Ingeniería de Sistemas para Misiones Espaciales

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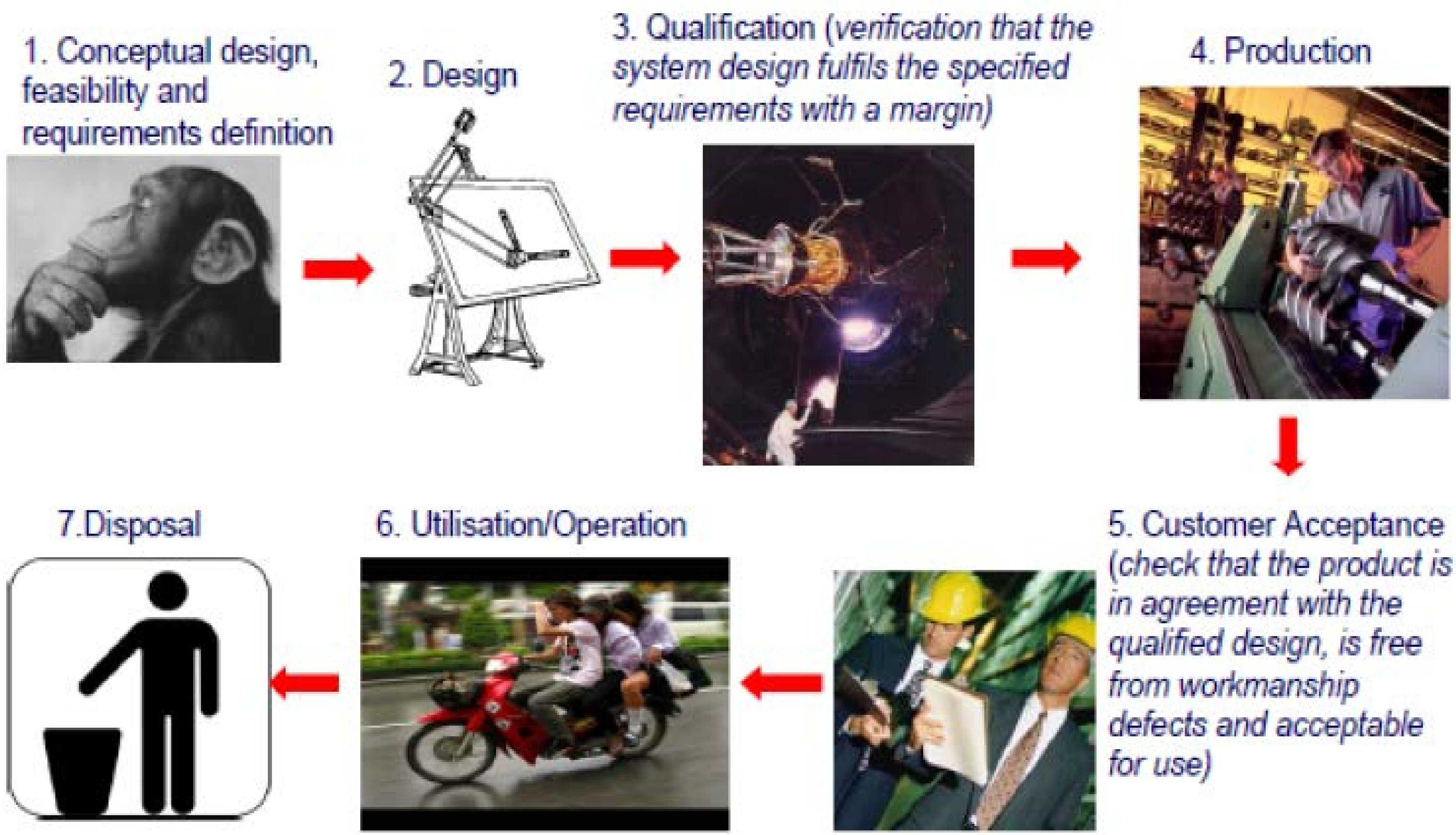
ESA Science Operations Center
Mars Express & ExoMars 2016
(slides courtesy of Christian Erd & Peter Falkner, ESA-ESTEC)

ESAC, 7th February 2017

Mission Flow Diagram

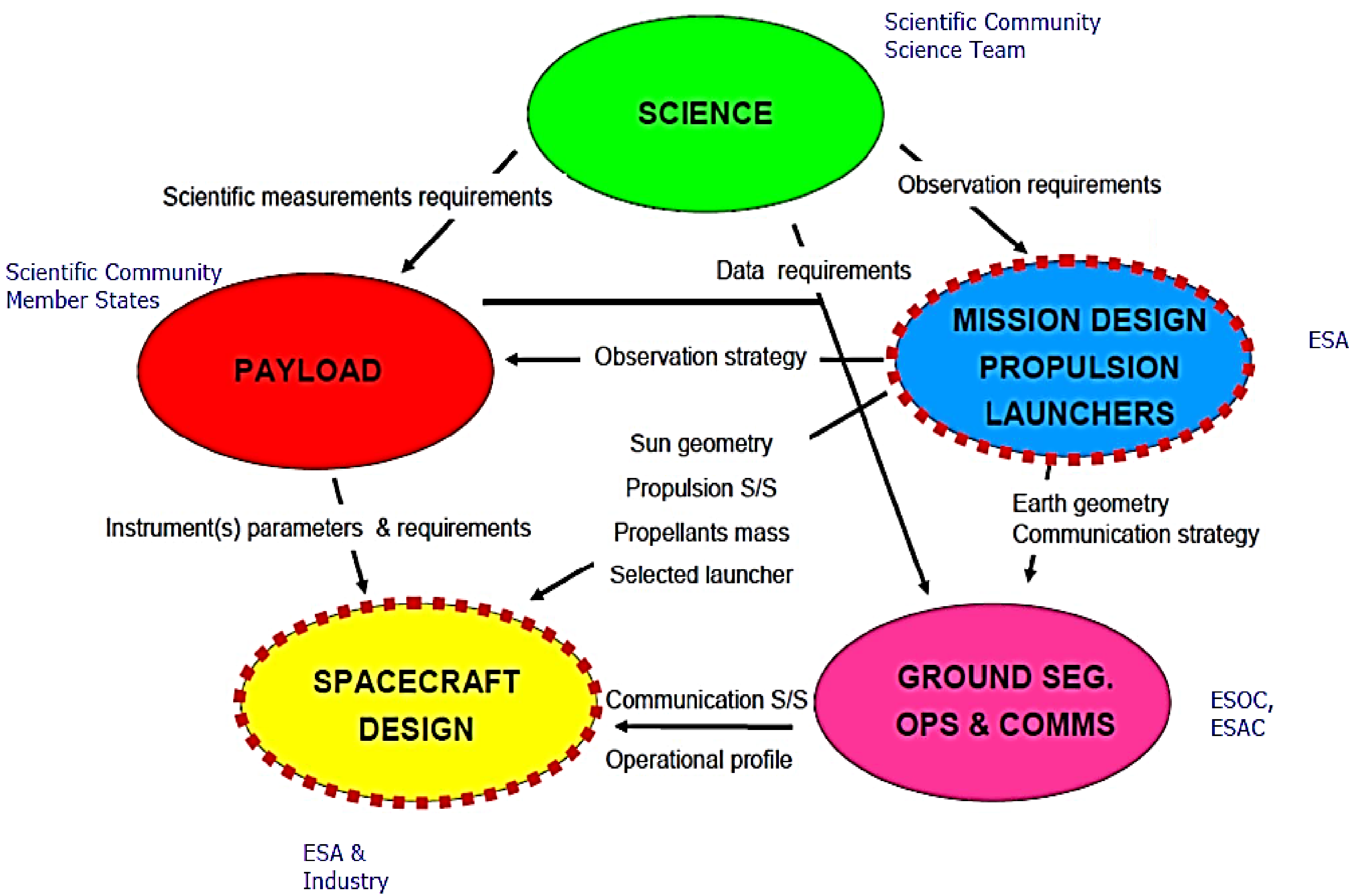


Space Mission Timeline: Phases

- **Pre-Phase A**: Concept Studies
- **Phase A**: Concept & Technology Development
- **Phase B**: Preliminary Design & Technology Completion
- **Phase C**: Final Design & Fabrication
- **Phase D**: System Assembly, Integration & Test, Launch
- **Phase E**: Operations & Sustainment

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

- **Science 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
  - Careful with conflicting requirements
- **Solution** is the response to the all requirements
  - There can be several solutions meeting requirements
  - Non-compliance needs to be negotiated

WHY?

WHAT?

HOW?

Conflicting Requirements

Farm “Optimal” Animal :

Eierlegende Wollmilchsau

(famous Austrian animal)

Trade-off

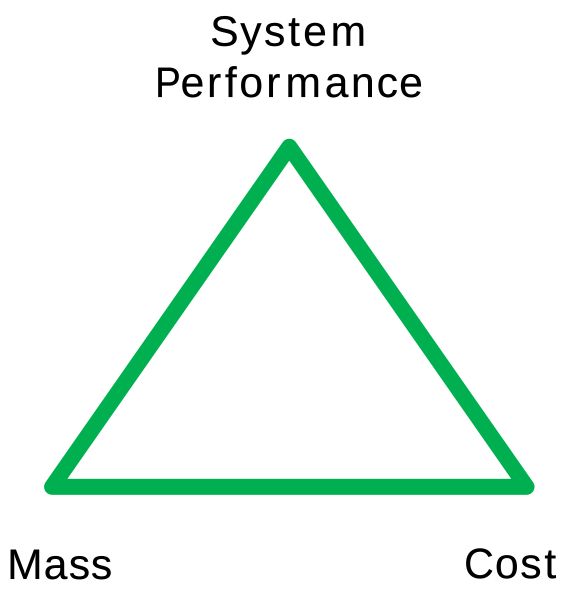
Trade-off allows exploring alternative solutions to a baseline

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

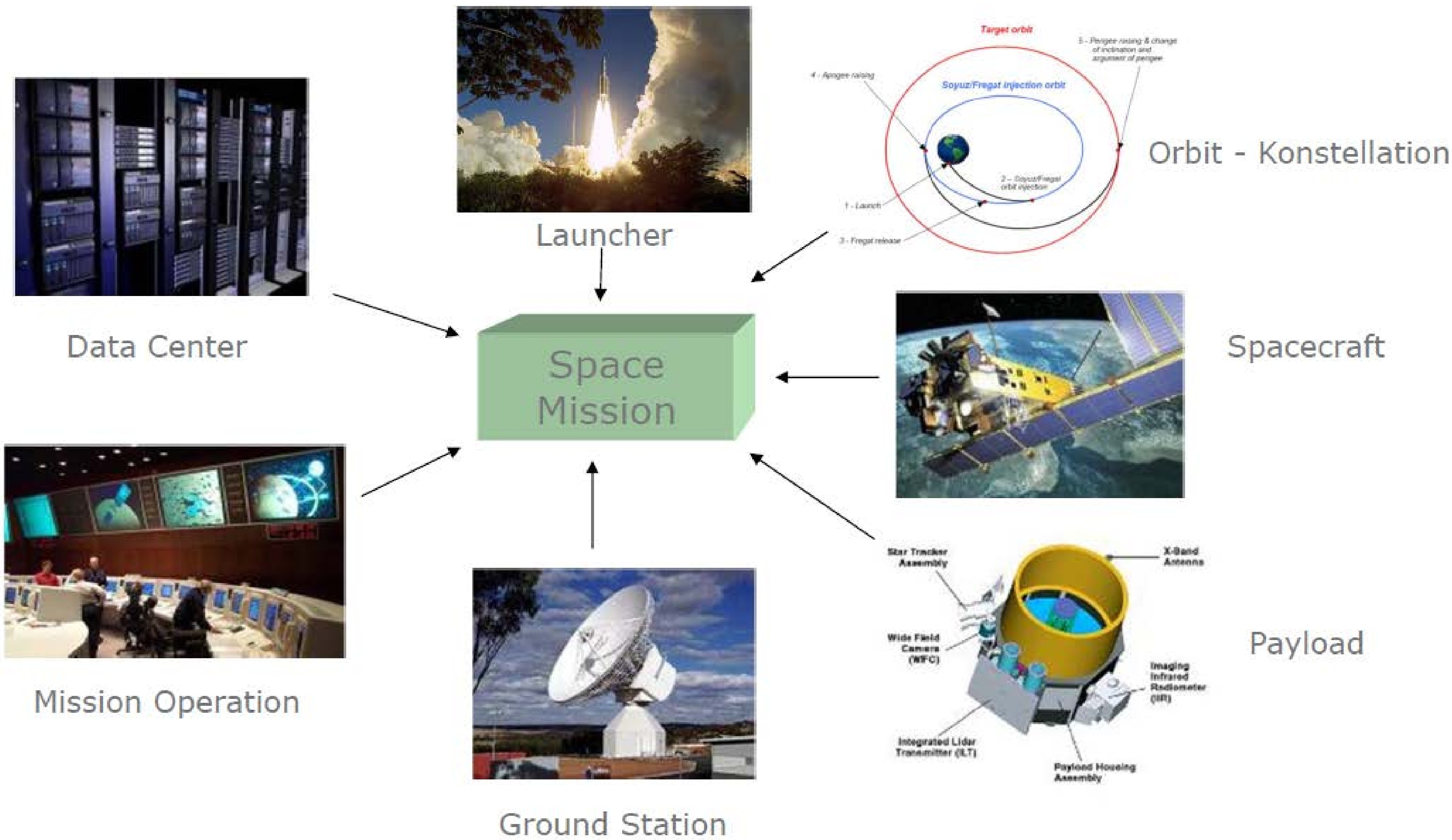
Power consumption → generation of more power → solar array size → mass

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

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

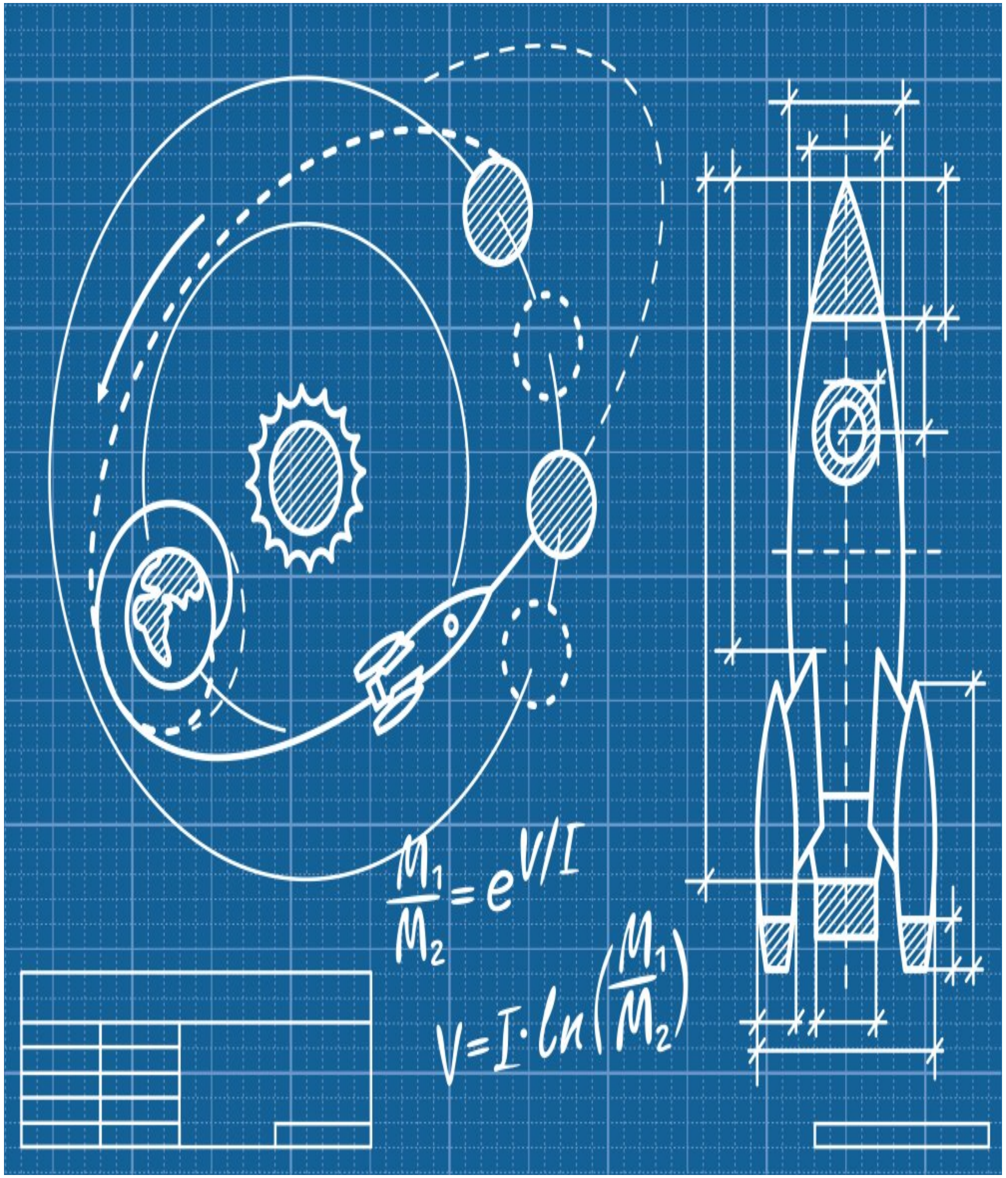


Mission Segments, Systems & Subsystems



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:

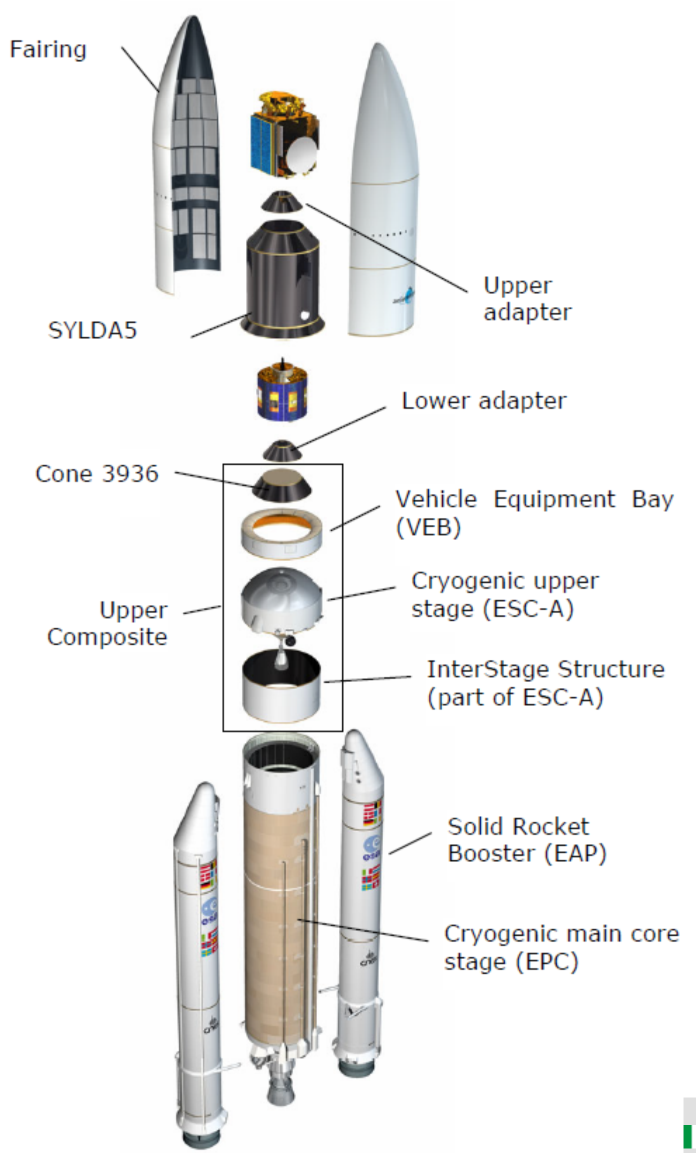
Vega (35M€) : 1500kg Low Earth Orbit

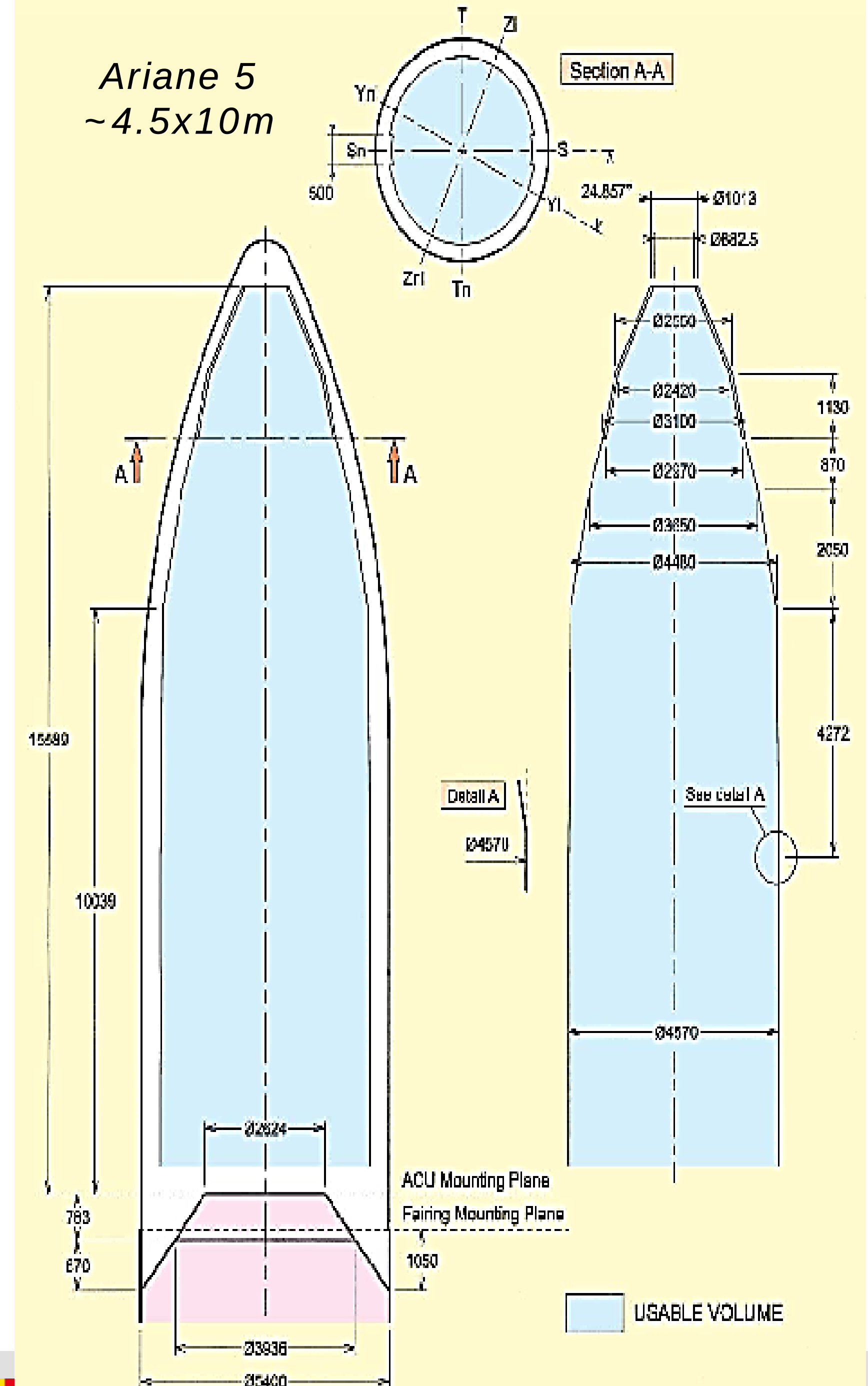
Soyuz-Fregat (70M€) : 3000kg GEO transfer

Ariane 5 (150M€) : 6000-10000kg GEO transfer

Mars Express on Soyuz-Fregat
Baikonur 2003

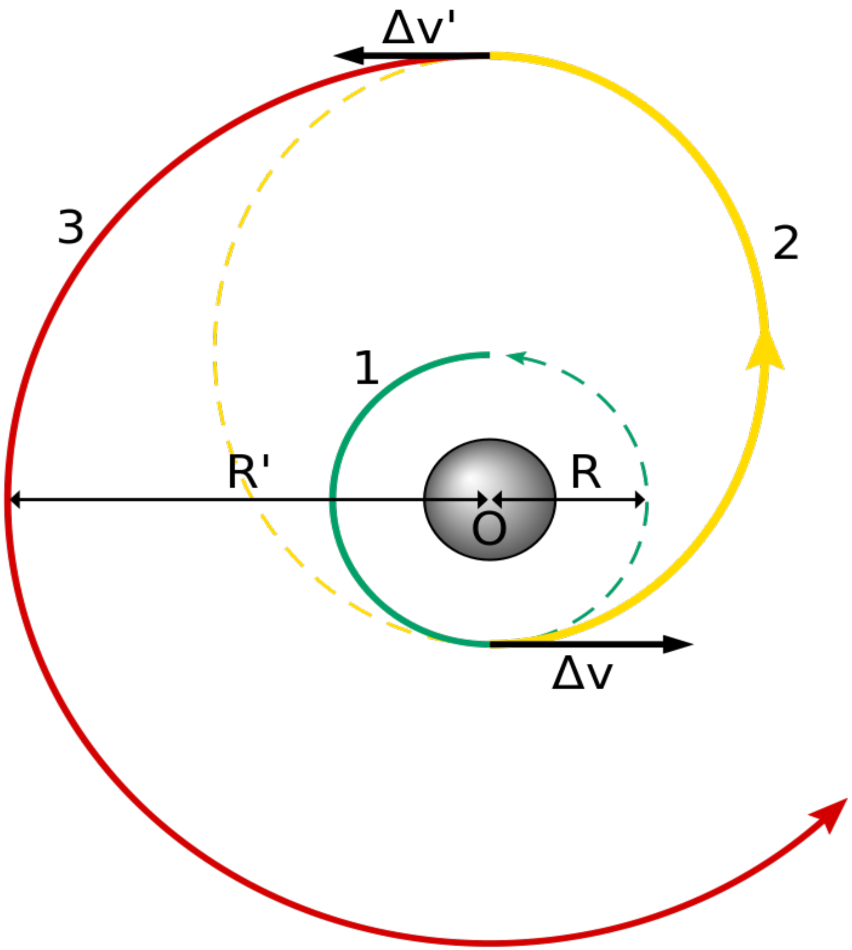
SC size needs to fit the Launcher!



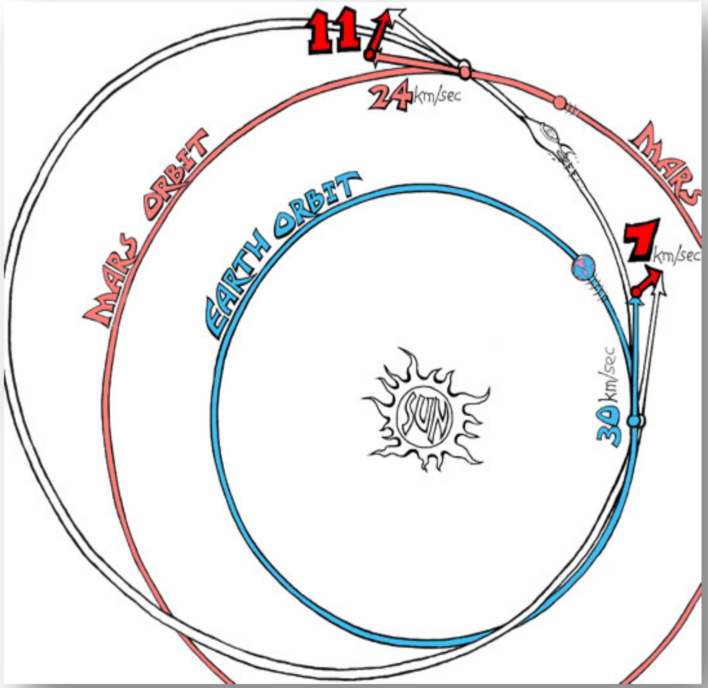


Interplanetary Transfer Orbit:
Simple Hohmann transfer trajectory

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



Other trajectories may be faster, but more expensive!



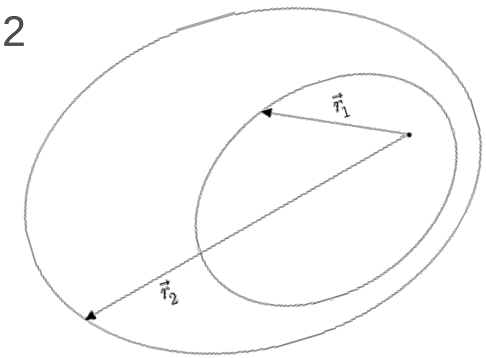
Lambert Problem - Cost function

Lambert Problem:

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

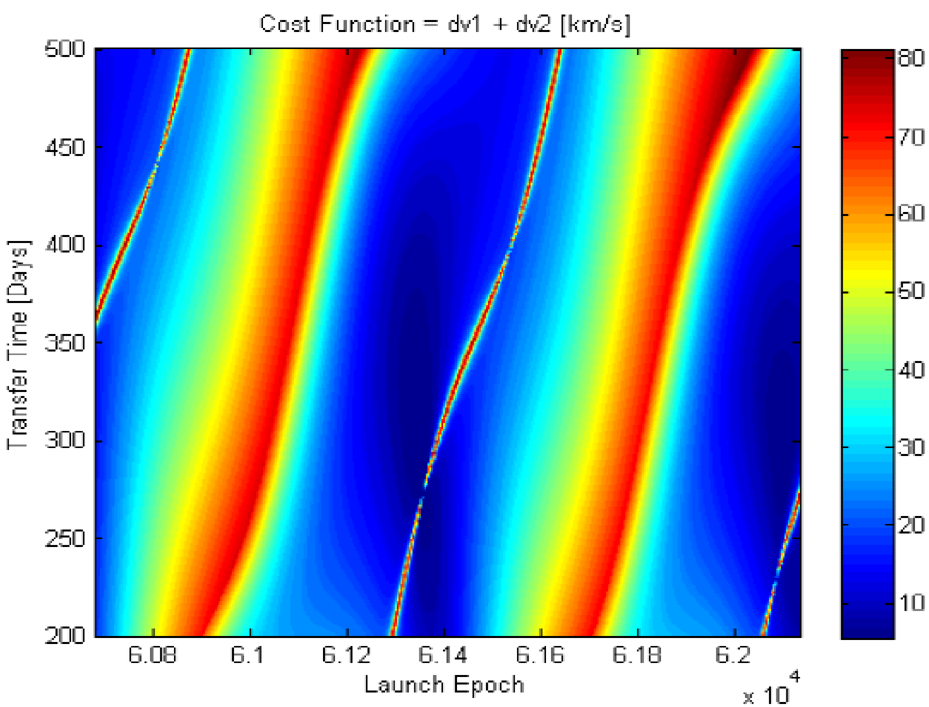
We are looking for the ellipse or hyperbola which connects r1 to r2

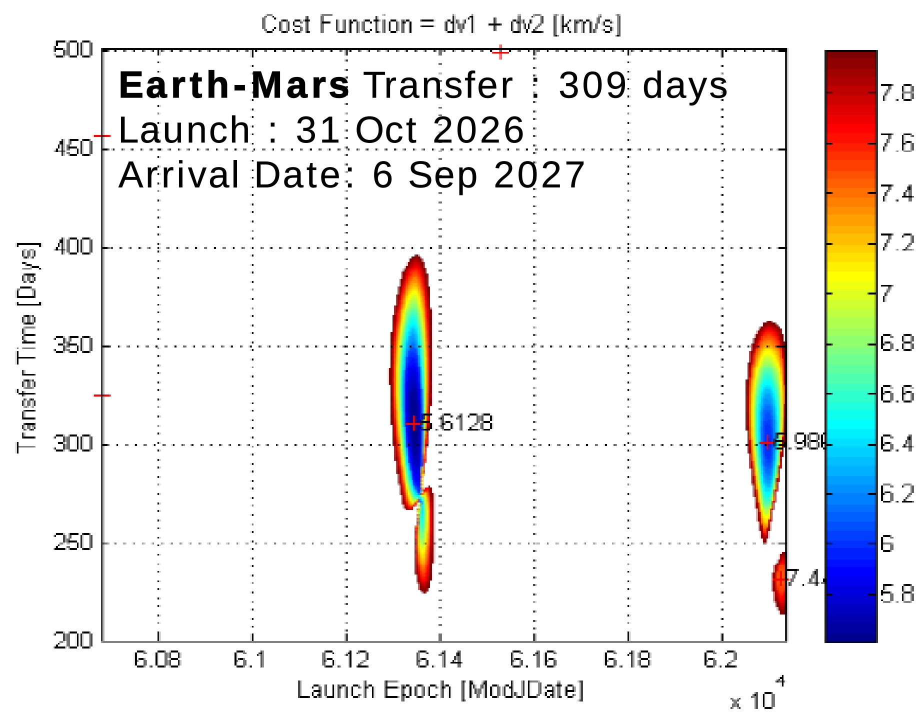
If we specify the time-of-flight(t -t = Dt), only one solution exists

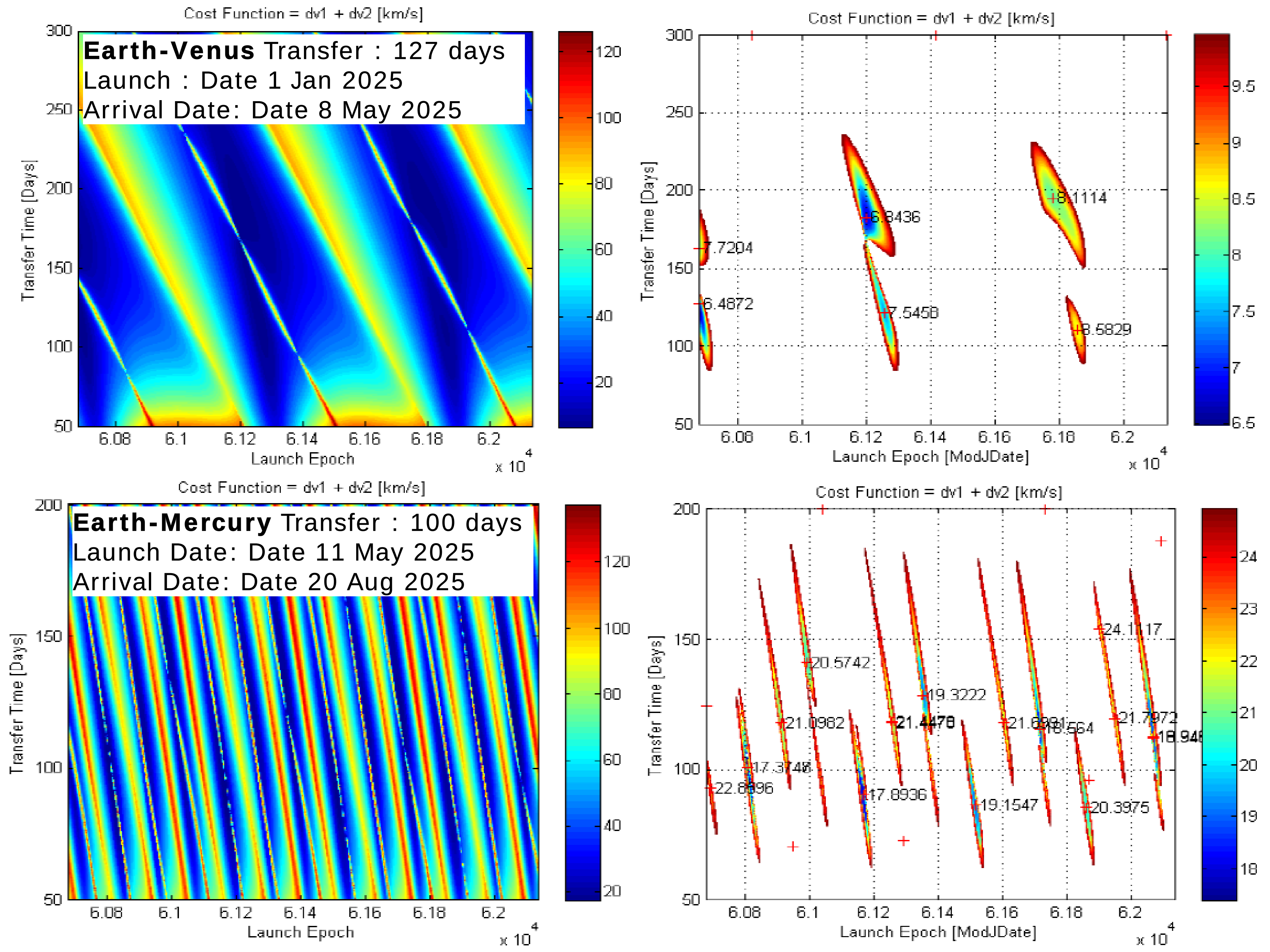


Cost function:

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

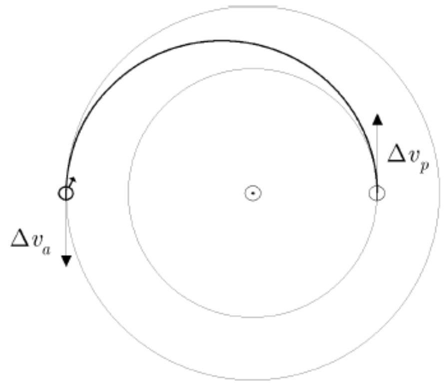


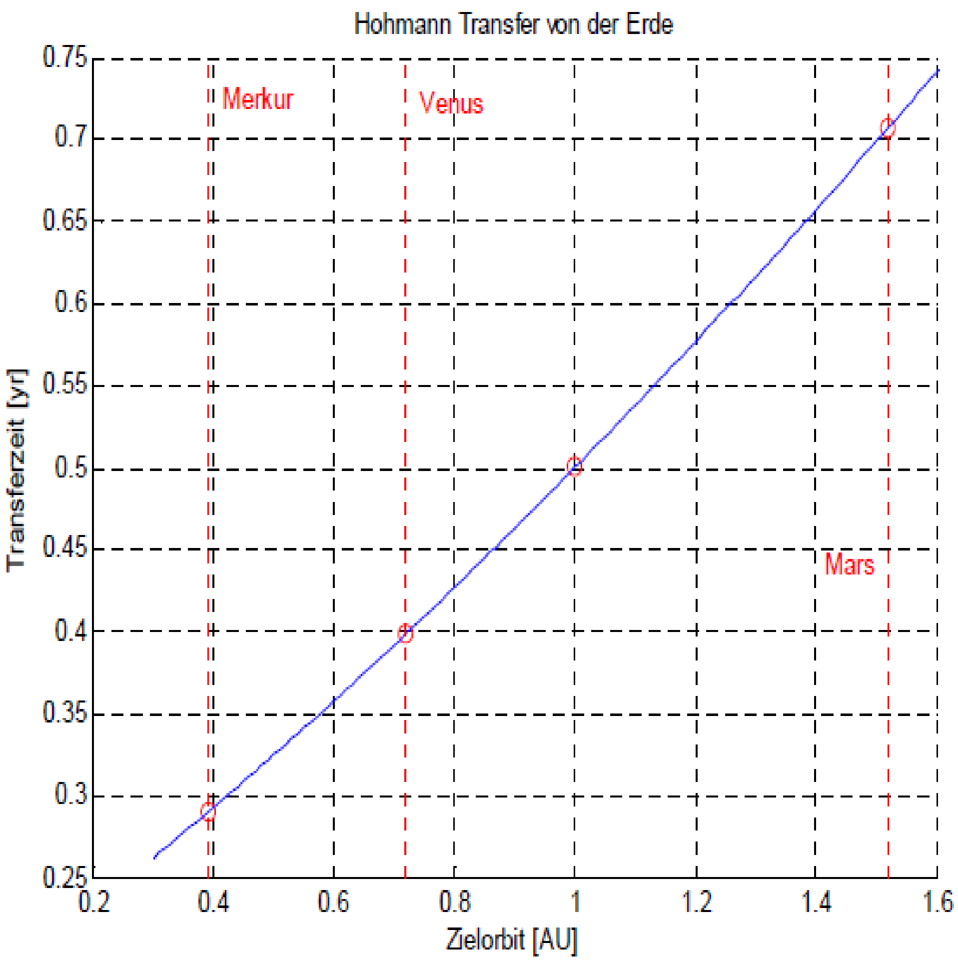


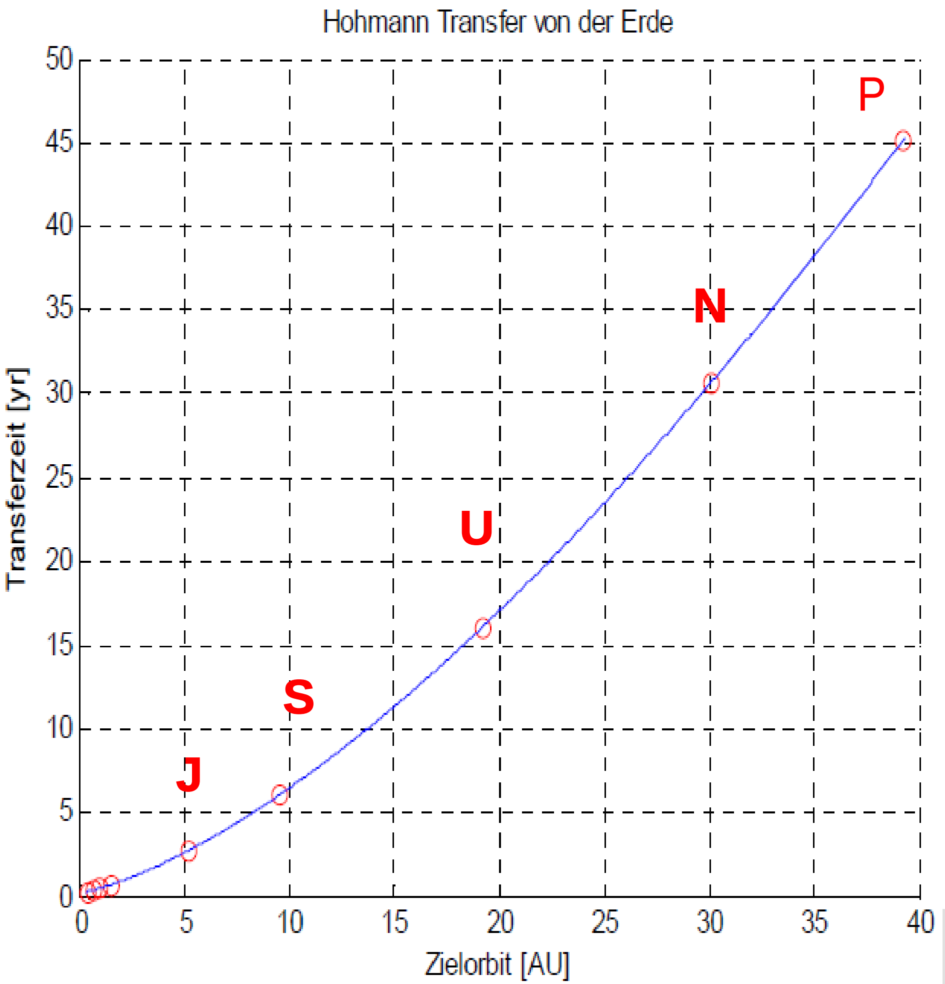


Cost Function = dv1 + dv2 [km/s]
Earth-Venus Transfer : 127 days
Launch : Date 1 Jan 2025
Arrival Date: Date 8 May 2025
Transfer Time [Days]
Launch Epoch
x 10^4
Cost Function = dv1 + dv2 [km/s]
7.7204
6.4872
6.8436
7.5458
8.1114
8.5829
Transfer Time [Days]
Launch Epoch [ModJDate]
x 10^4
Cost Function = 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]
24.1117
20.5742
19.3222
21.0982
21.4478
21.7972
17.3748
22.8396
17.8936
19.1547
20.3975
Transfer Time [Days]
Launch Epoch [ModJDate]
x 10^4

Hohhman Transfer time durations from Earth

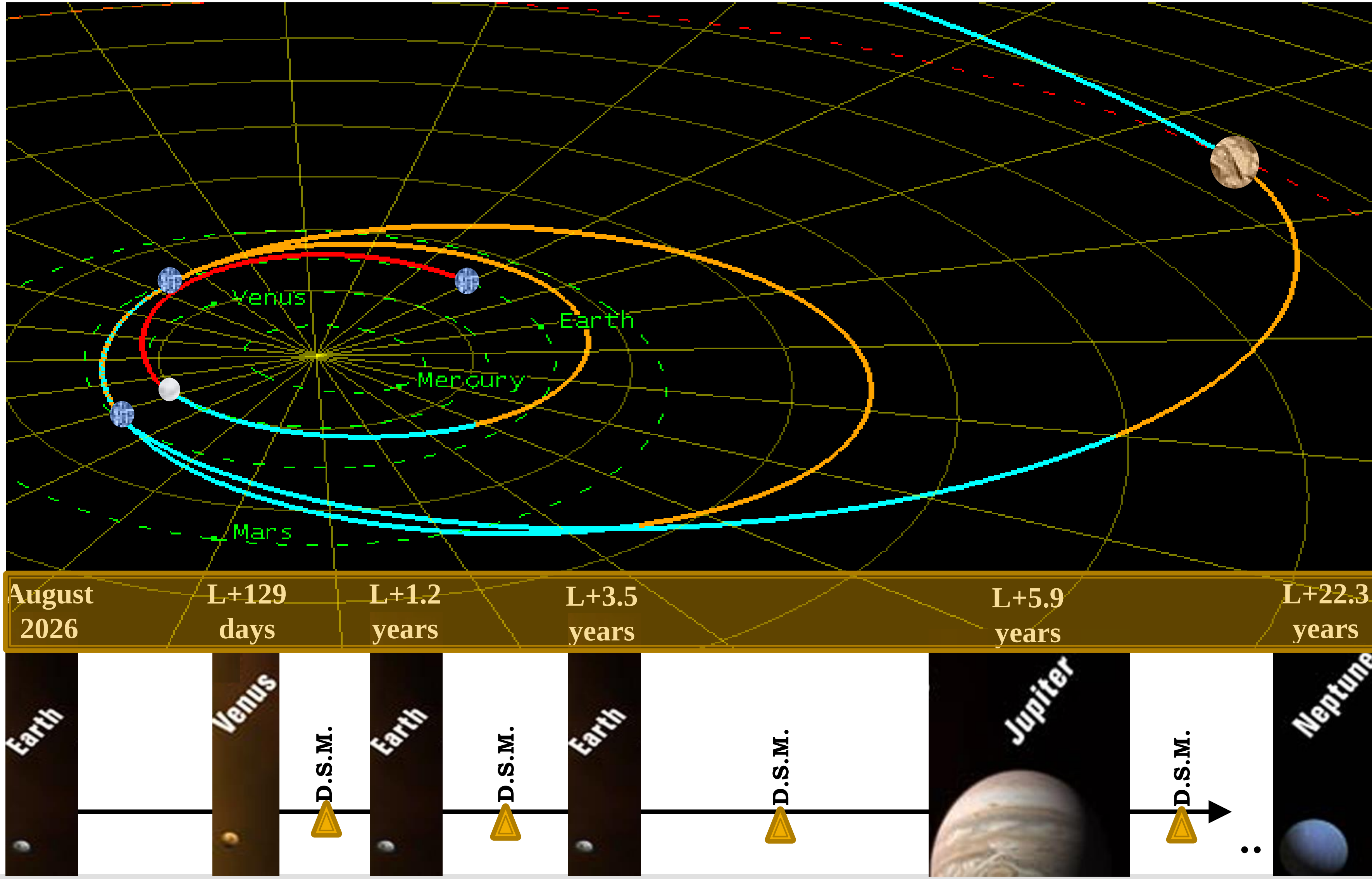




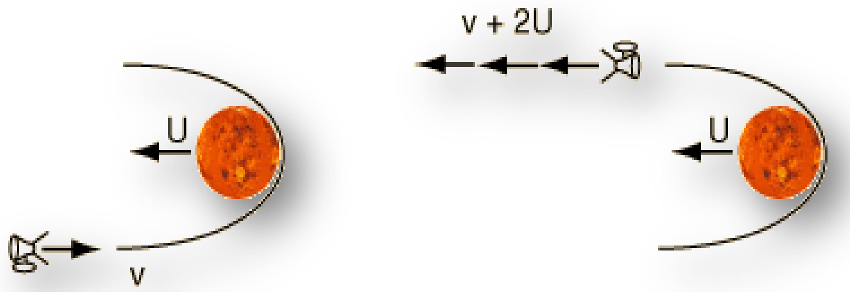


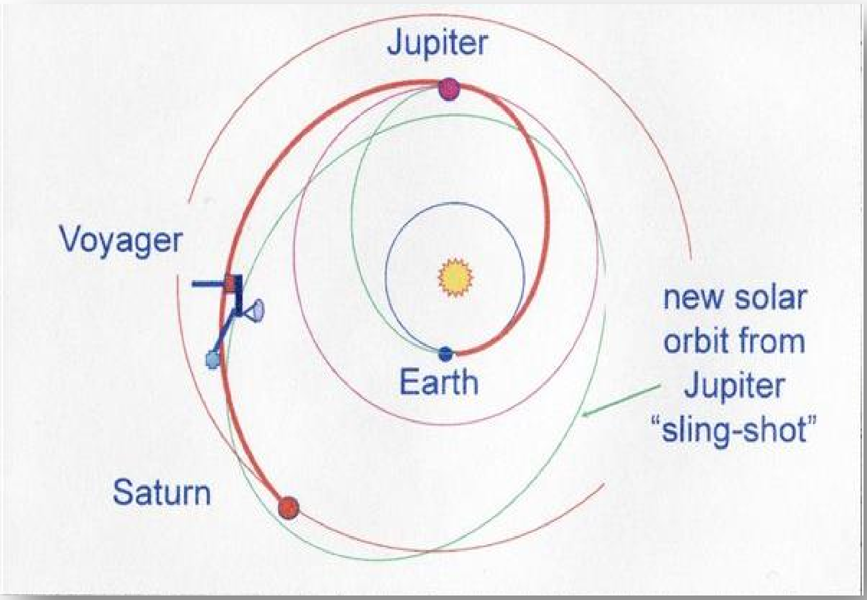
Interplanetary Transfer Orbit:

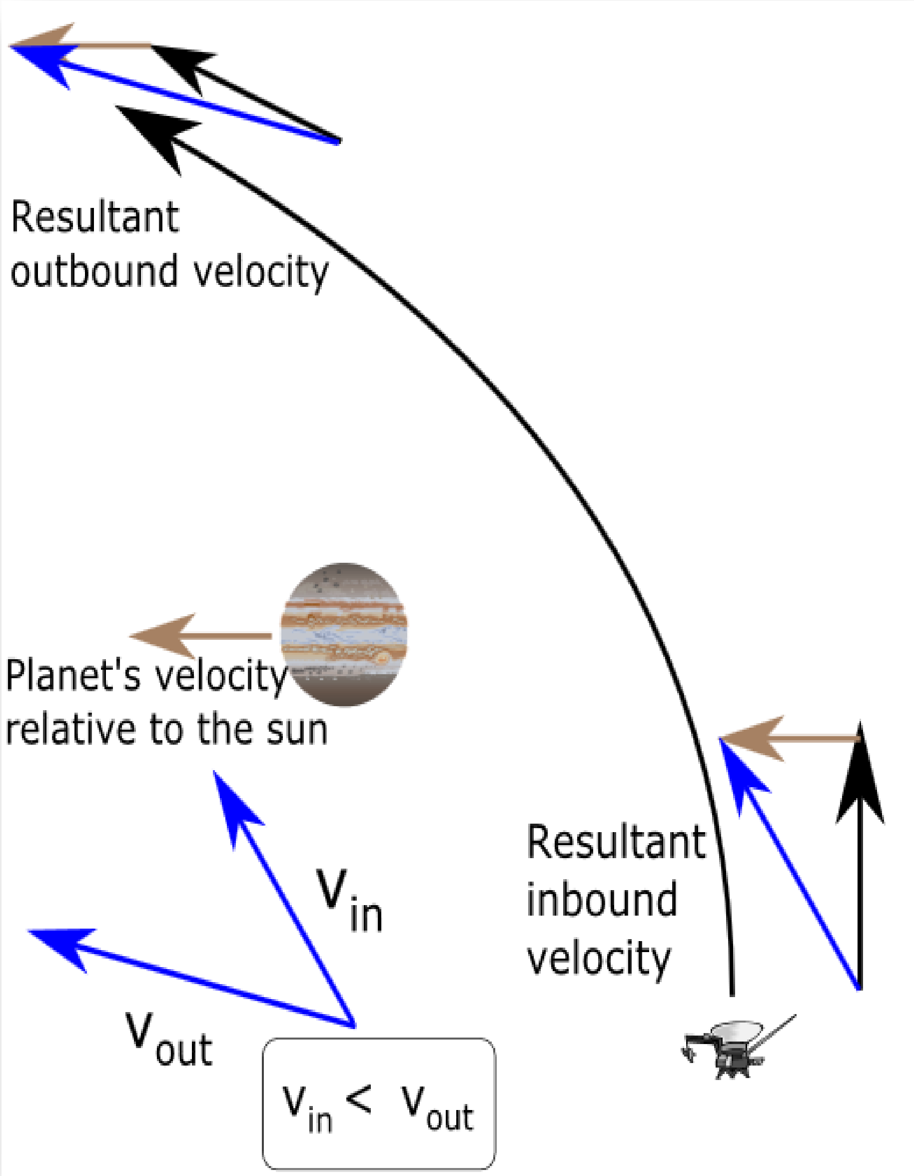
Other trajectories can be much more complex…



