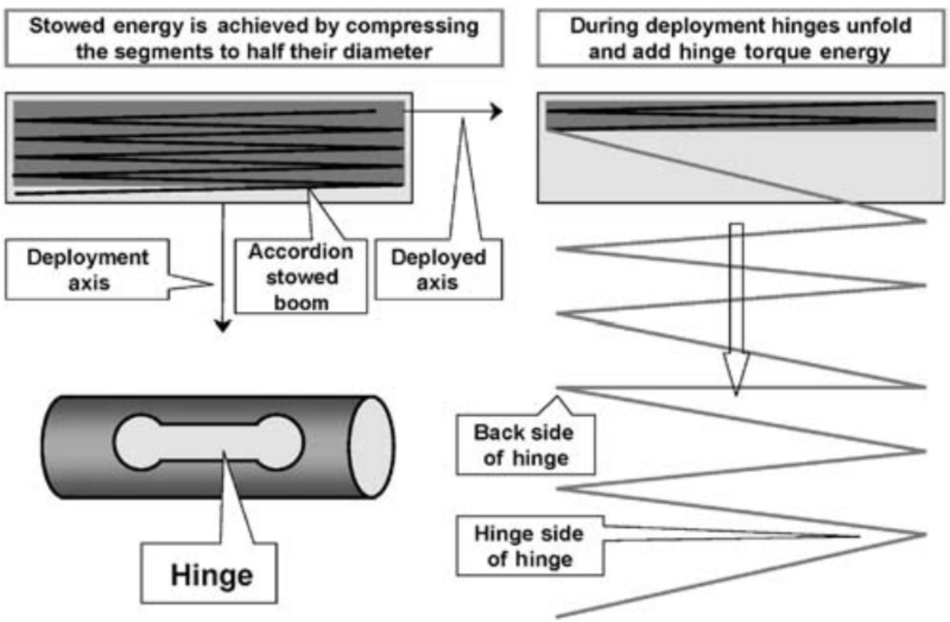
HDRM

**Mechanism are a source of mechanical noise**

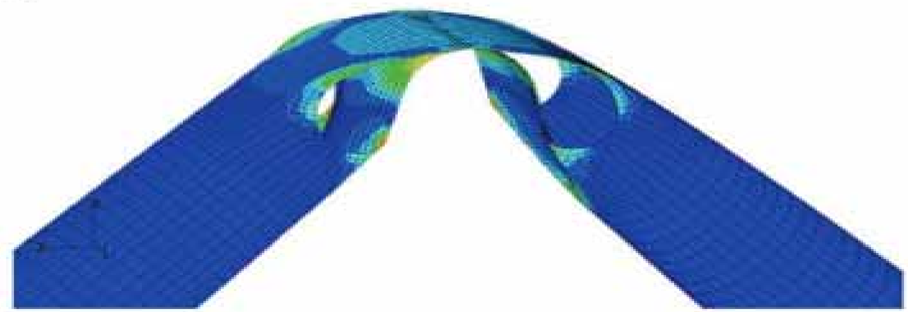
- Motors and gears: e.g. Maxon
- Frangibolt Actuator



# MARSIS Antenna deployment 2005




a)



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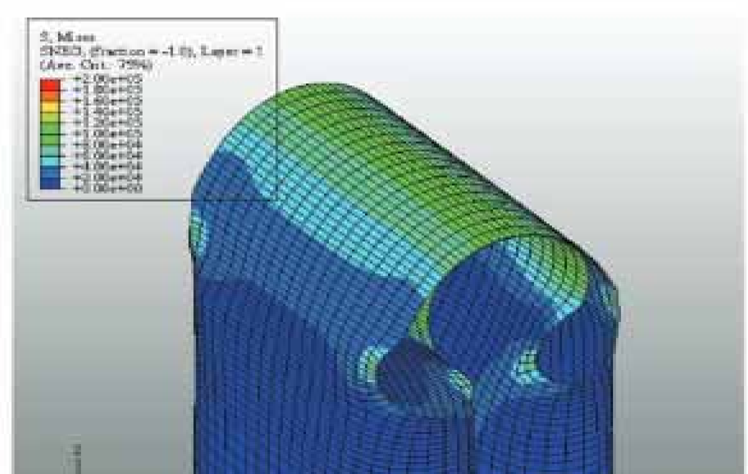

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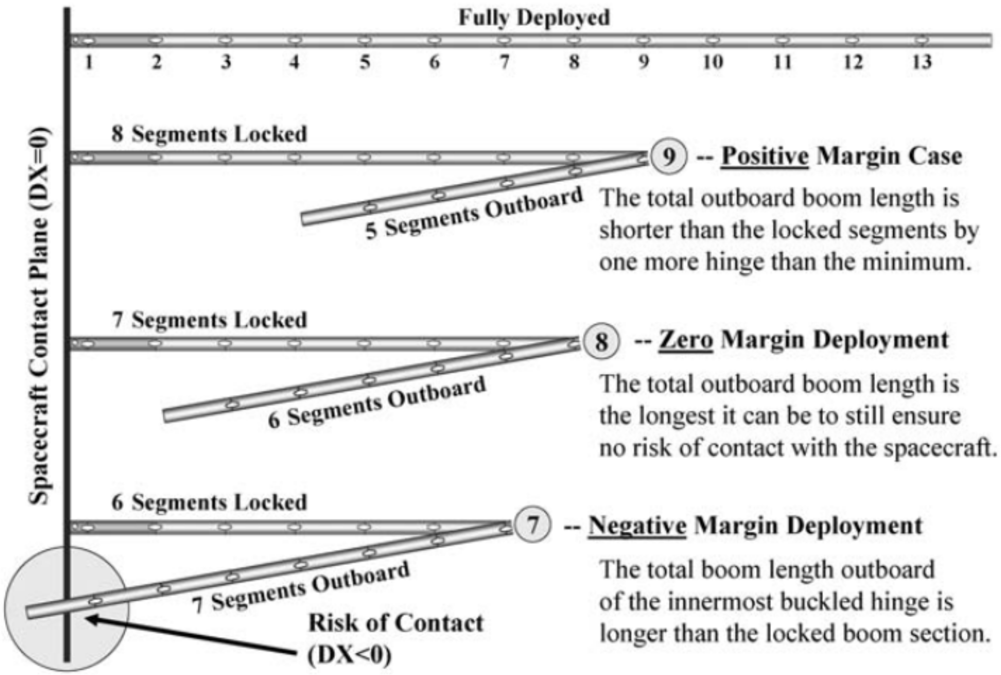
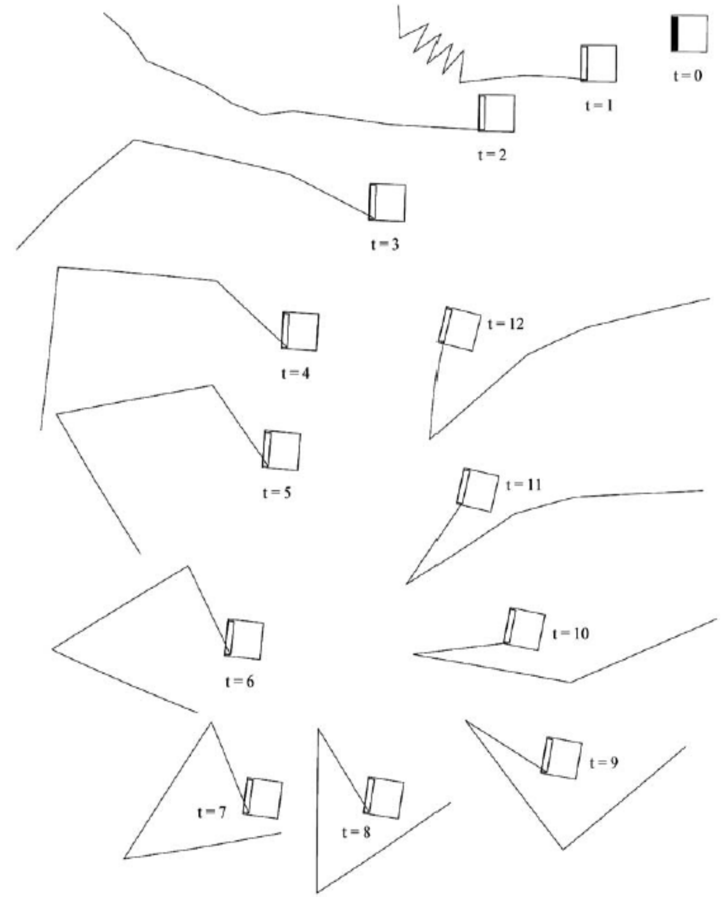
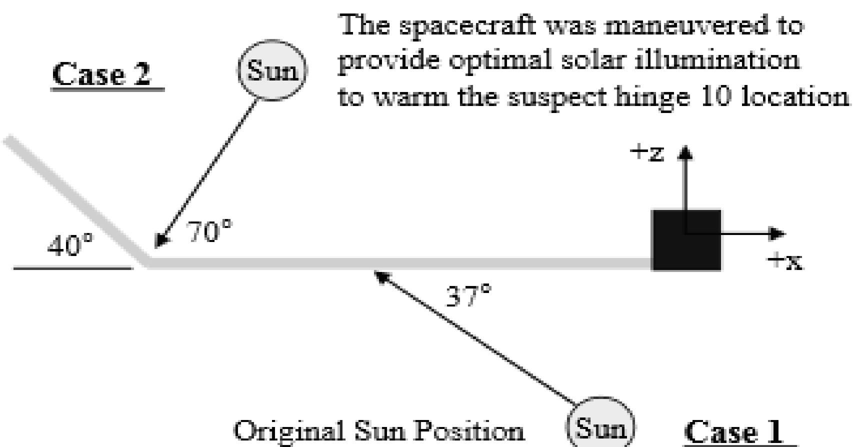
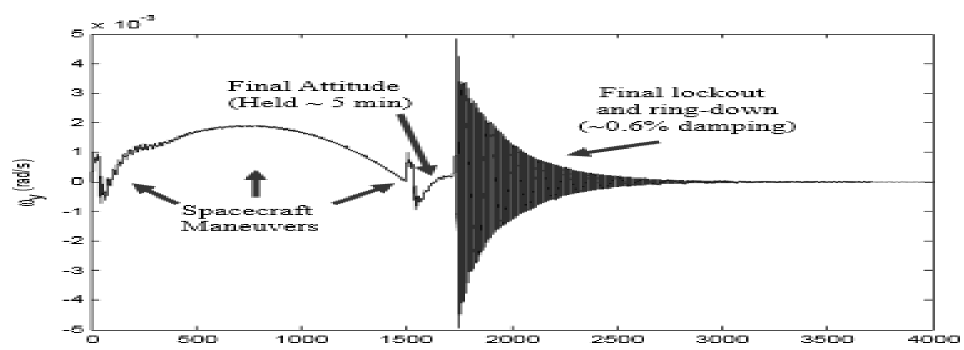