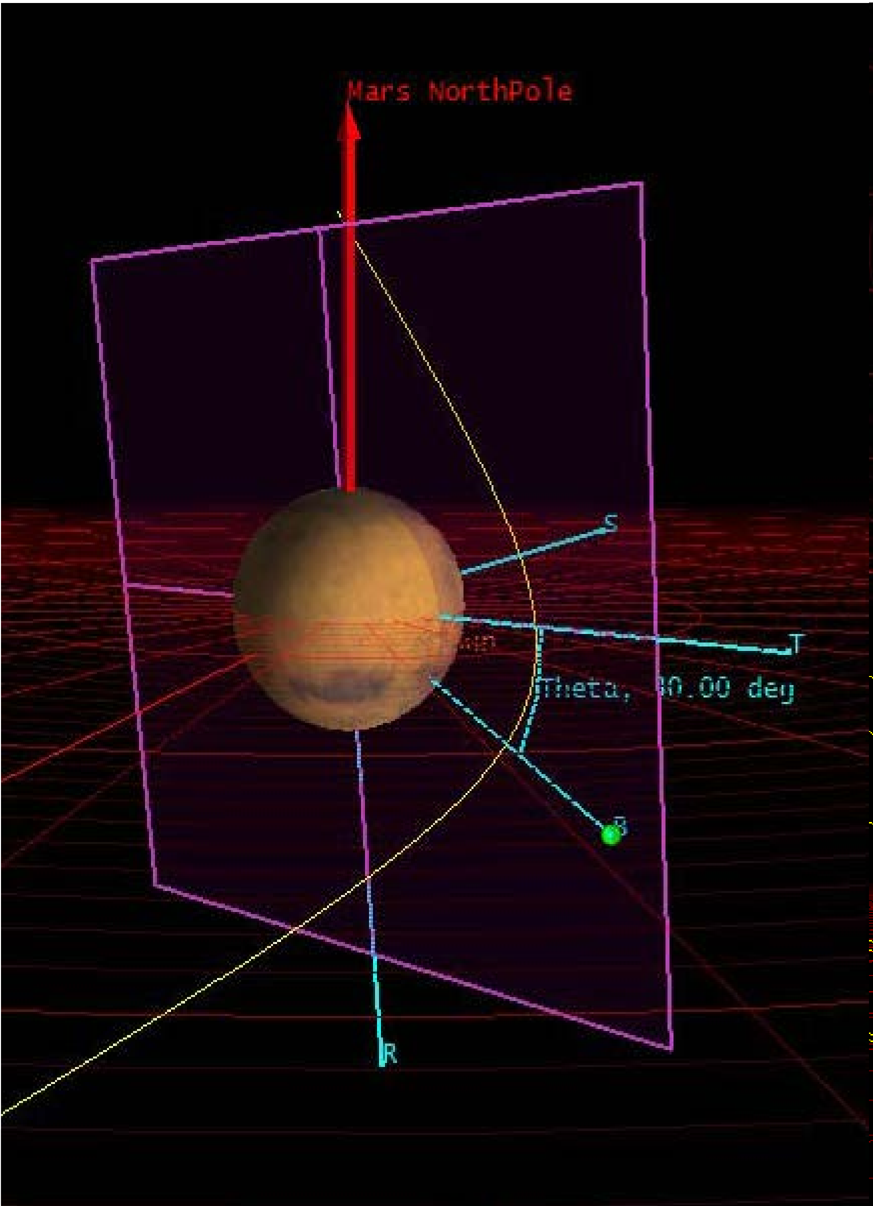
Gravity Assist concept

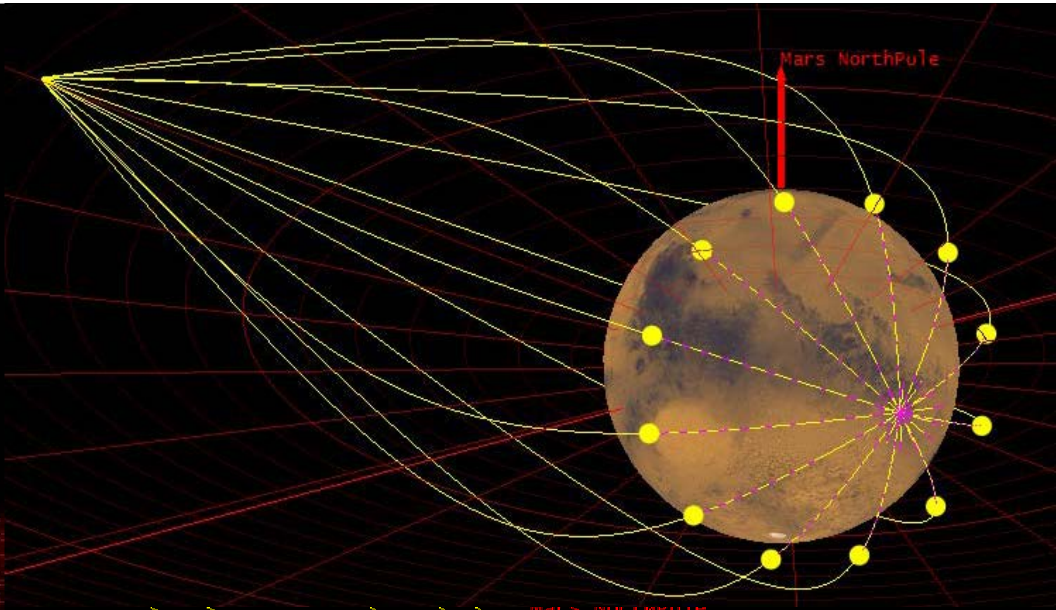


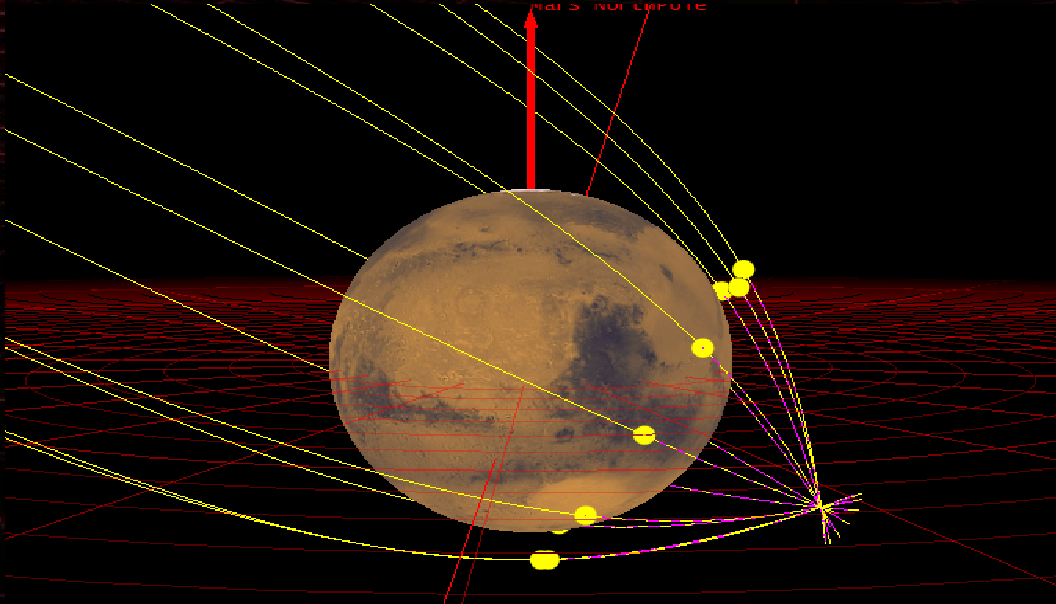




B-Plane

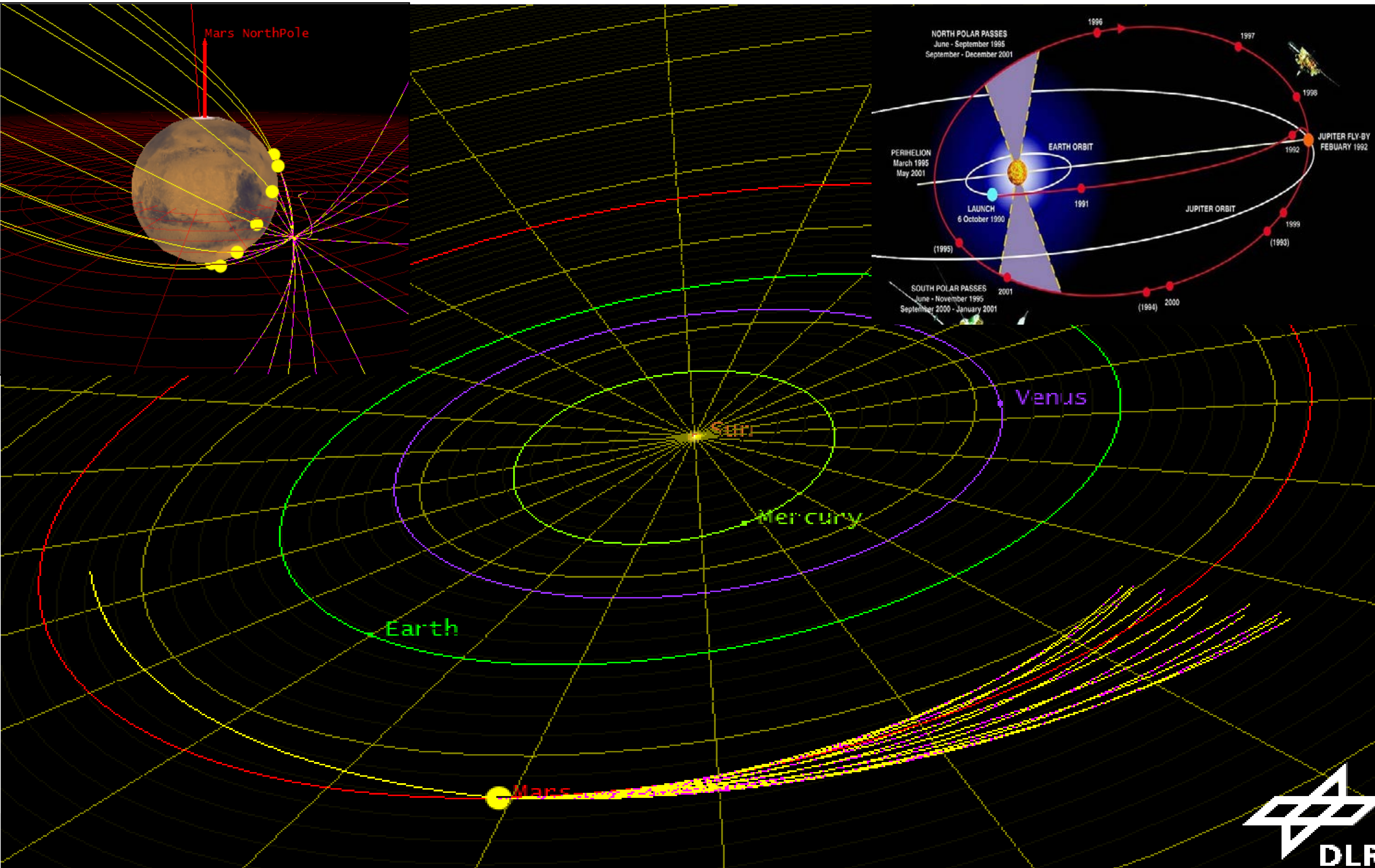




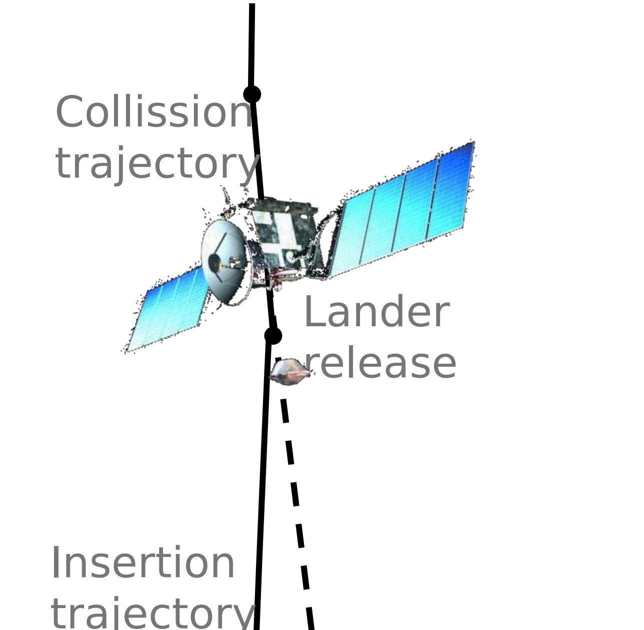


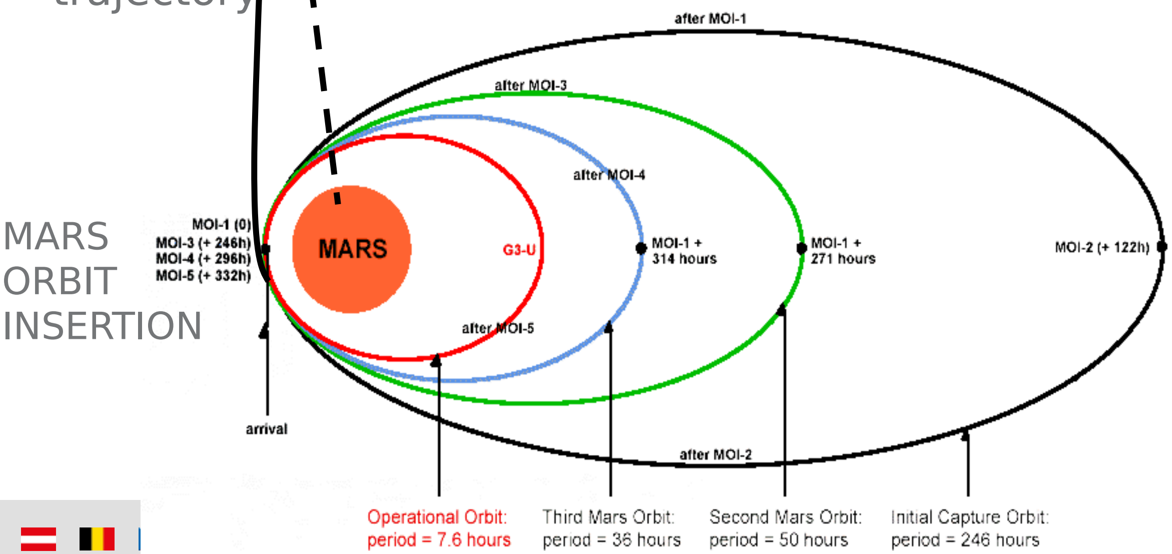
Off-plane swing-bys

Ulysses Polar orbit

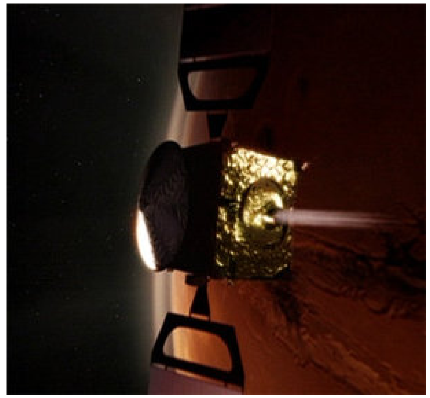


Orbit Insertion

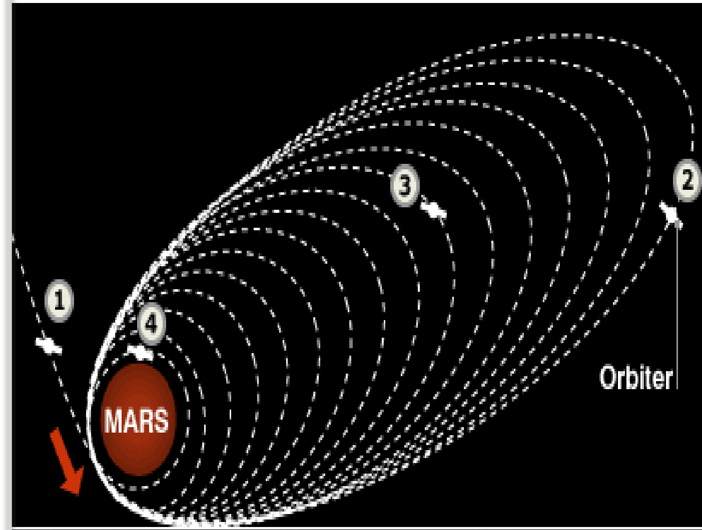




1300kg fuel (TGO)

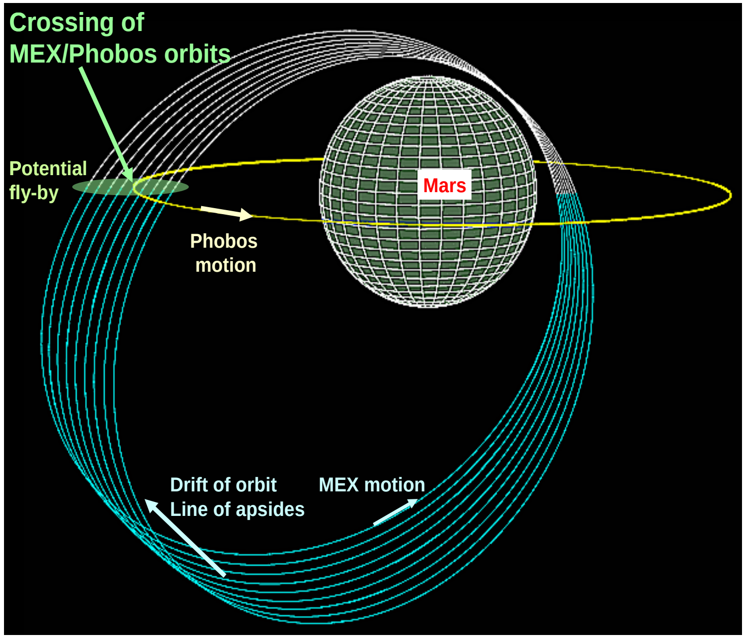


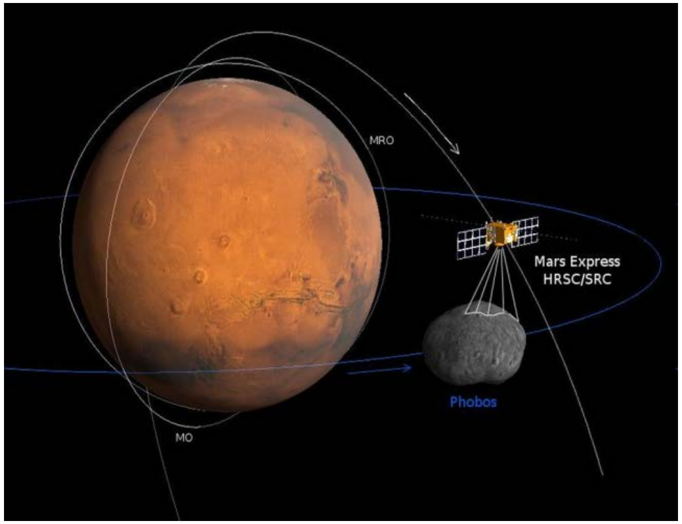
"Aerobraking" for Orbit Circularizatión

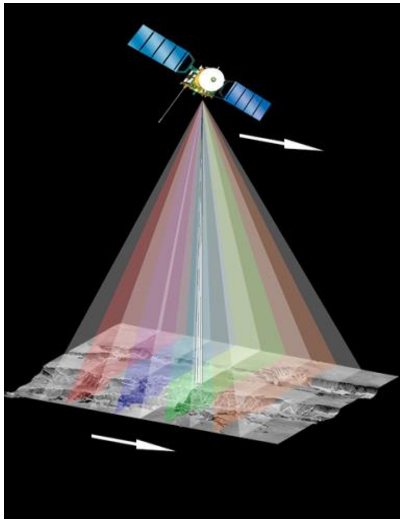


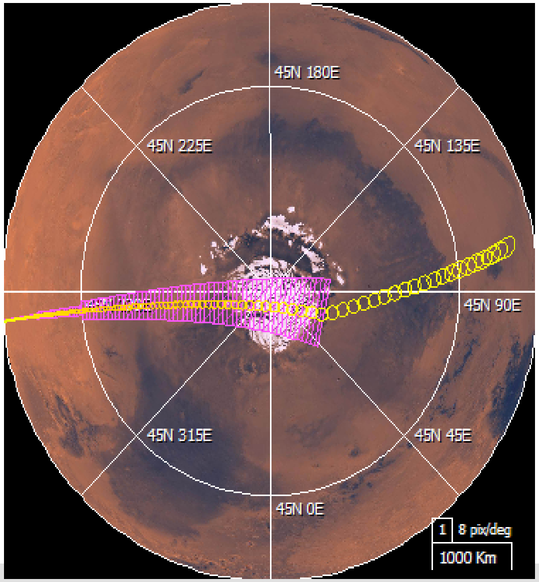
Target orbit selection

- 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

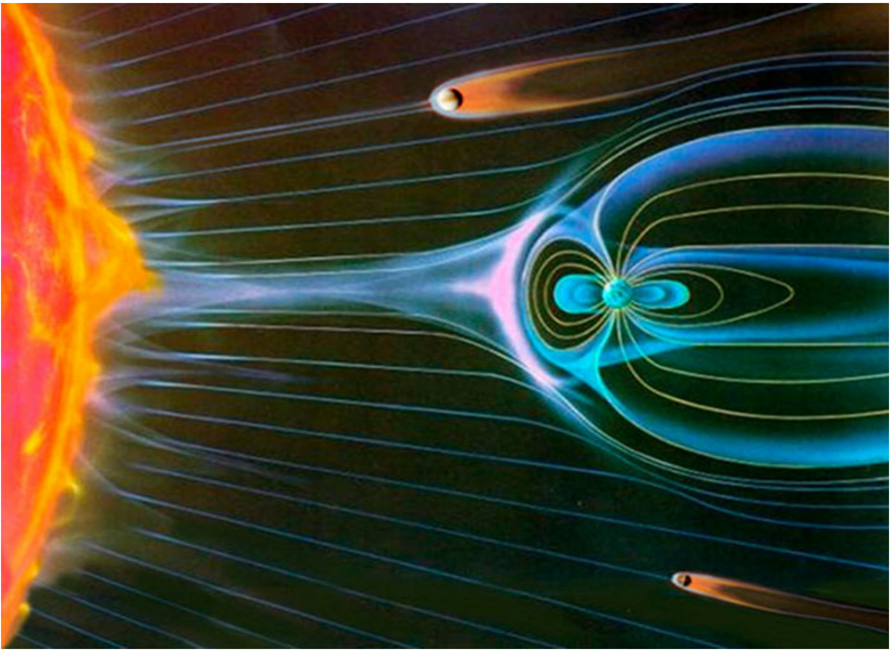








Space Environment



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)