# Propulsion

**Subsystem in charge of satellite manoeuvring**

Includes thrusters, tanks, piping and valves

Many technologies available

- Solid thruster: single one off, high thrust
- Monopropellant
- Bi-propellant:
- Solar Electric

For orbital manoeuvres with high $\Delta V$: “high” $I_{sp}$ (> 300 s), e.g. bipropellant or electric propulsion

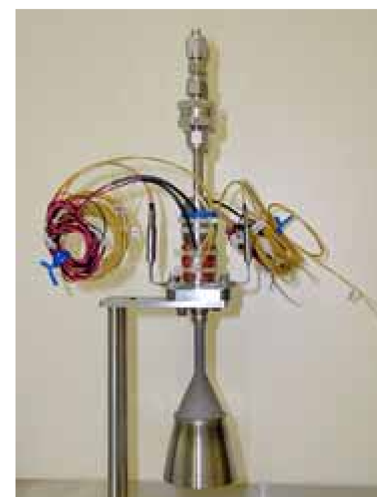
For orbital manoeuvres with low $\Delta V$: “medium” $I_{sp}$ and thrust (~1 N) – e.g. monopropellant - hydrazine

For fine control: “low” thrust: ($\leq$10 mN) – cold gas or FEEP based

Specifics for deep space missions:

- Pressurized tanks will be necessary (engine re-start)
- Valve isolation and redundancy

500 N engine

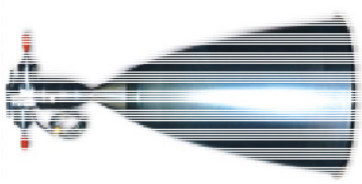


22N thruster

# Propulsion
## Example (Mars Express)

### Main Engine for Orbit Insertion

– 1 x 400 Newtons (for $\Delta v$=800m/s)

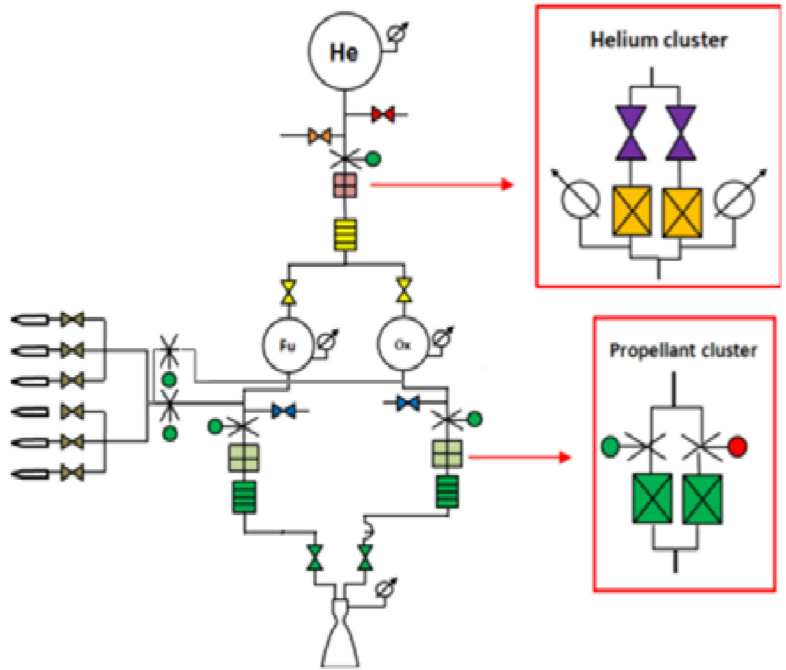


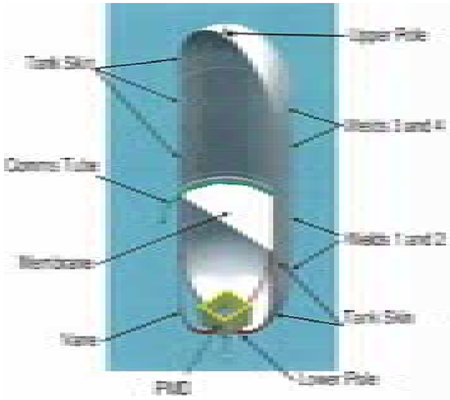
### Thrusters for Attitude Control

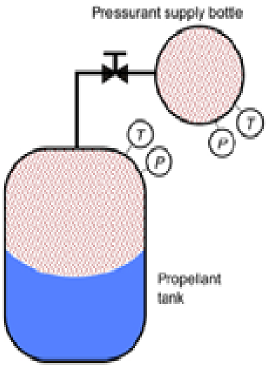
– 8 x 10 Newtons

### Bi-propellant system

– 2 tanks 270L: Oxidizer + Propellant
– 1 tank 35L : Helium for pressure
– 500 kg in total



# Future: Solar Sail

Force on solar sail

$$F = p \cdot c \cdot A \cdot cos\vartheta$$

- $p$ ~ 4.6 µN / R$^2$
- $c$ = 1 for ideal absorption
- $c$ = 2 for ideal reflection

Requirements on sail

- Large area
- Low mass – few µm!
- Deployment of large area (~100 m booms)

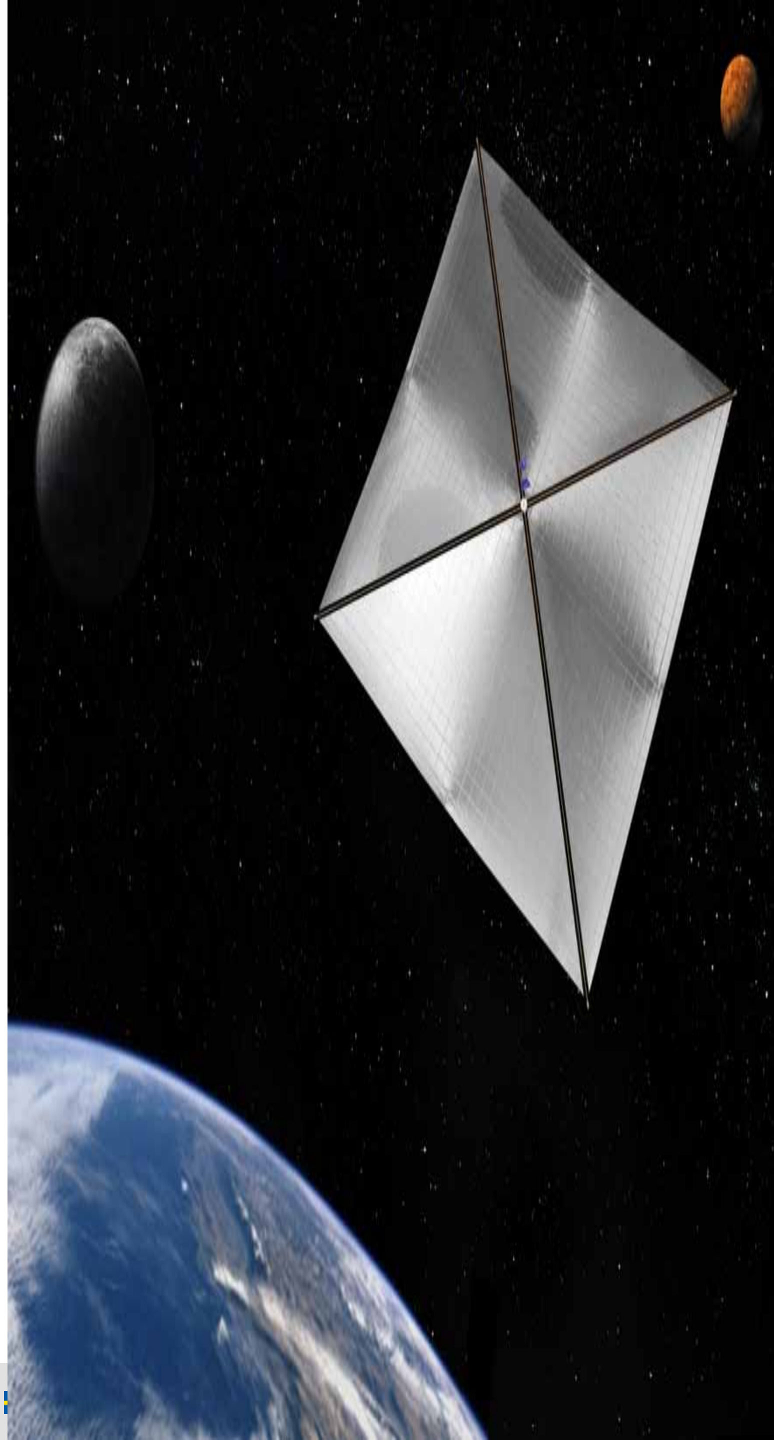
Container required on spacecraft

Large angular inertia – consider the torque needed of attitude control system

Communications: large distance

Contingency (SAFE mode) recovery strategy needed

- Communications
- Attitude control
- Solar power generation



# Propulsion Design

Propellant mass $M_{\text{prop}} = M_{\text{dry}} \times (e^{\frac{\Delta v}{g \cdot I_{\text{sp}}}} - 1)$

Needs to include:

- All deterministic manoeuvres ($\Delta V$)
- Navigations manoeuvres (stochastic)
- All AOCS needs (momentum wheel off-loading, SAFE mode, etc)

Propulsion system dry mass rule of thumb: $0.2 \times M_{\text{prop}}$

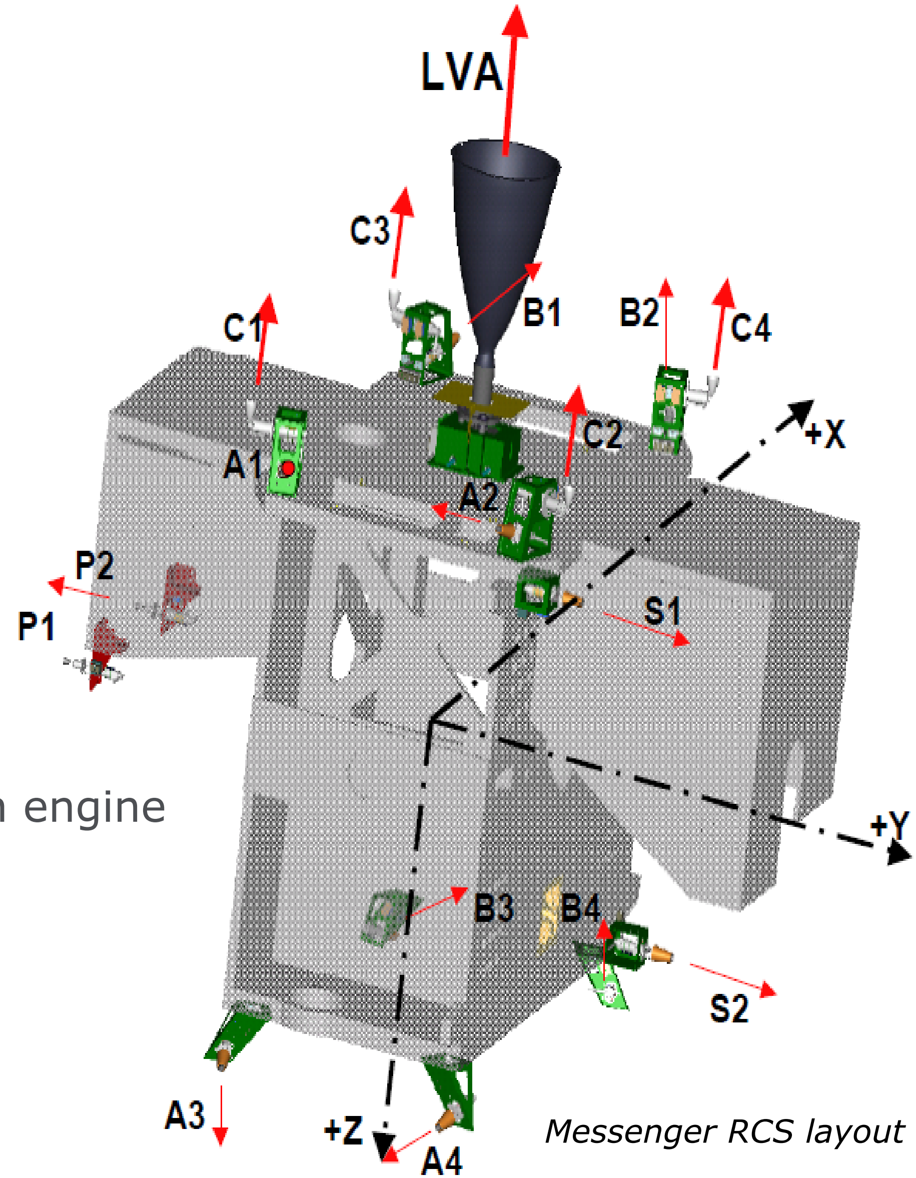
# Propulsion
## Reaction Control System Thrusters