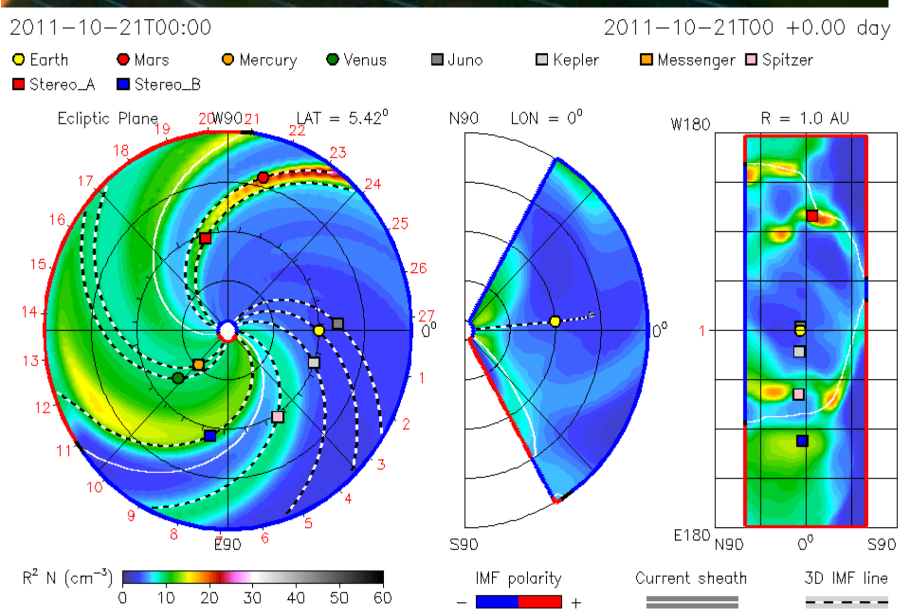
Magnetic Field

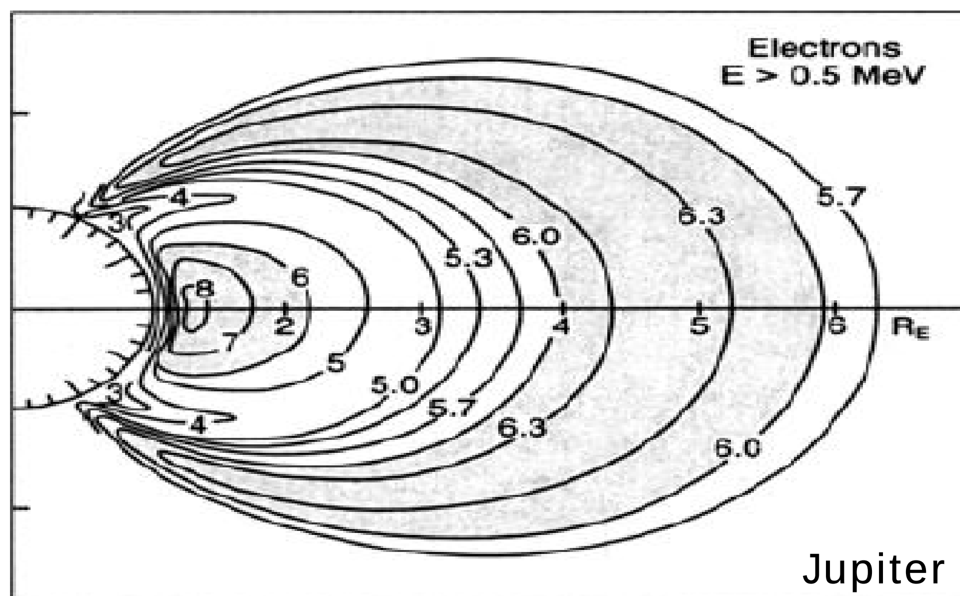
Solar Radiation Pressure

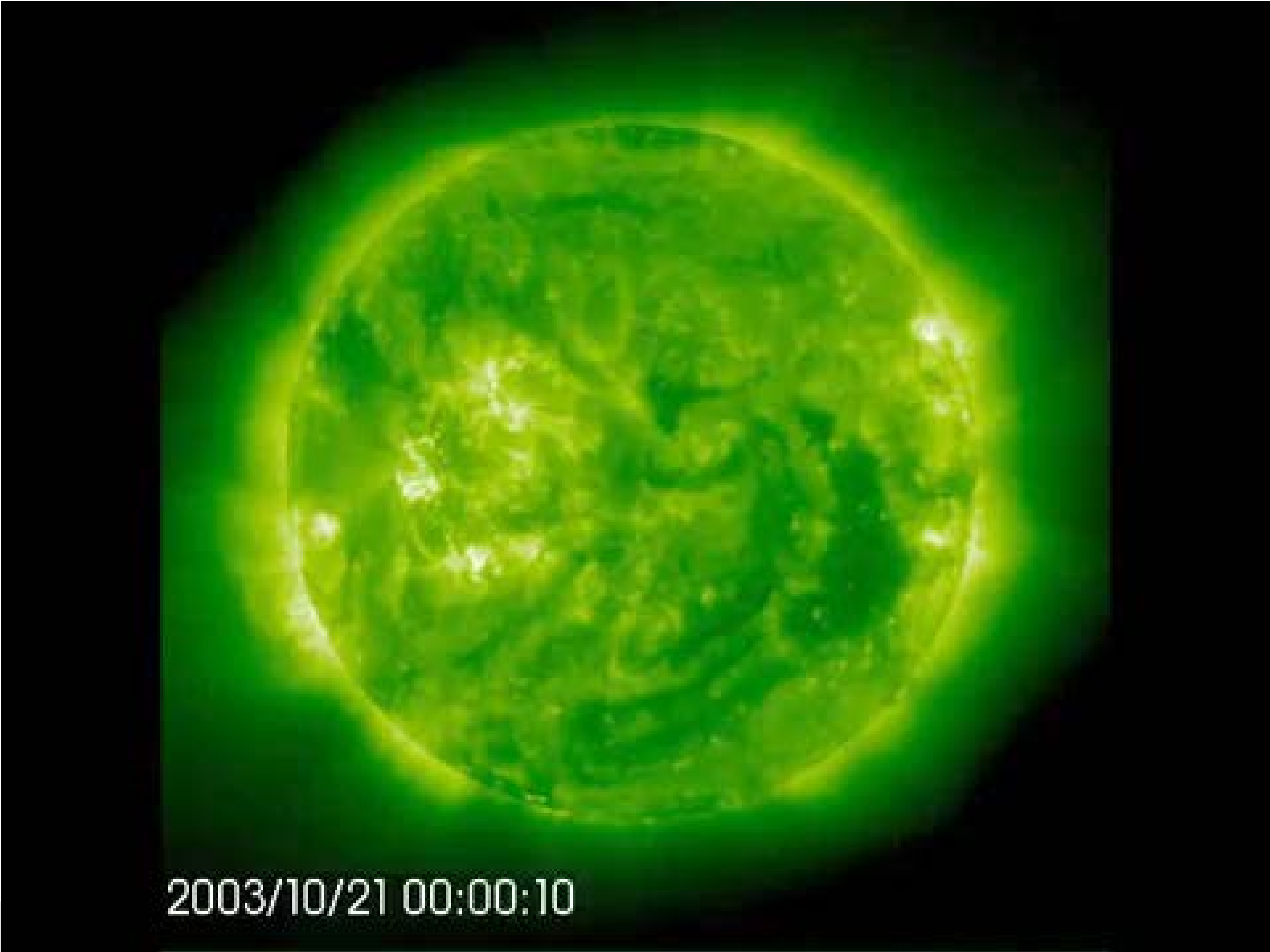
Thermal environment

Vacuum: Atomic Oxygen

Radiation belts of Earth, Jupiter,... electrons, protons



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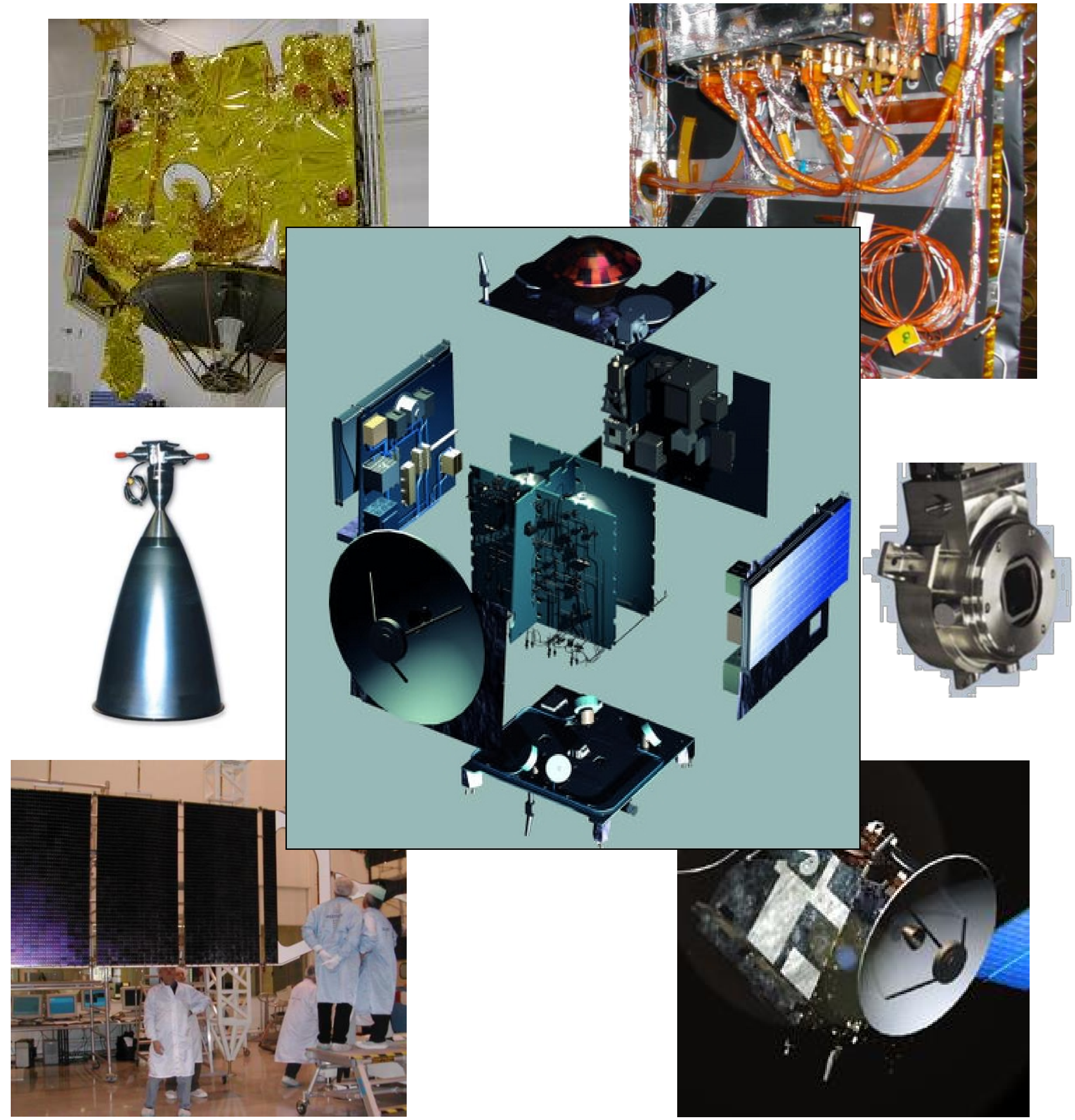
Spacecraft Sub-systems

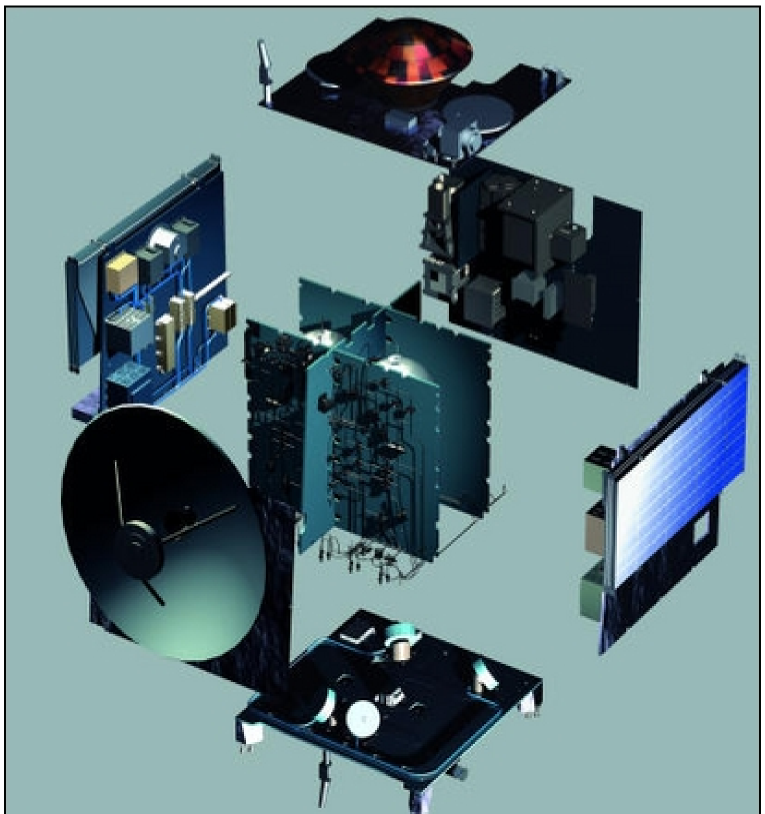
- Structure
- Propulsion
- Orientation
- Power
- On-board Computer
- Communications
- Thermal Control
- Payload



Structures

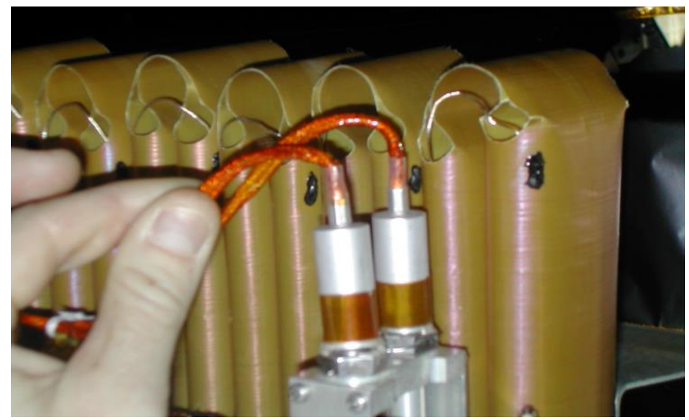
Primary structure

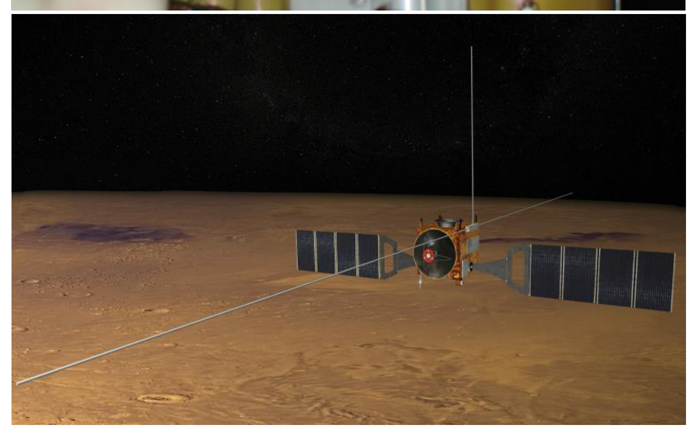
(platform harness)



Secondary Structure

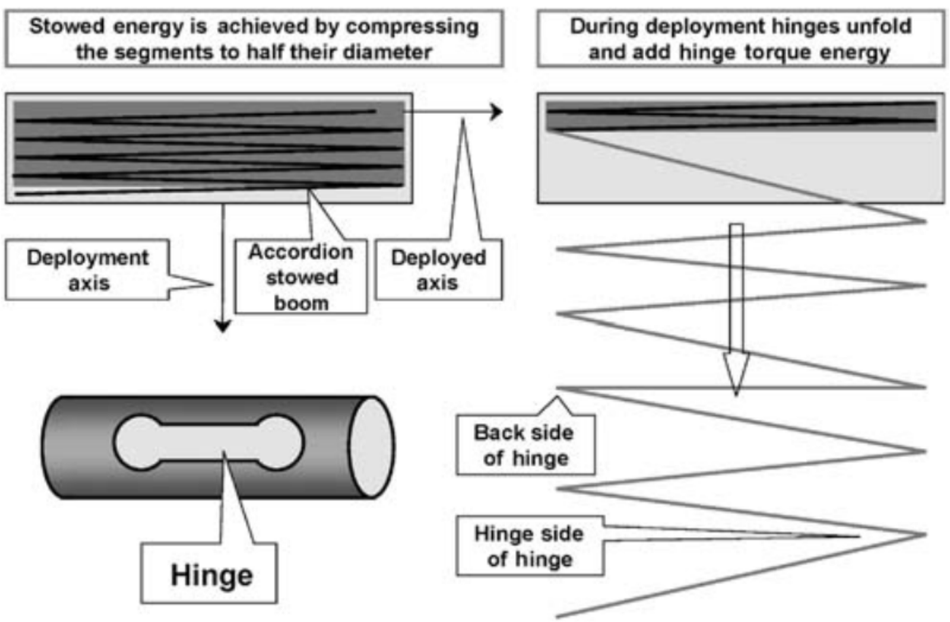
(equipment + mechanisms)

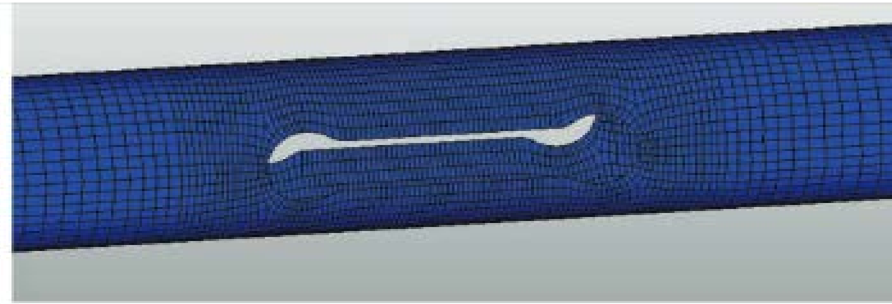




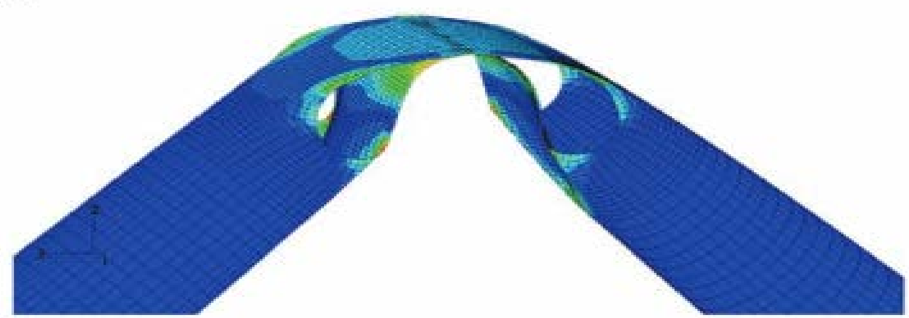
Structure can be a large fraction of the total satellite mass: ~30 %

MARSIS Antenna deployment 2005





a)