Definition and location of thrusters

Thrusting in any direction in any attitude

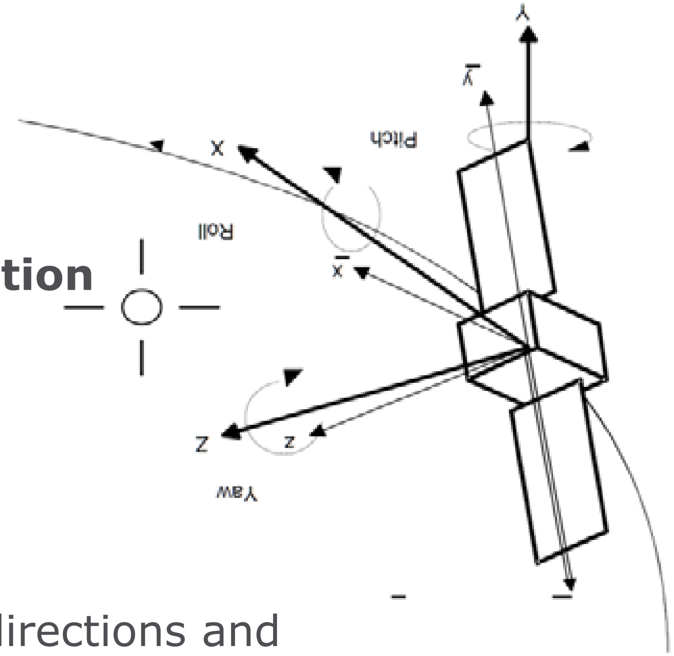
Redundancy required

RCS Thrusters could act as backup for main engine




*Messenger RCS layout*

# GNC Guidance and Navigation Control
## (AOCS Attitude and Orbit Control System)




**Allows maintaining the desired spacecraft orientation**

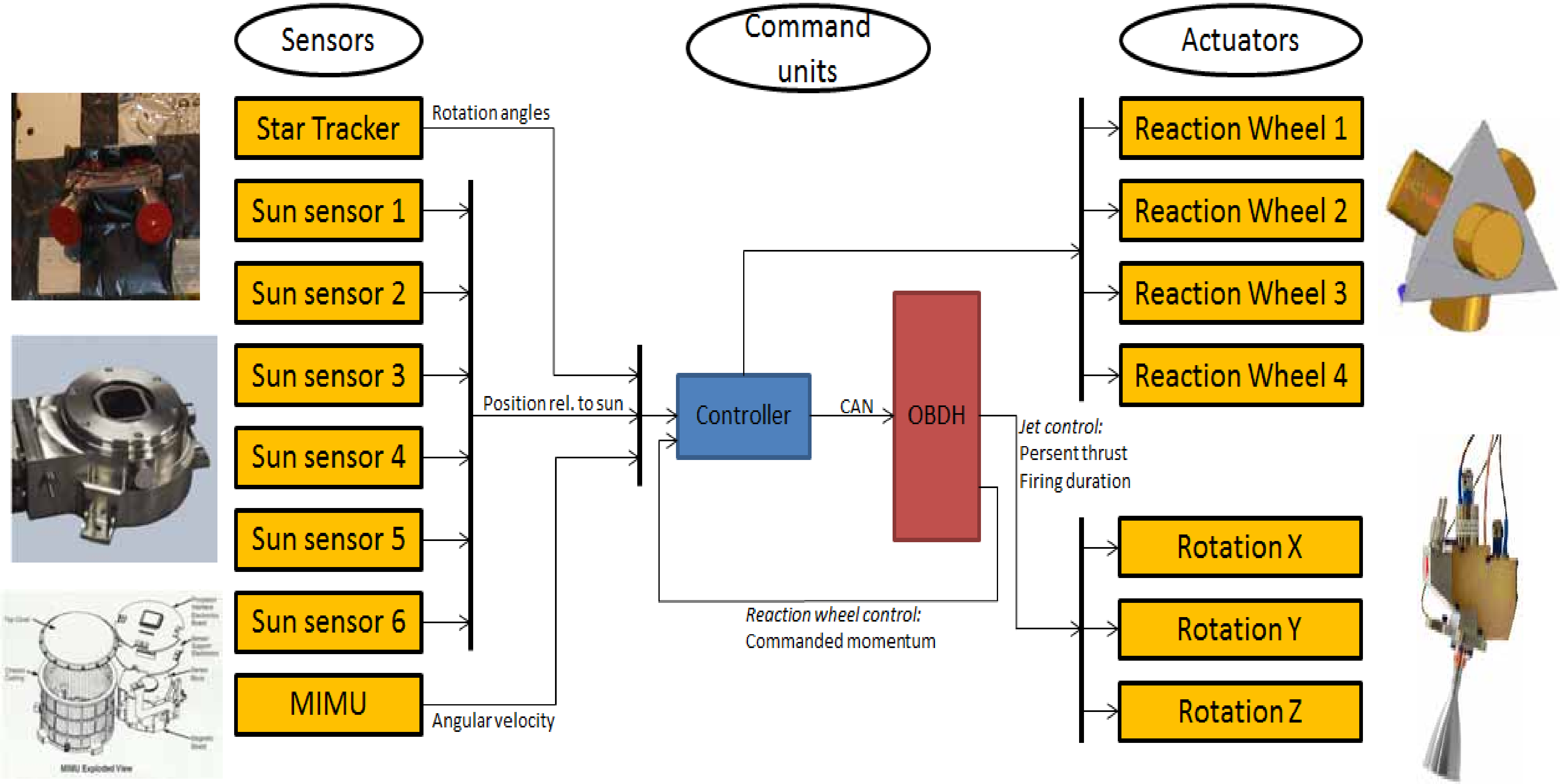
Trade-off: spinner versus 3-axis stabilized

Composed of

- Sensors (to measure actual attitude)
  - Star trackers measure attitude wrt to inertial directions and have accuracy between 1 arcsec and 1 arcmin
  - Sun sensors have accuracy between 20 arcsec and 1 deg
  - Gyros measure angular rates and can be used together with Star trackers for high accuracy pointing
- Actuators (to achieve desired attitude)
  - Reaction wheels
  - Thrusters

Choice of sensors and actuators widely depends on requirements

# GNC Guidance and Navigation Control
## (AOCS Attitude and Orbit Control System)

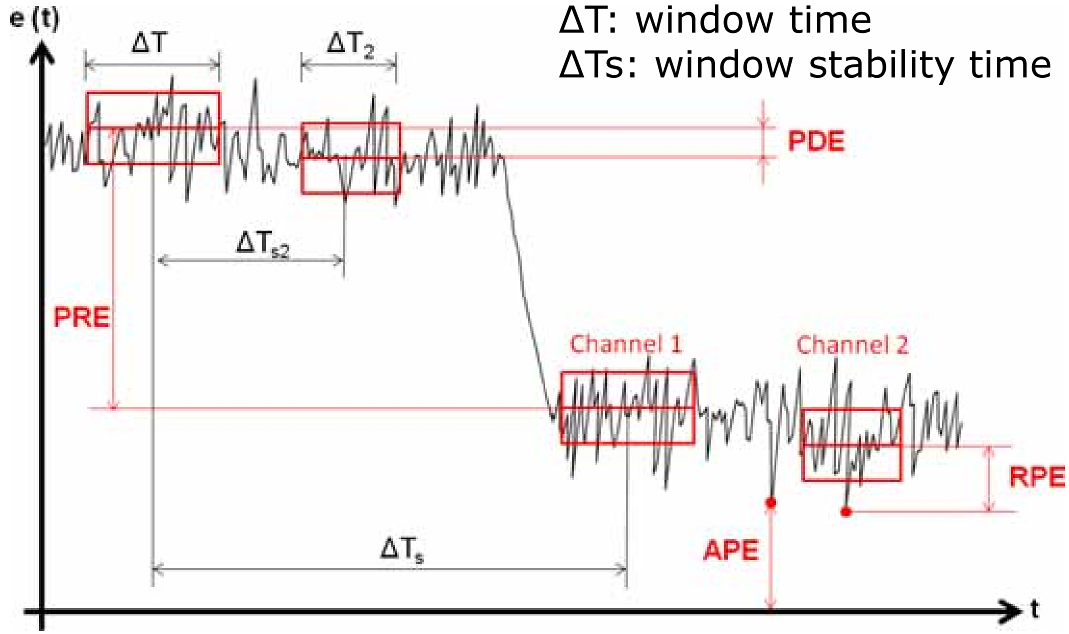


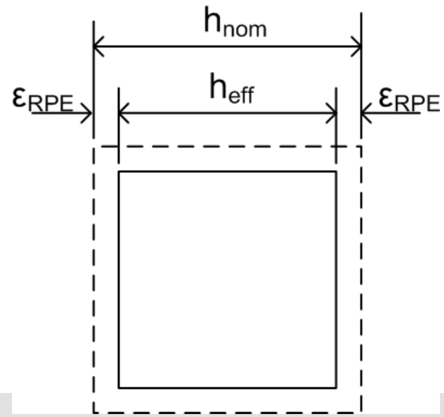

**Fully redundant system , 3-axis stabilized, Δφ <0.05deg, ω < 0.15 deg/s**

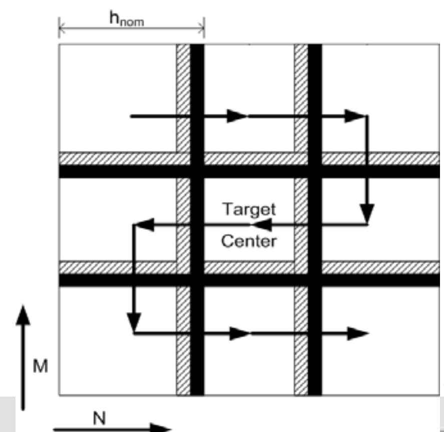
# Terminology – Pointing Error Indices

- APE=absolute performance error: instantaneous pointing at the right scene
- MPE=mean performance error: average pointing at the right scene
- RPE=relative performance error: stability during payload integration time
- PRE=performance reproducibility error: overlapping of two observation series
- PDE=performance drift error: drift between repeated observations
- AKE, MKE, RKE: absolute/ mean/ relative knowledge error

t: instantaneous time
ΔT: window time
ΔTs: window stability time



# Electrical Power Subsystem (EPS)

**How Much Power Does a Spacecraft Need?**

**Small** (Light-Bulb Sized)

Mars Climate Orbiter; Mars Odyssey: 300W

Mars Polar Lander; Mars Exploration Rover: 150W

Stardust; Genesis: 200W

**Medium** (Hair Dryer Sized)

Mars Reconnaissance Orbiter (1kW)

Metereological Satellites (2kW - 5kW)

Commercial & Military Communication Satellites (1kW - 15kW)

**Large** (House-Sized)

Hubble Space Telescope (25kW)

NASA / International Space Station (50kW)

**Monster** (City-Sized)

Lunar & Martian Stations (100kW - 1MW)

# Electrical Power Subsystem (EPS)

**Provides electrical power to S/C and payload**

**Solar Panels**

Panels need to point to the sun

Need special design or rotation mechanism

Solar Flux at Earth ~1400 W/m2, at Mars ~600W/m2

MEX: 11.4m2 sillicon cells ~10% eff. => ~500 Watts

(new technologies increase efficiency ~30%)

**Battery**

Needed for eclipses, emergency, …

MEX: Lithium-Ion battery 67 Amp hour

(60 times a normal phone battery)

**Alternative Power sources**

Nuclear Power (RTG, RHU's, ASGR's), constant power

(necessary for missions beyond Jupiter)

# Electrical Power Subsystem (EPS)

## Nuclear Sources

Basically using heat generated by radioactive decay and a thermo-electric converter

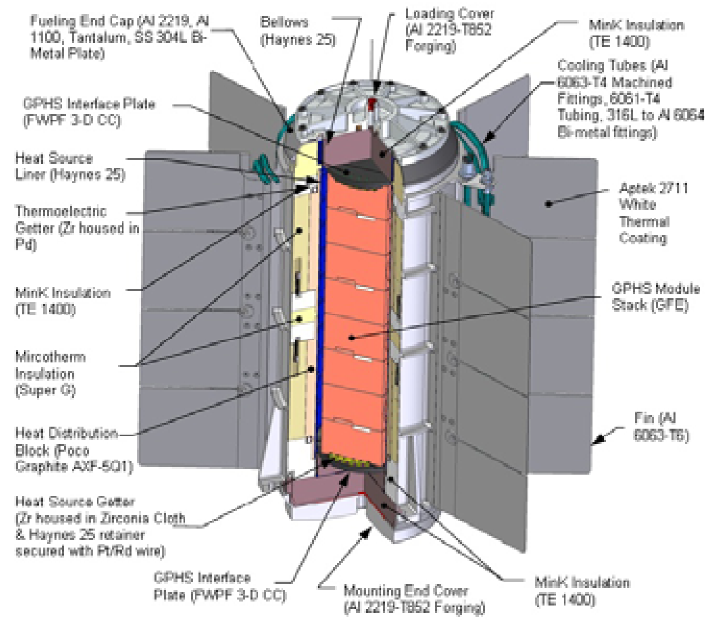
| Name | Electrical | Thermal | Mass |
|---|---|---|---|
| MMRTG ($^{238}Pu$) | 110 W | 2000 W | 45 kg |
| ASRG ($^{238}Pu$) | 160 W | 500 W | 34 kg |
| ESA ($Am_2O_3$) | <1 W/kg | | |

Degradation

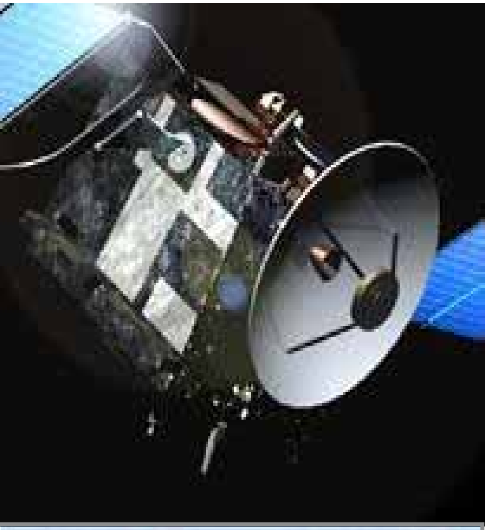
- Half life: Pu (88 yrs), $^{241}Am$ (433 yrs)
- Themo-electric element: ~0.8% /yr

Radioisotope Heating Unit (RHU)

- US: 1 W 40 g
- Rus: 8 W 200 g




# Communications



**Earth-Spacecraft-Lander transmission**

Science Data, Commands, housekeeping,

Tracking (location, velocity), Radio science

**Radio Link**

Data rate increases with antenna size and frequency,

Data rate decreases with distance (MEX maximum 228kbps X-band)

- At short distance low frequency is enough:
  - S band (2 GHz), UHF (<1GHz) for lander
- At longer distance higher frequency needed:
  - X (or Ka)-band (7/32 GHz)

**Spacecraft Antenna**

MEX High Gain Antenna 1.6m diameter X/S-band

MEX Low Gain Antenna (emergency) 20cm S-band + UHF for lander

**Ground Stations**

ESA 35m diameter: Madrid, Australia, Argentina

NASA 35/70m diameter: Madrid, Australia, USA

# Link Budget

Telemetry budget, receiver margin:

$E_b/N_0$ ratio of received-energy-per-bit to noise-density

Transmitter

$P$ transmitter power

$L_l$ transmitter line loss

$G_t$ transmitter antenna gain (area, shape, $\lambda^{-2}$)

Transmission

$L_s$ space loss (slant range)

$L_a$ transmission path loss (atmosphere, etc)

Receiver

$G_r$ receiver antenna gain

$kT_s$ system noise energy

Data rate: $R$

$$\frac{E_b}{N_0} = \frac{P \cdot L_l \cdot G_t \cdot L_s \cdot L_a \cdot G_r}{k \cdot T_s \cdot R}$$

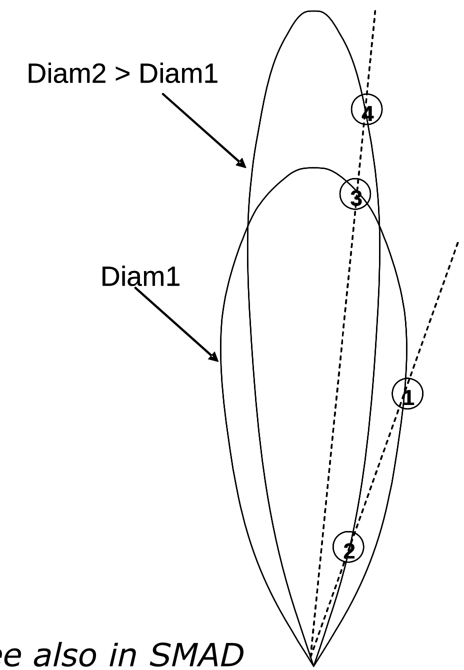
*see also in SMAD*

# HGA Performance

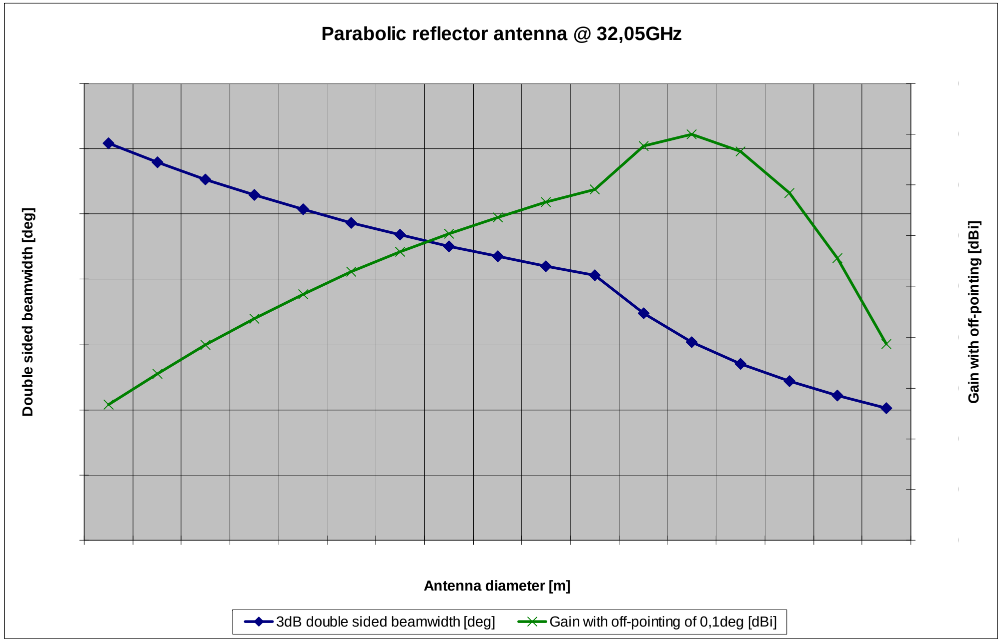
High Gain Antenna (HGA) versus pointing performance

Optimum antenna diameter for known AOCS off-pointing

Further iteration to be done once AOCS performance is known




*see also in SMAD*



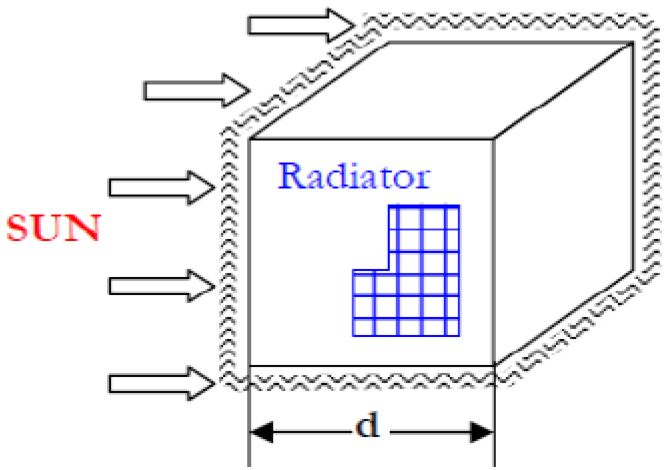
# Thermal Control Subsystem

The subsystem that allows keeping the spacecraft and payload temperatures within allowable limits

Generally, separated thermal control for spacecraft and payload due to different temperature requirements

Basic principles:

- Insulate the spacecraft from the environment to keep stable temperatures inside and provide an aperture for dissipation of excess heat (radiator).
- During eclipse provide heating power to keep the spacecraft warm




- thermal blankets (MLI)
- external paints to modify optical properties
- radiator(s), associated heat transport devices (heat pipes, high conductivity paths)