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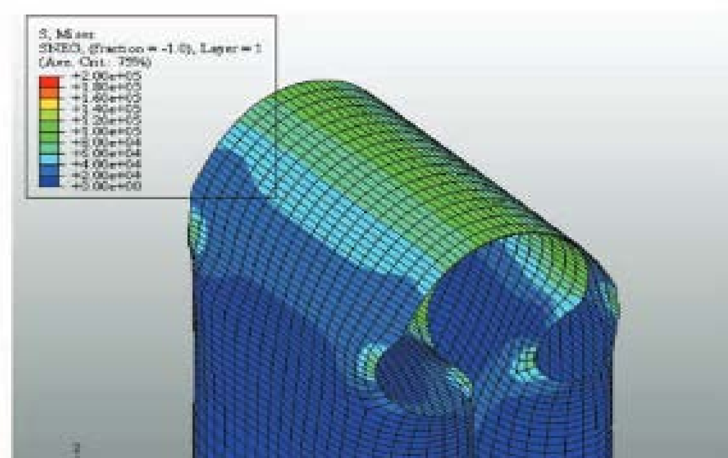
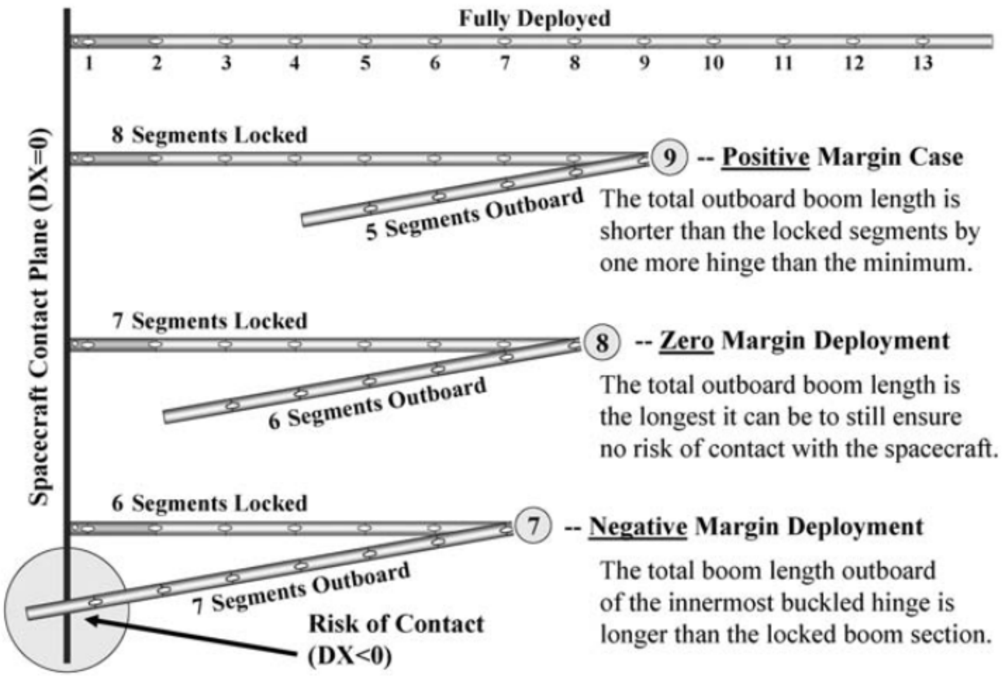
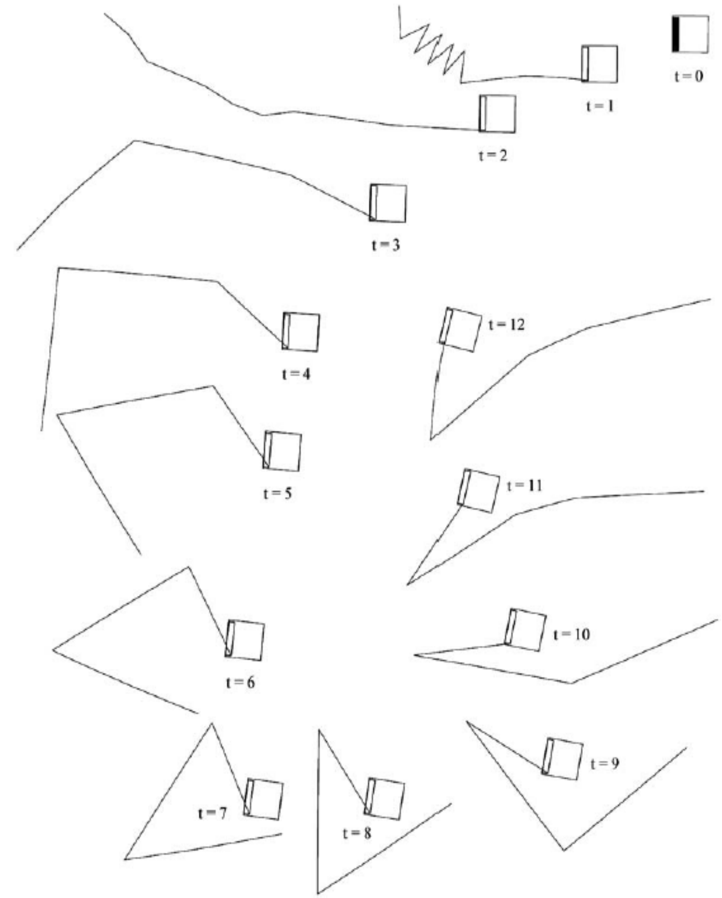
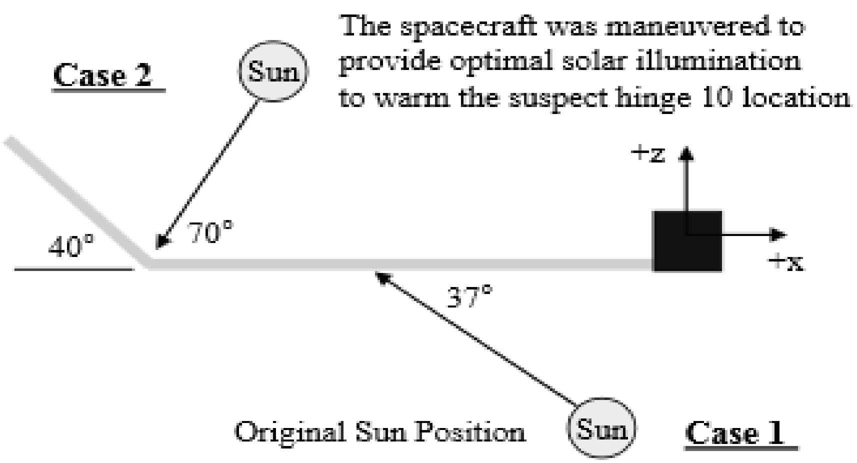
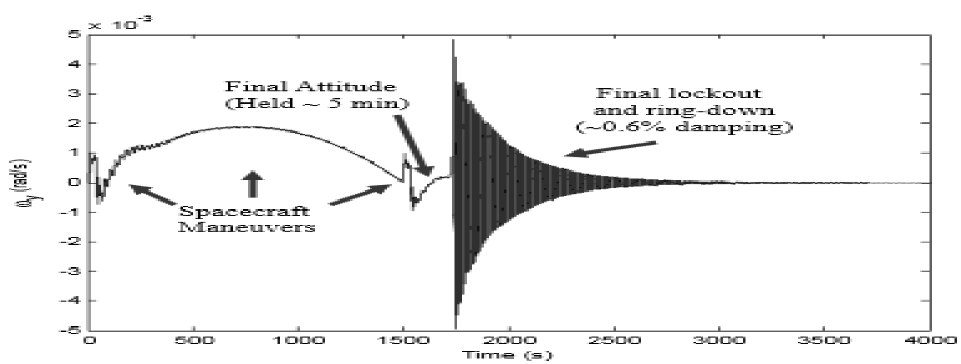


MARSIS Antenna deployment 2005









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 ΔV: "high" I_{sp} (> 300 s), e.g. bipropellant or electric propulsion

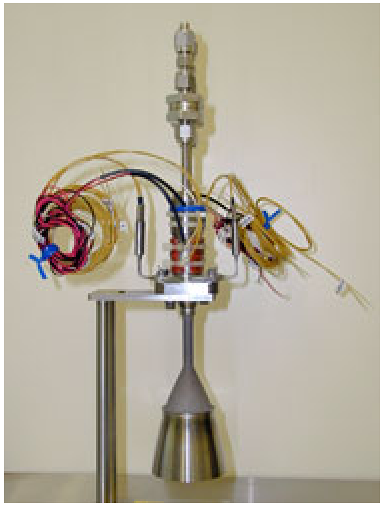
For orbital manoeuvres with low ΔV: "medium" I_{sp} and thrust (~1 N) – e.g. monopropellant - hydrazine

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

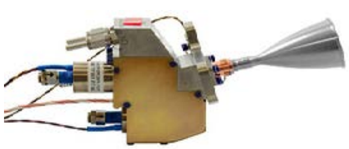
Propulsion
Example (Mars Express)

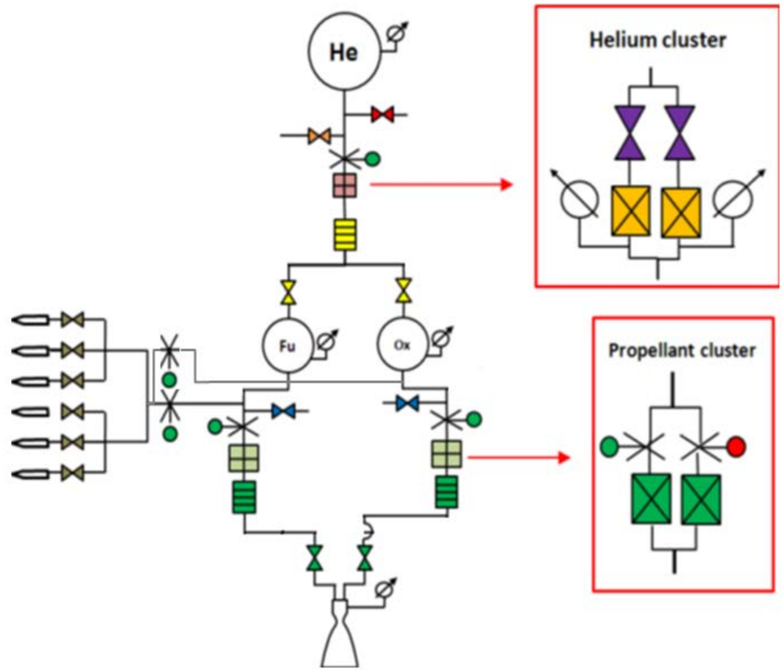
Main Engine for Orbit Insertion

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

Thrusters for Attitude Control

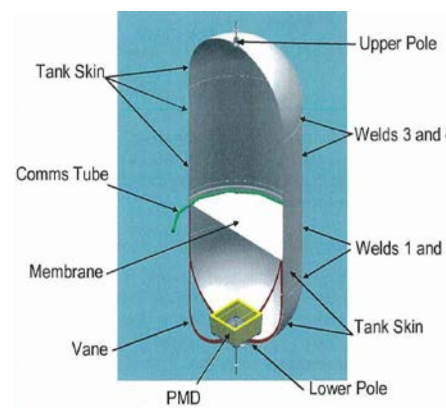
– 8 x 10 Newtons

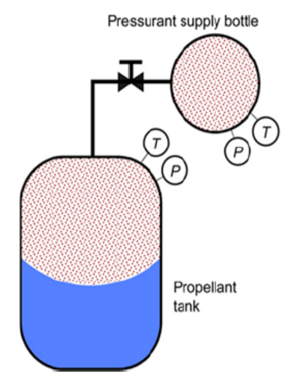




Bi-propellant system

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





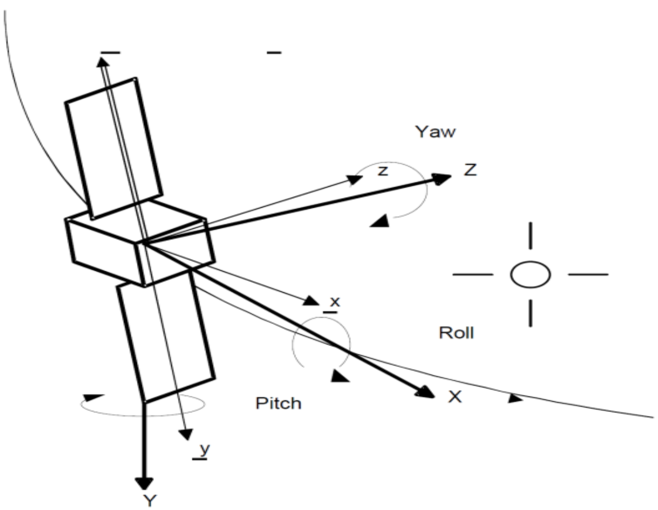
Propulsion
Reaction Control System Thrusters

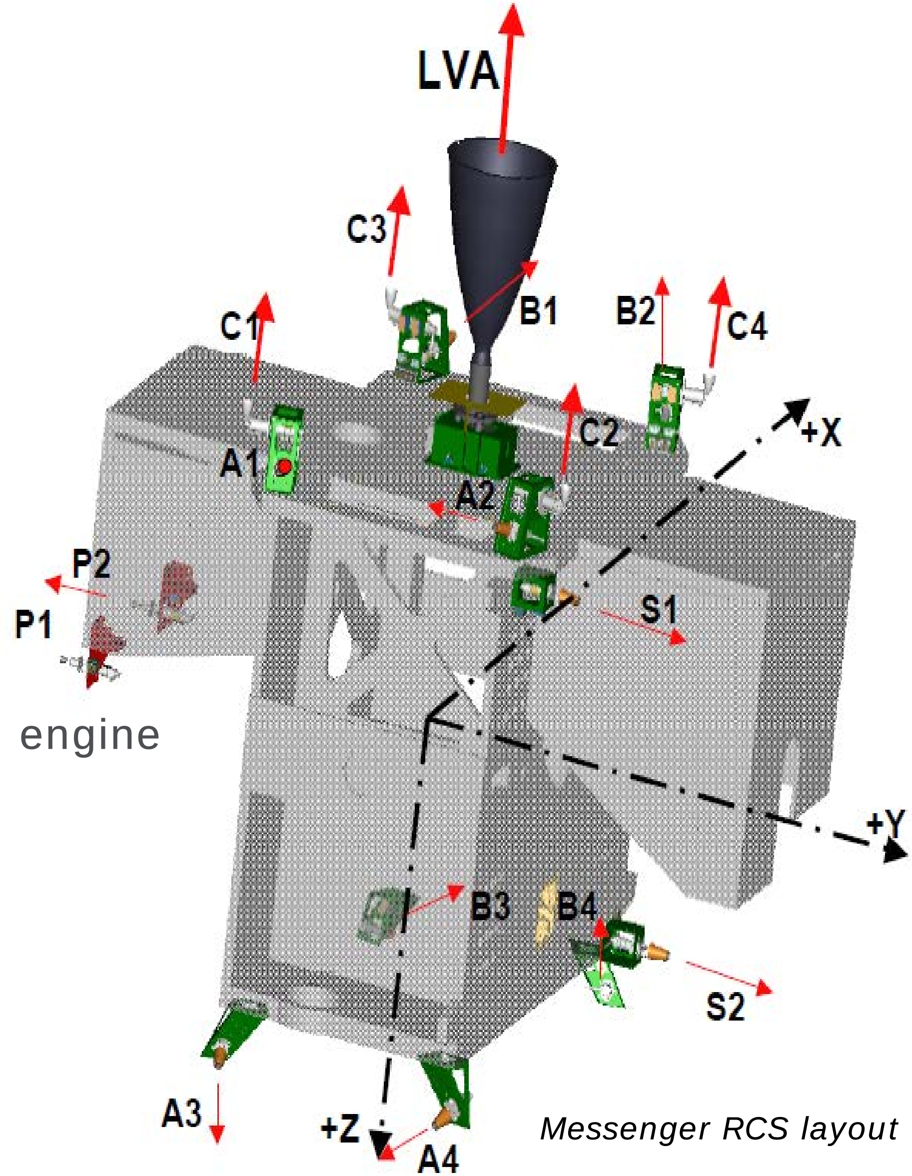
Definition and location of thrusters

Thrusting in any direction in any attitude

Redundancy required

RCS Thrusters could act as backup for main engine





Messenger RCS layout

Future: Solar Sail

Force on solar sail $F = p \cdot c \cdot A \cdot \cos\vartheta$

p ~ 4.6 µN / R^2

c = 1 ideal absorption ~ 2 ideal reflection

Sail Requirements:

- Large area
- Low mass – few µm!

Container + Deployment mechanism (~100 m booms)

Contingency (SAFE mode) recovery strategy needed

Limitations for

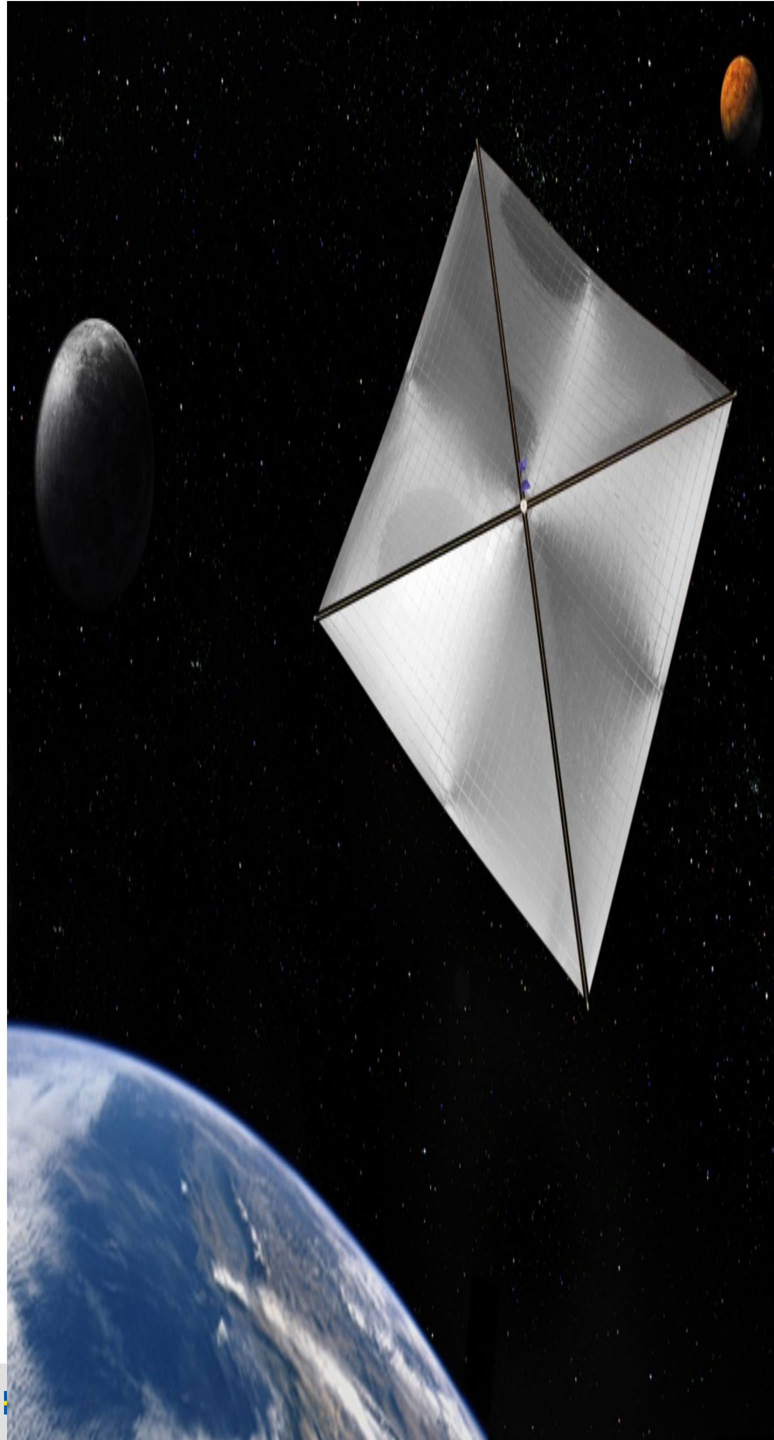
- Communications
- Attitude control (Large angular inertia – high torque)
- Solar power generation

JAXA IKAROS mission (2010)

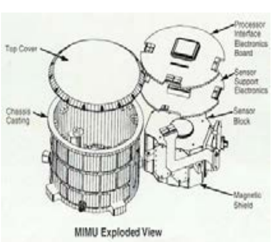
7 µm 14x14m Sail (200m^2) → 100m/s after 6 months

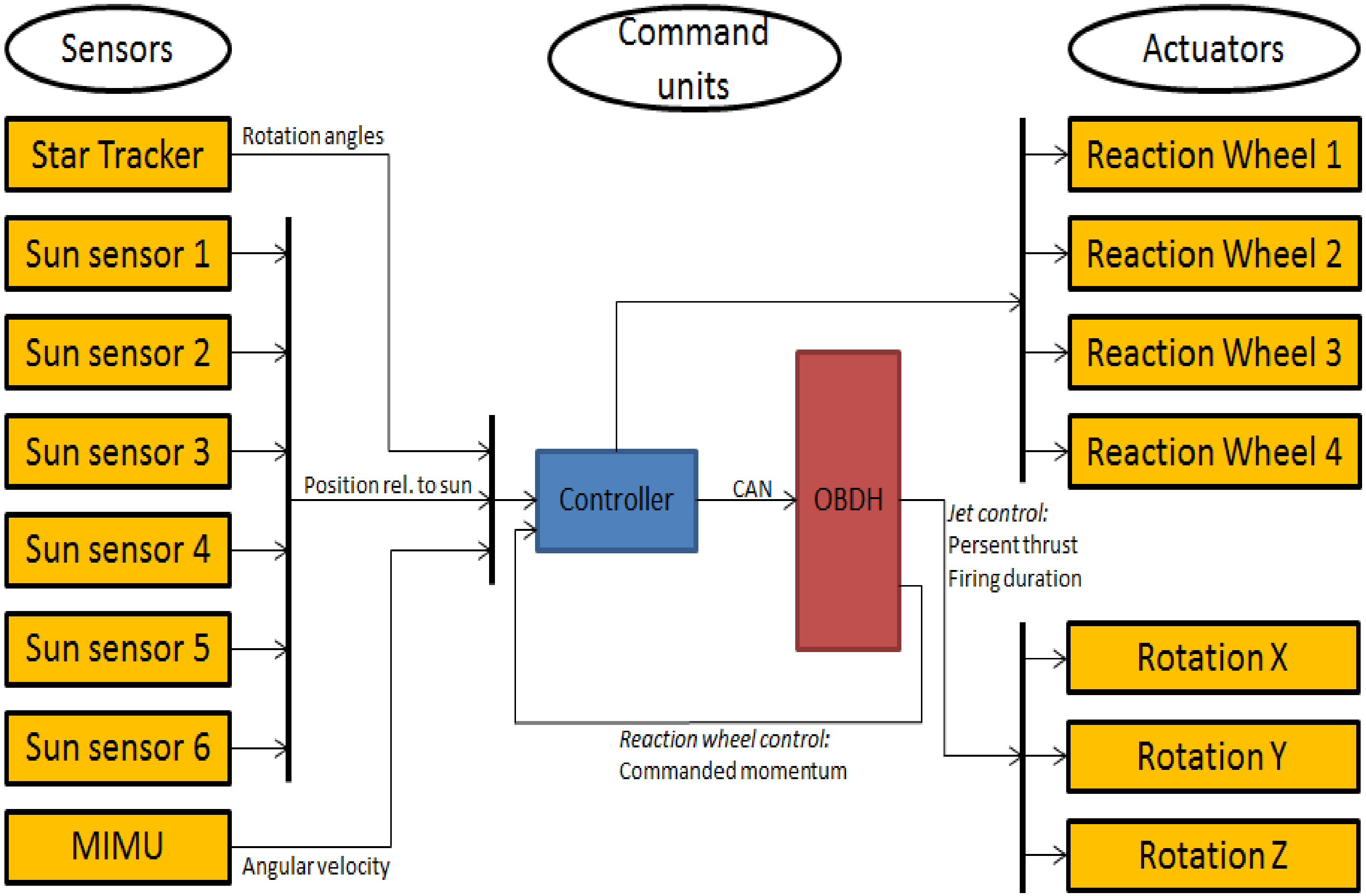
NASA NanoSail-D2 (2010)

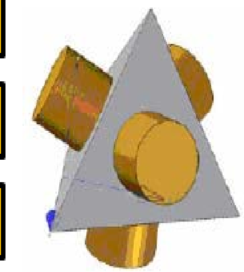
4Kg CubeSat (30x10x10 cm): 10m^2 sail for de-orbit



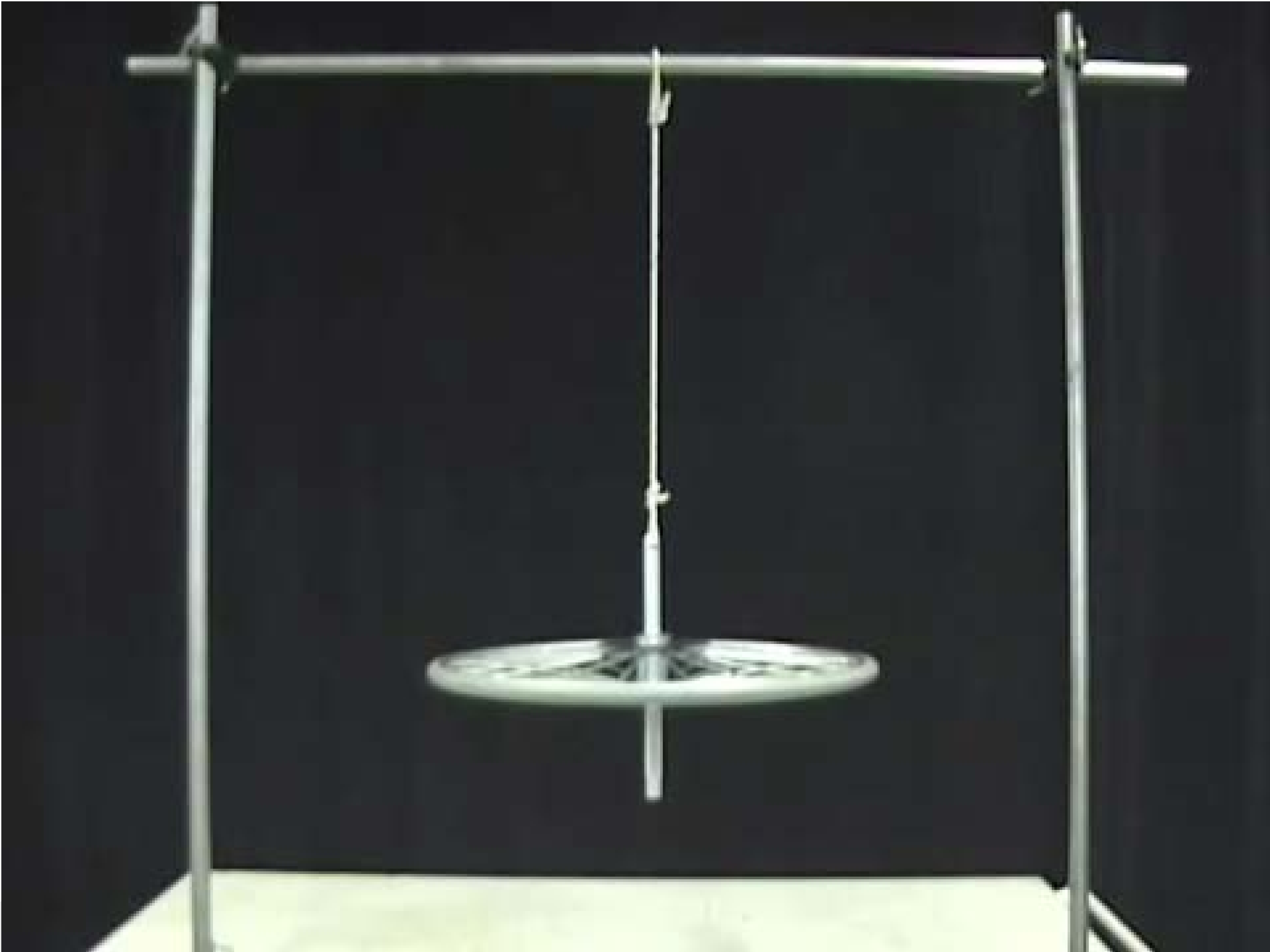
GNC Guidance and Navigation Control
(AOCS Attitude and Orbit Control System)





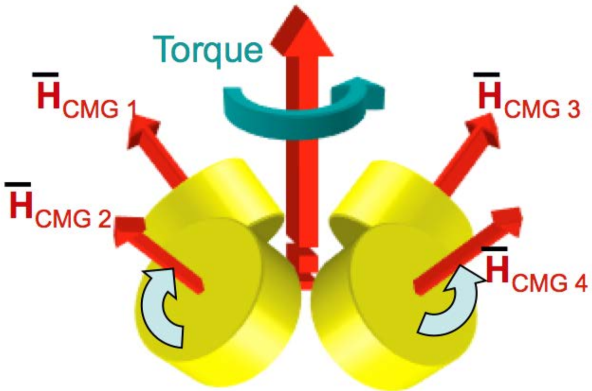


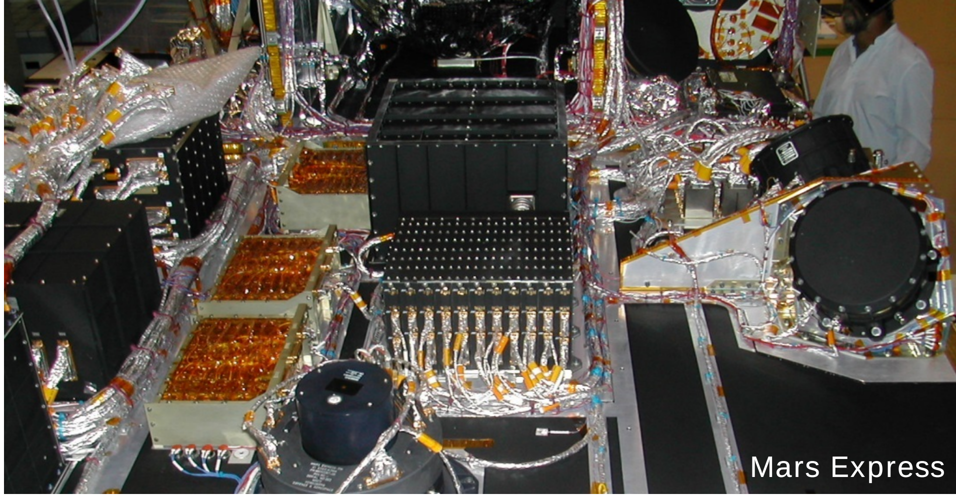
Fully redundant system , 3-axis stabilized, $\Delta\varphi < 0.05$deg, $\omega < 0.15$ deg / s



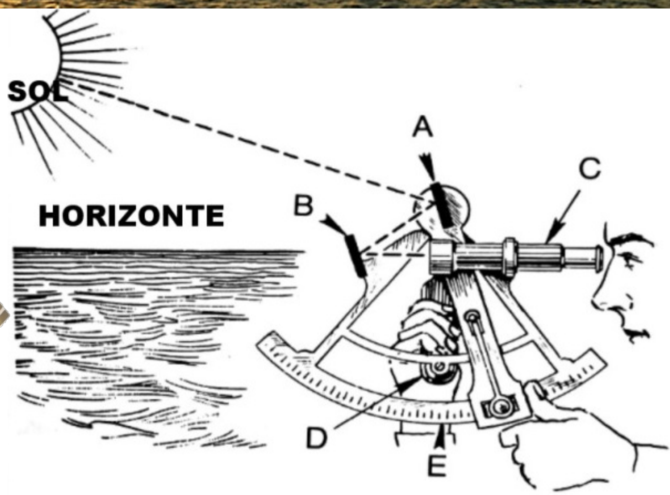
Actuators: Reaction wheels

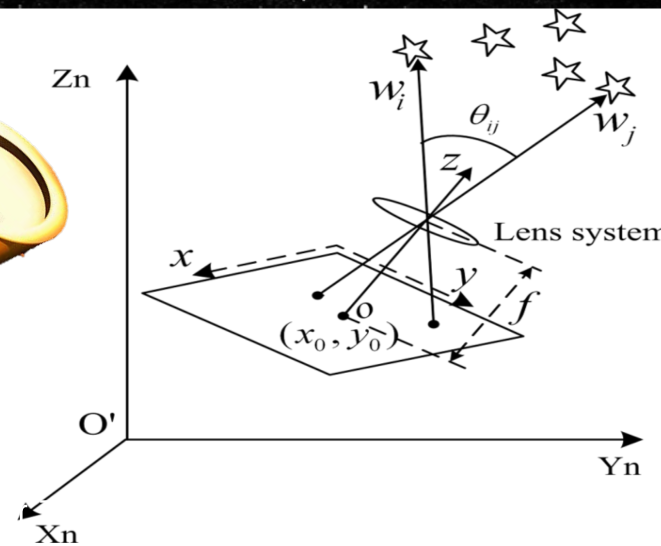
ExoMars TGO Reaction Wheels:
- 4 wheels
- 5 Kilograms each wheel
- 23Nms Angular Momentum
- 7,500 RPM Spin speed:
- 35x12cm centimeters





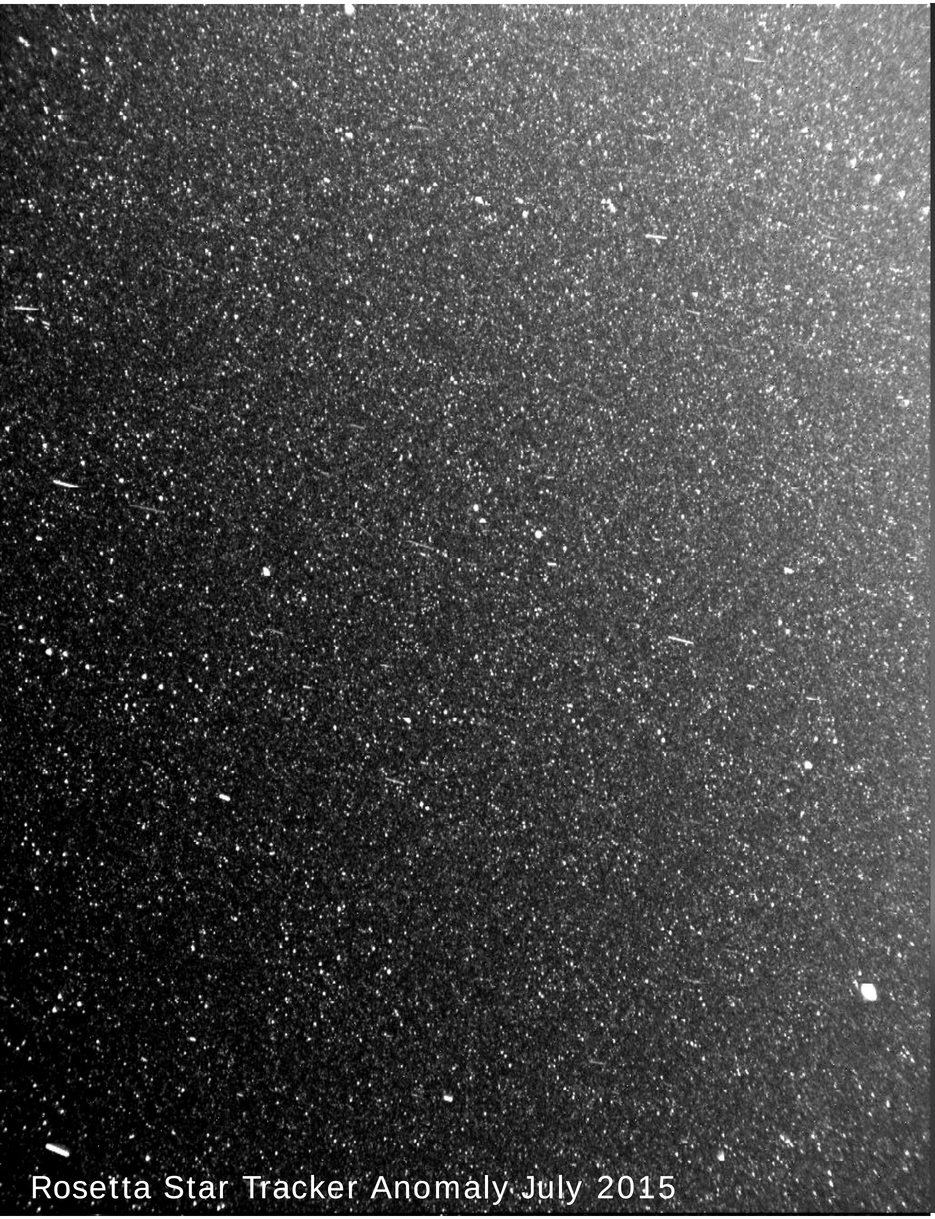
Inertial Sensors: Star Trackers





Inertial Sensors: Star Tracker Examples

Star Tracker Rosetta Anomaly 2015



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. => ~<u>500 Watts</u>

(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

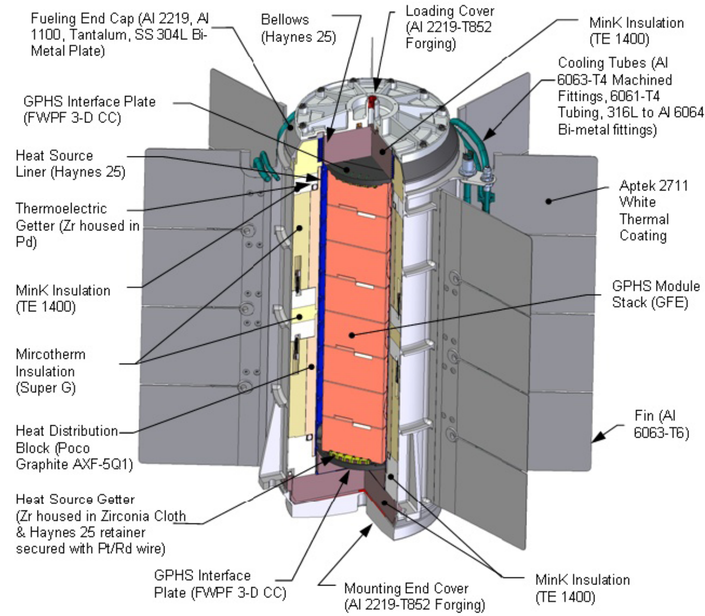
(necessary for missions beyond Jupiter)

Electrical Power Subsystem (EPS)
Nuclear Sources

Uses 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		

(Plutonium heat flux ~0.5 W/g)



Degradation

Half life: Pu (88 yrs), ^{241}Am (433 yrs)

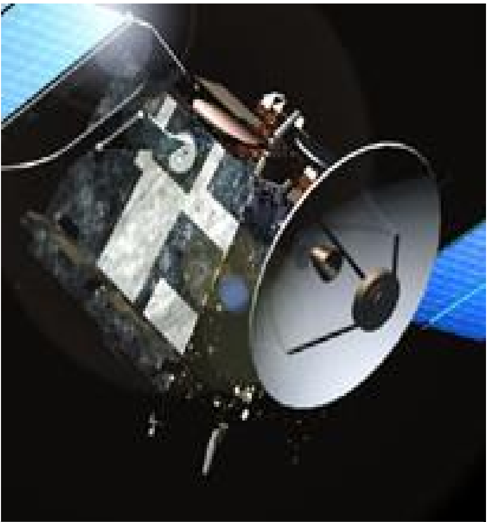
Themo-electric element: ~0.8% /yr

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

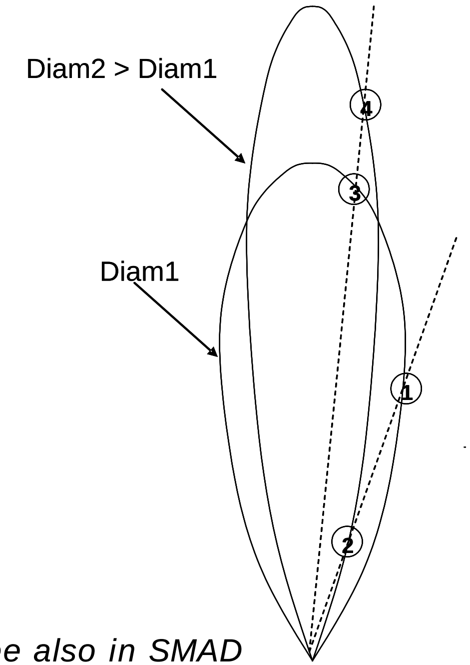
NASA 35/70m diameter: Madrid, Australia, USA