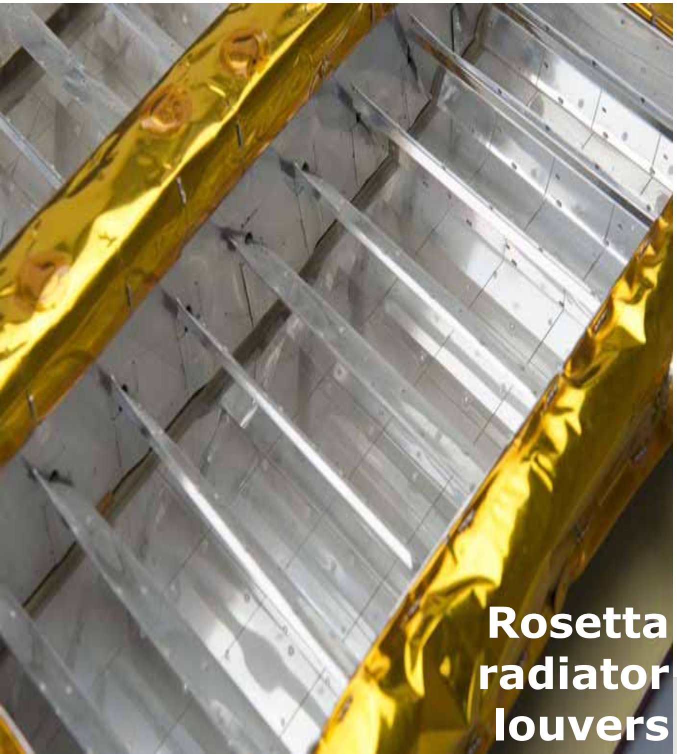
# Thermal Control

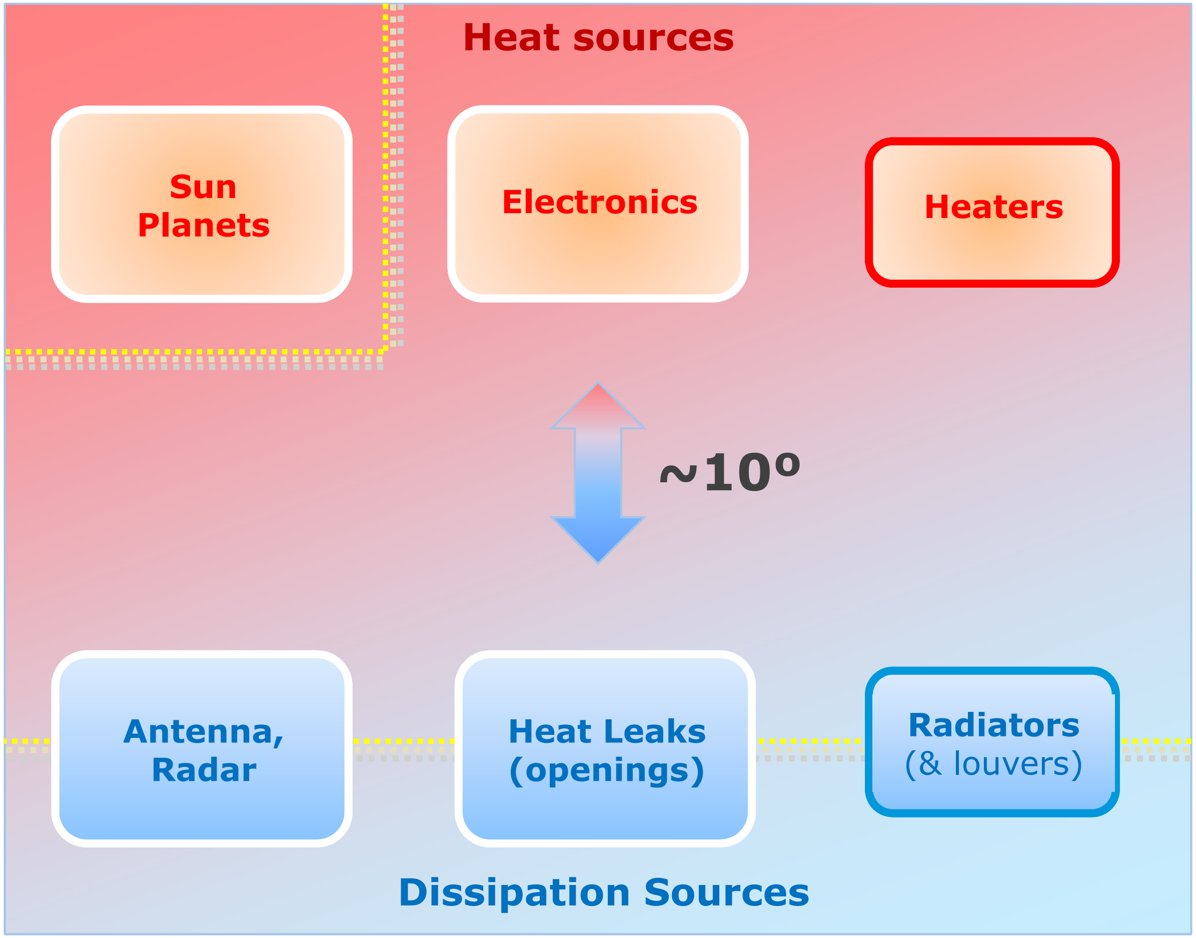
External Temperatures -100 ~ +150ºC

Multi-Layer Insulator to avoid illumination and dissipation

**Most electrical power is converted into heat**

Radiators + Heaters + pipes…



# Thermal Control

**Controls spacecraft thermal environment  in various mission phases**:

Launch, transfer, science mode, safe mode, eclipse, etc…

Driven by equipment and payload requirements

Need careful analysis of all modes (internal dissipation and external input) under various aspect angles and for all mission phases.

**Sensors:** temperature sensors

**Control Components:**

– Coatings, MLI, paint, radiators, sun shields, foam, heat pipes, optical reflectors, louvers, fillers, thermal insulators, cooler, cold plates, phase change devices, electrical heaters & thermostats, RHU's,….

– Using of emission (ε), absorption (α) values

– Requires: Geometrical Mathematical Model (GMM) and Thermal Mathematical Model (TMM) – e.g. ESATAN

– Need to understand the characteristics and dissipation of all s/c equipment

**Attention to: thermo-elastics (co-alignment), outgasing**

# Thermal Design

Spacecraft Thermal verification:

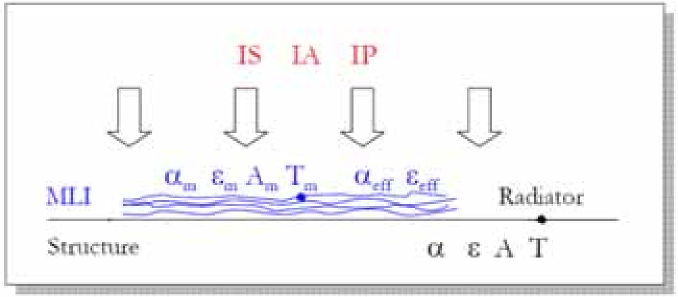
Assume single node and make thermal balance

$$Q_{\text{int}}^{\text{in}} + Q_{\text{ext}}^{\text{in}} = Q^{\text{out}}$$

$$Q^{\text{out}} = \varepsilon\sigma T^4 A_{\text{rad}}$$

$$Q_{\text{int}}^{in} = P_{\text{dissipated}}$$

$$Q_{\text{ext}}^{\text{in}} = \alpha A_{\text{exposed}} S + Q_{\text{PlanetIR}} + Q_{\text{Albedo}}$$




3. Solve first for $A_{\text{rad}}$ (radiator area) fixing max allowed $T$ and optical properties – hot case
4. Solve then to find $P_{\text{dissipated}}$ needed to keep $T$ within limits in eclipse for $A_{\text{rad}}$ –cold case

# On Board Data Handling (OBDH)

**Data Management System**

Telecommand distribution

Telemetry data

Events, housekeeping, …

**Control and Data Managament Units**

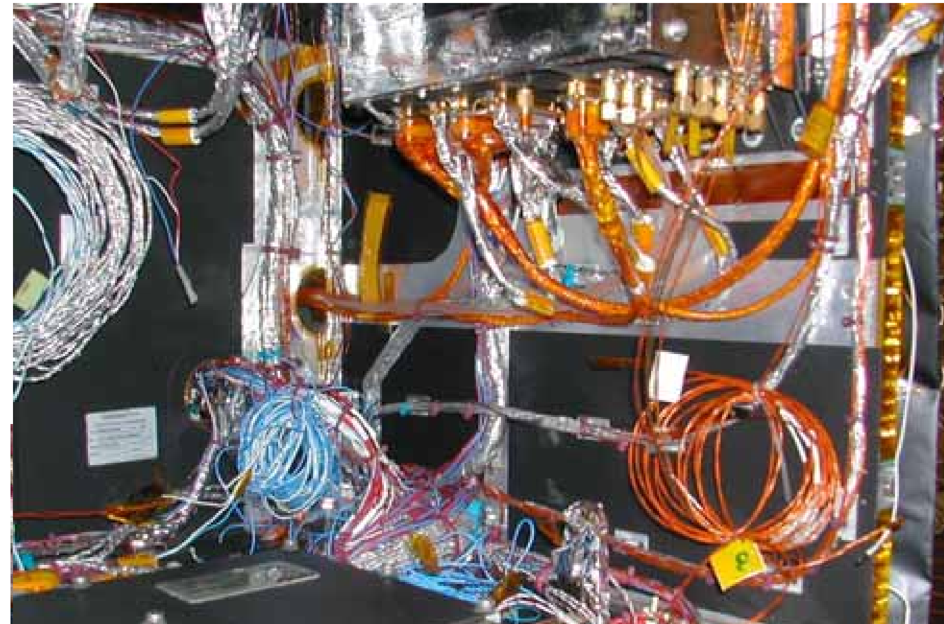
4 Processor Modules (2 DMS + 2 AOCS)