HGA Performance

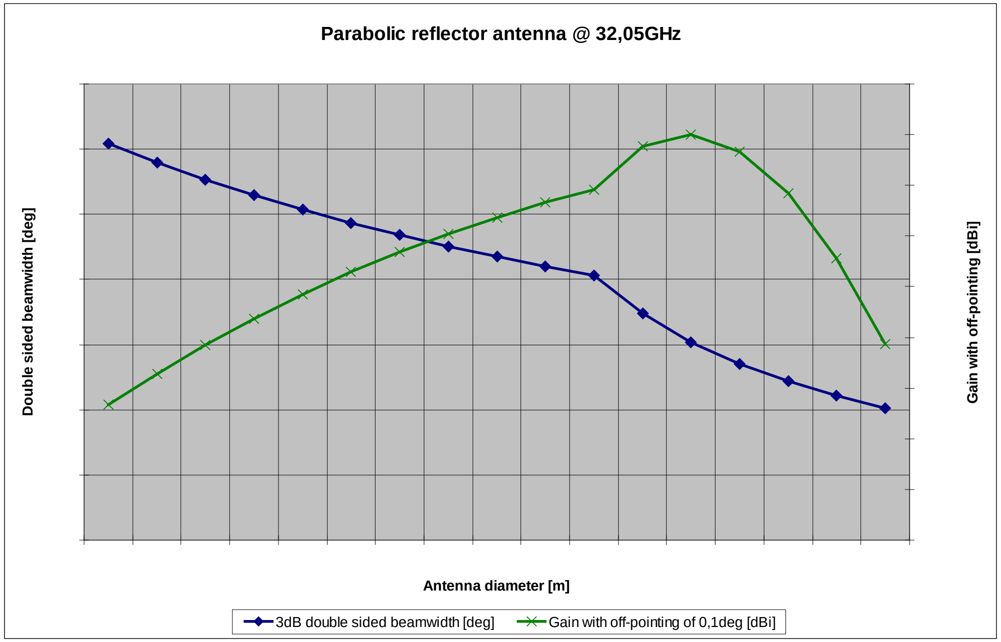
High Gain Antenna (HGA) versus pointing performance

Optimum antenna diameter for known AOCS off-pointing

Further iteration to be done once AOCS performance is known



see also in SMAD



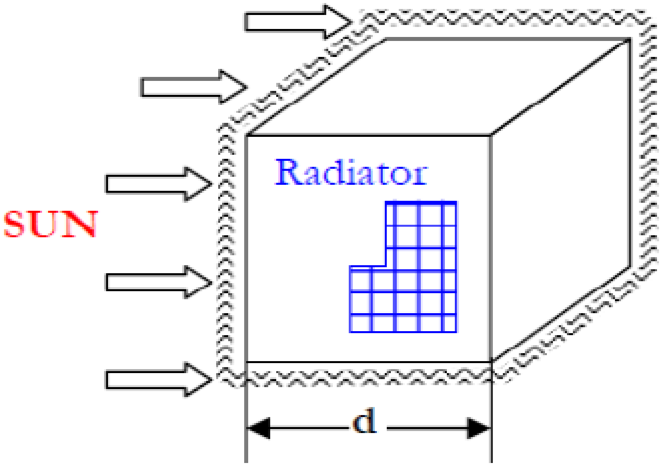
Thermal Control Subsystem

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)

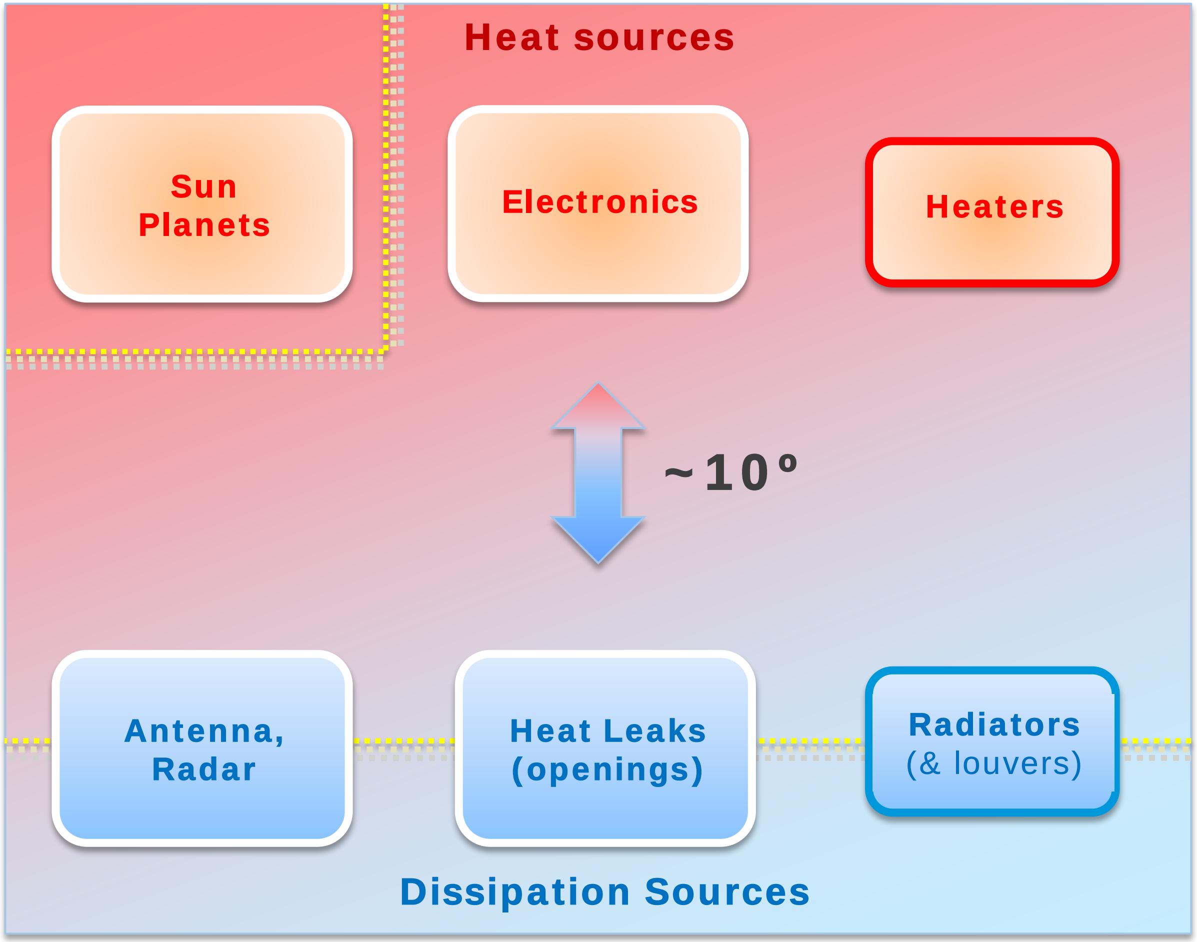
Thermal Control

External Temperatures -100 ~ +150ºC

Multi-Layer Insulator to avoid illumination and dissipation

Most electrical power is converted into heat

Radiators + Heaters + pipes...



On Board Data Handling (OBDH)

Data Management System

Telecommand distribution

Telemetry data

Events, housekeeping, …

Control and Data Managament Units

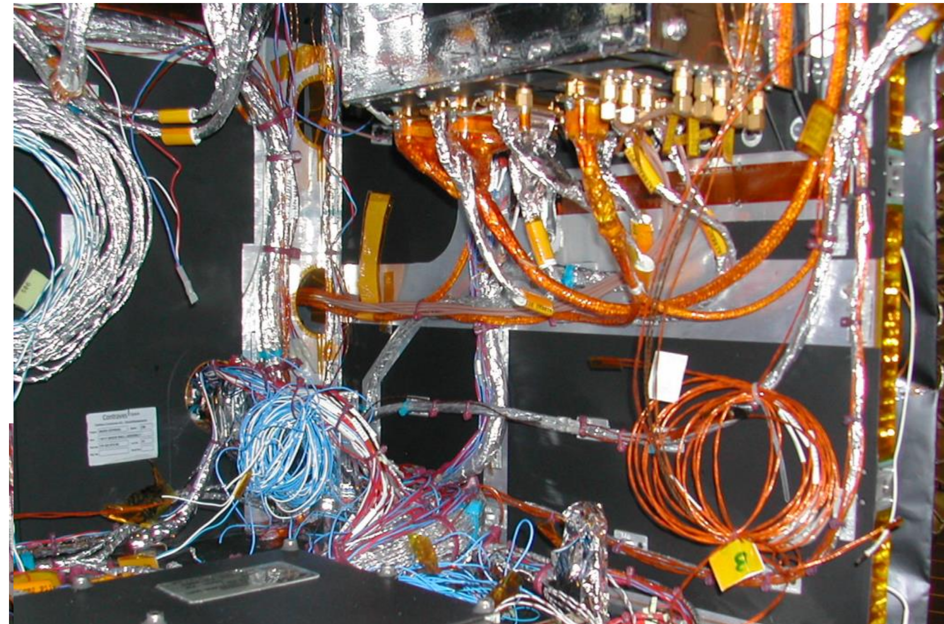
4 Processor Modules (2 DMS + 2 AOCS)

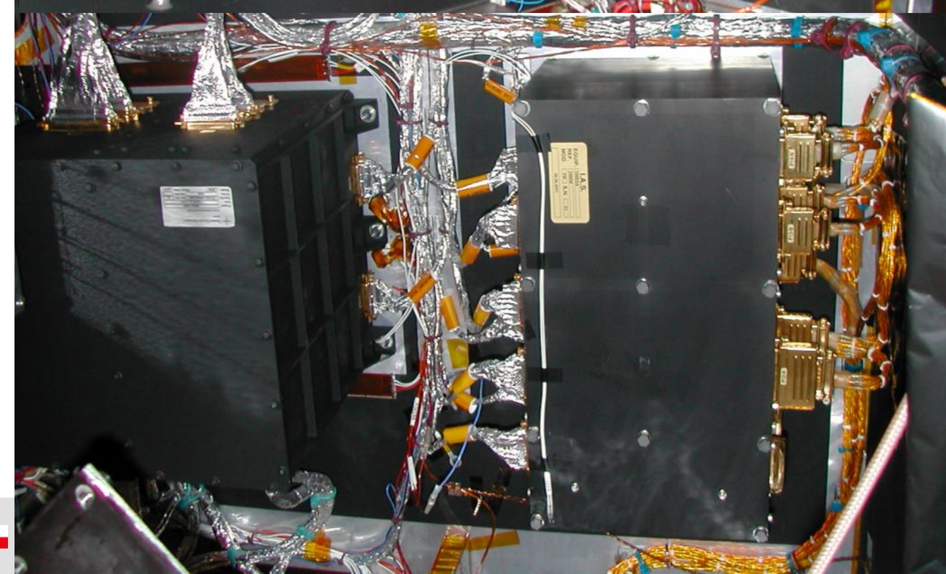
Bus Architecture + High speed Link

Solid State Mass Memory

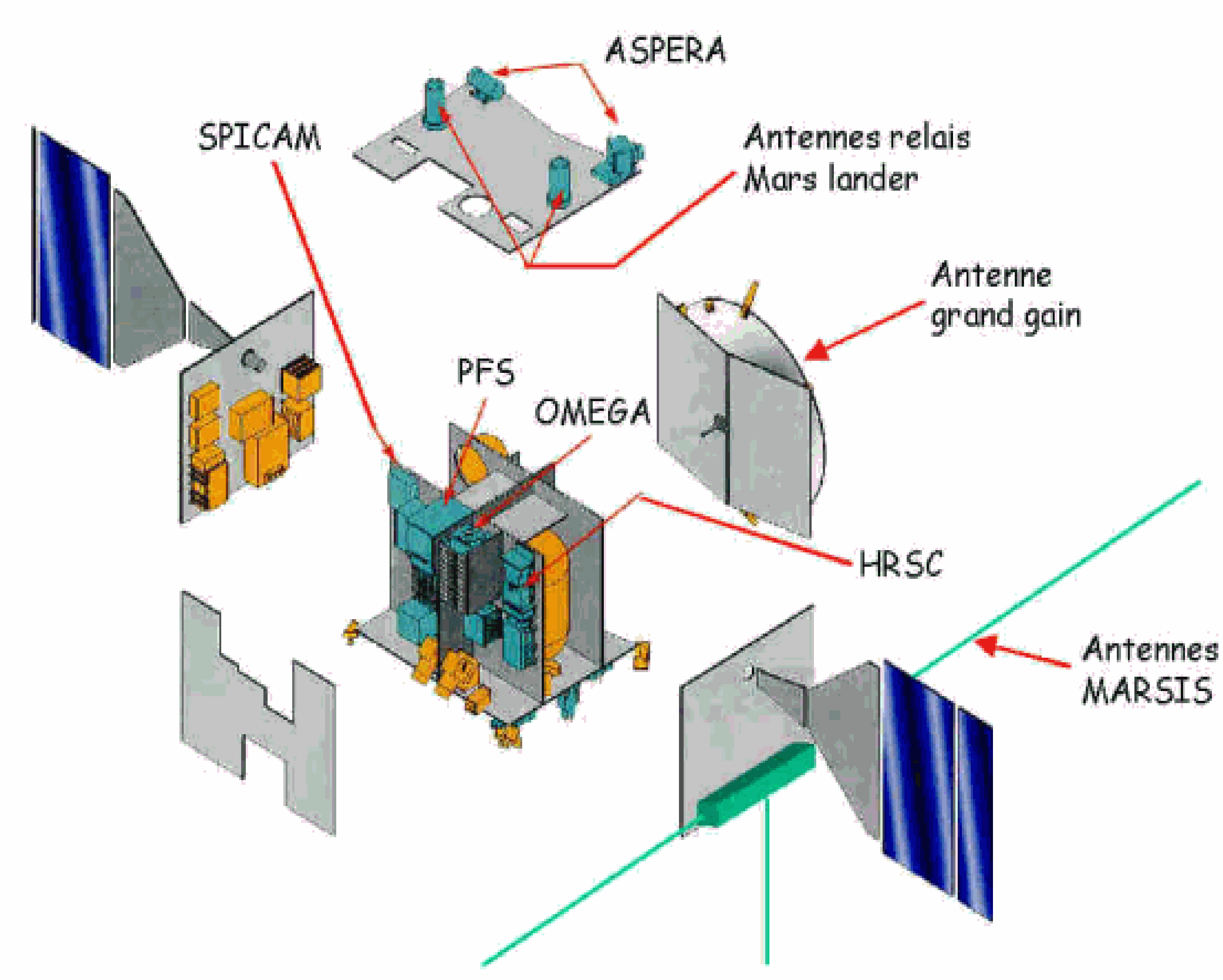
Payload Data Handling Unit

(MEX: 8 Gbit → EXM: 1024Gbit)





Payload Subsystems



System Summary

Mission profile & lifetime

Launcher: launch mass, fairing

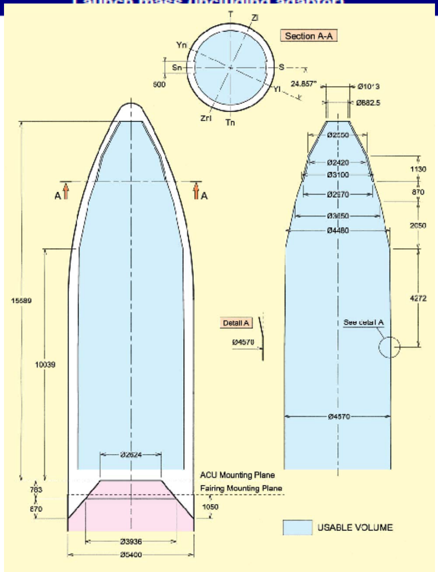
Budgets: Mass, Power

Total system margin

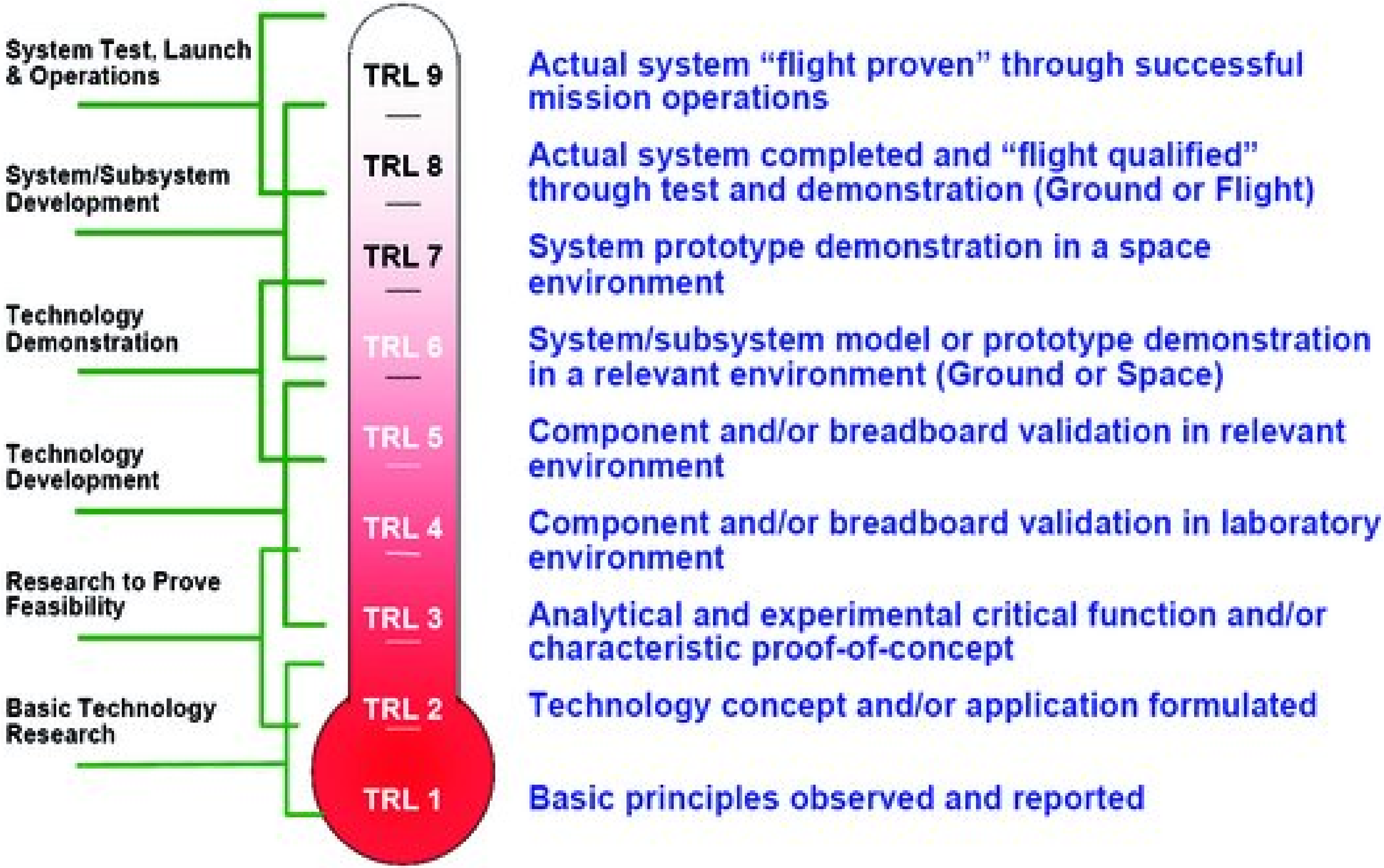
- Equipment 5~20% (based on TRL)
- Total System Level 20%

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	



Technology Readiness Levels (TRL)



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 …

Cosmic Vision Science Program Missions

L-missions (L1 JUICE , L2 ATHENA, L3 Gravitational Waves Observatory)

- Cost to ESA of around 2 annual budgets (**1000 M€**)
- European led flagships with <20% international contributions
- May need technology development

M-missions (M1 Solar Orbiter, M2 Euclid, M3 Plato, M4 ARIEL/THOR/XIPE?, ...)

- Cost to ESA of around one annual budget (**500 M€**)
- ESA led or contribution to international collaboration.
- No technology development

S-missions (S1 CHEOPS, S2 SMILE, ...)

- Cost to ESA of 0.1 annual budgets (**50 M€**)
- National agencies play a leading role
- No technology development

O-missions

- Missions of opportunity, led by other agencies, small contributions.

GRACIAS

¿preguntas?