**Bus Architecture + High speed Link**

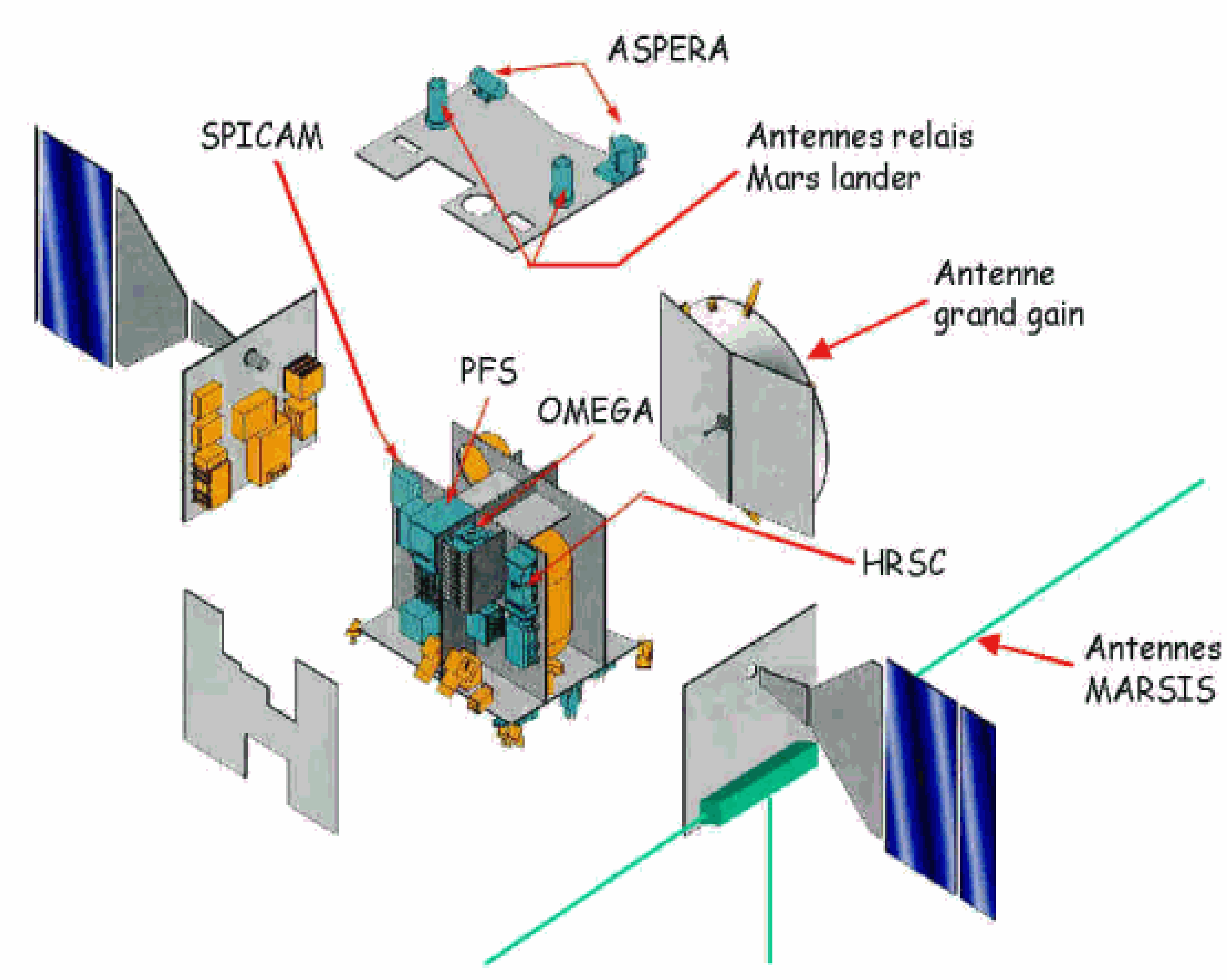
**Solid State Mass Memory**

Payload Data Handling Unit

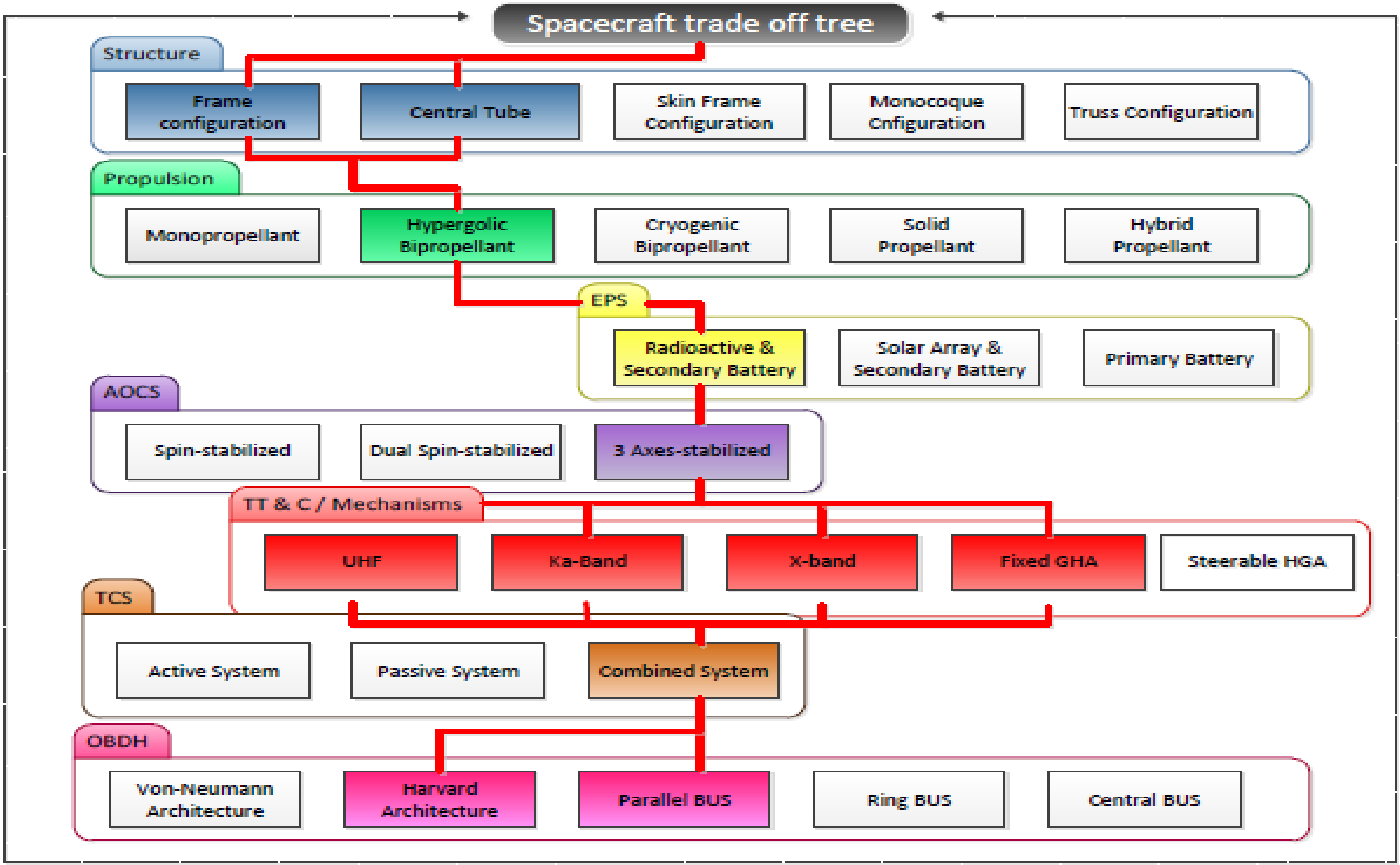
(MEX: 8 Gbit → EXM: 1024Gbit)



# Payload Subsystems



# Subsystem Trade-off Tree



# System Summary

**Mission profile & lifetime**

**Launcher**: launch mass, fairing

**Budgets**: Mass, Power

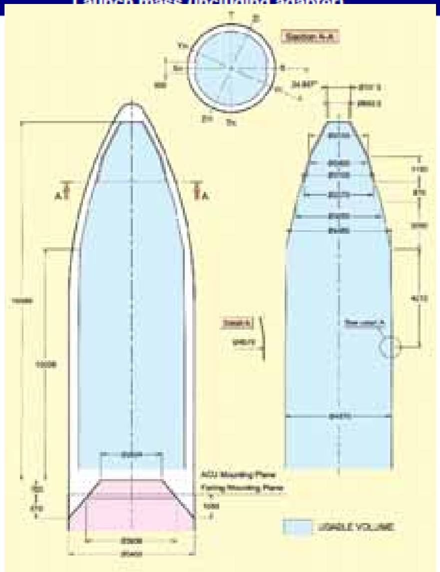
**Total system margin**

Equipment 5~20% (based on TRL)

Total System Level 20%

**Operations, Cost, risks, schedule**

| | | | | | | | | |
|---|---|---|---|---|---|---|---|---|
| | | | Target Spacecraft Mass at Launch | | | | 950.00 | kg |
| | | | Below Mass Target by: | | | | 920.00 | kg |
| | Input Mass | Input Margin | Dry mass contributions | Without Margin | Margin % | Margin kg | Total kg | % of Total |
| EL | | | Structure | 0.00 kg | - | - | - | - |
| EL | | | Thermal Control | 0.00 kg | - | - | - | - |
| EL | | | Mechanisms | 0.00 kg | - | - | - | - |
| EL | | | Communications | 0.00 kg | - | - | - | - |
| EL | | | Data Handling | 0.00 kg | - | - | - | - |
| EL | | | AOCS | 0.00 kg | - | - | - | - |
| EL | | | Propulsion | 0.00 kg | - | - | - | - |
| EL | | | Power | 0.00 kg | - | - | - | - |
| DI | | | Harness | 0.00 kg | 0.00 | 0.00 | 0.00 | 0.00 |
| EL | | | Instruments | 0.00 kg | 0.00 | 0.00 | 0.00 | 0.00 |
| | | | Total Dry(excl.adapter) | 0.00 | | | 0.00 kg | |
| | | | System margin (excl.adapter) | | 20.00 % | | 0.00 kg | |
| | | | Total Dry with margin (excl.adapter) | | | | 0.00 kg | |
| | | | Other contributions | | | | | |
| | | | Wet mass contributions | | | | | |
| EL | | | Propellant | 0.00 kg | - | - | - | - |
| | | | Adapter mass (including sep. mech.), kg | 30.00 kg | 0.00 | 0.00 | 30.00 | 1.00 |
| | | | Total wet mass (excl.adapter) | | | | 0.00 kg | |
| | | | Launch mass (including adapter) | | | | 30.00 kg | |



# Margins – Contingencies

**Equipment level margins according to maturity**

- **+5%** for off-the-shelf items (no changes)
- **+10%** for off-the-shelf items with minor modifications
- **+20%** for new designs, new developments, major modifications

**System margin** (at least **+20%**)

- On top of and in addition to equipment margins; applied after summing best estimates + margin
- Two options for the propellant calculation +10% margin + 2% residuals
  - Margin on total dry mass and margin on launcher: typically used during early study phases +10% margin
  - Margin on maximum separated mass: typically used later, when mission analysis and launcher analysis become available

**Always keep lots of margins!**

“Margin philosophy for Science Assessment Studies” (*reference*)

# Operations

- Spacecraft and Instruments need to be controlled from Ground
- Launcher authority takes control until successful launcher insertion orbit and separation from upper/transfer stage

**ESA Science missions:**

- Mission Operations done by ESOC
  - Navigation and tracking, commissioning, control of s/c, upload of commands, monitoring of health status, planning of manoeuvres etc.
  - Organise download of data for next passes
- Science Operations done by ESAC
  - Instrument control, definition of instrument commands
- Science data distribution centres
  - Distribution of onboard data together with housekeeping to interested scientists
- Definition of observation cycles/modes

Do not underestimate cost and complexity of mission and science operation and data management

# Cost

**Cost estimate is very difficult !**

**3 basic methods:** Bottom up approach, parametric analysis or by analogy with other missions

Need cost model and data base with cost info

Most difficult is the estimate on engineering, validation & verification cost, manpower etc.

extra cost of technology TRL upgrade!

Cost is driven by complexity of mission

**Mission CaC: Cost at Completion comprises:**

Development cost

Procurement cost of the space segment (industrial cost)

Test facilities cost

Launch cost

Mission operation cost

Science operations cost (science planning, data processing and archiving)

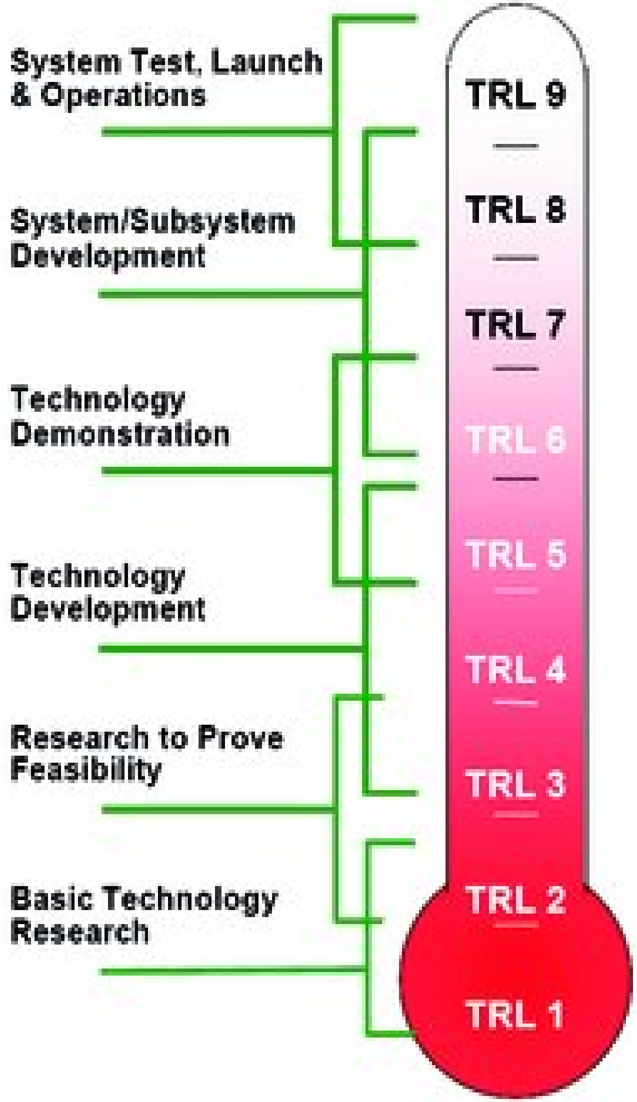
Agency cost and margins

Management costs

Payload cost

Contingency …

# Technology Readiness Levels (TRL)




Actual system "flight proven" through successful mission operations

Actual system completed and "flight qualified" through test and demonstration (Ground or Flight)

System prototype demonstration in a space environment

System/subsystem model or prototype demonstration in a relevant environment (Ground or Space)

Component and/or breadboard validation in relevant environment

Component and/or breadboard validation in laboratory environment

Analytical and experimental critical function and/or characteristic proof-of-concept

Technology concept and/or application formulated

Basic principles observed and reported

# TRL context

The TRL of a given technology is always evaluated in the **context** of a specific application, not by itself.

| Readiness Level | Definition | Explanation |
|---|---|---|
| TRL 1 | Basic principles observed and reported | Lowest level of technology readiness. Scientific research begins to be translated into applied research and development. (See Paragraph 4.2) |
| TRL 2 | Technology concept and/or application formulated | Once basic principles are observed, practical applications can be invented and R&D started. Applications are speculative and may be unproven. For SW, individual algorithms or functions are prototyped. (See Paragraph 4.3). |
| TRL 3 | Analytical and experimental critical function and/or characteristic proof-of-concept | Active research and development is initiated, including analytical / laboratory studies to validate predictions regarding the technology. For SW, a prototype of the integrated critical system is developed. (See Paragraph 4.4) |
| TRL 4 | Component and/or breadboard validation in laboratory environment | Basic technological components are integrated to establish that they will work together. For SW, most functionality is implemented. (See Paragraph 4.5) |
| TRL 5 | Component and/or breadboard validation in relevant environment | The basic technological components are integrated with reasonably realistic supporting elements so it can be tested in a simulated environment. For SW, Implementation of the complete software functionality. (See Paragraph 4.6) |
| TRL 6 | System/subsystem model or prototype demonstration in a relevant environment (ground or space) | A representative model or prototype system is tested in a relevant environment. For SW, ready for use in an operational/production context, including user support. (See Paragraph 4.7) |
| TRL 7 | System prototype demonstration in a space environment | A prototype system that is near, or at, the planned operational system. For SW, used in IOD or applied to pilot project. (See Paragraph 4.8) |
| TRL 8 | Actual system completed and "flight qualified" through test and demonstration (ground or space) | In an actual system, the technology has been proven to work in its final form and under expected conditions. For SW, ready to be applied in the execution of a real space mission. (See Paragraph 4.9) |
| TRL 9 | Actual system "flight proven" through successful mission operations | The system incorporating the new technology, or software, in its final form has been used under actual mission conditions. (See Paragraph 4.2.10) |

# Risks

- Space missions are large investments
- Understanding and analysis of risk is important to avoid catastrophic events as much as possible
- Identification of risk followed by risk mitigation is the approach
- Make a risk Register with main risks
- Rated with likelihood (A-E) and severity (1-5)

| Score | Likelihood | Definition |
|---|---|---|
| E | Maximum | Certain to occur, will occur once or more times per project. |
| D | High | Will occur **frequently**, about 1 in 10 projects |
| C | Medium | Will occur **sometimes**, about 1 in 100 projects |
| B | Low | Will occur **seldom**, about 1 in 1000 projects |
| A | Minimum | Will **almost never** occur, 1 in 10000 projects |

European Space Agency

# THANK YOU!

questions?