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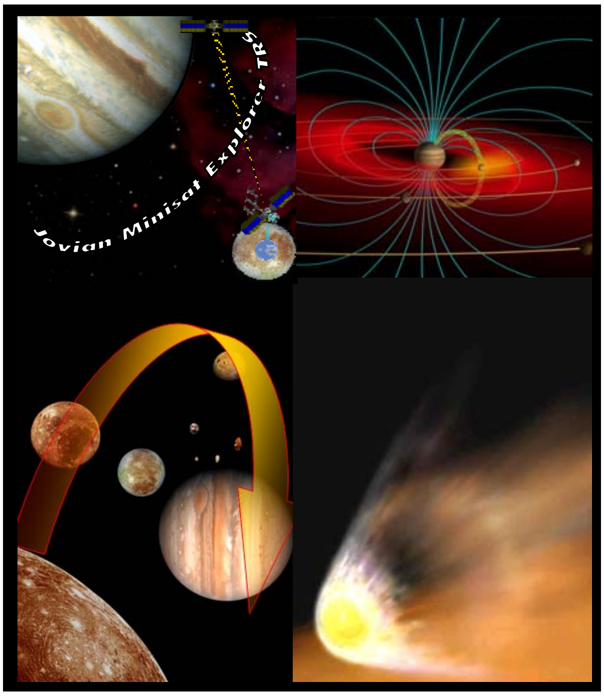
DOCUMENT

Overview of the ESA Jovian Technology Reference Studies



prepared by/*préparé par* Alessandro Atzei, Arno Wielders, Anamarija Stankov, Peter Falkner

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European Space Agency
Agence spatiale européenne

ESTEC
Keplerlaan 1 - 2201 AZ Noordwijk - The Netherlands
Tel. (31) 71 5656565 - Fax (31) 71 5656040

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INTRODUCTION

Technology reference Studies

In 2003 ESA's Science Payload & Advanced Concepts Office started a combination of activities that go by the name "Technology Reference Studies" (TRS). The goal of the TRS is to identify and to start the development of critical technologies that will be required for future scientific missions. This is done through the study of several challenging and scientifically relevant missions, which are not part of the current ESA science programme, and focus on the medium term enabling technology requirements.

The TRS's share the same baseline approach: the use of one or more small spacecraft using a suite of highly miniaturised and integrated strawman payloads, with strongly reduced resource requirements. The purpose is to achieve the science objectives with a cost efficient exploration when compared to a large "one-shot" mission.

Purpose of this document

This document has been prepared to give a concise overview of the studies that have been performed in the framework of the Jovian related Technology Reference Studies. The goal of these studies is the identification of technologies that are required to enable possible low resource missions to the Jovian System. These activities are subdivided in three main topics:

- The Jovian Minisat Explorer (JME): The exploration of Europa and the Jovian System
- The Jupiter Entry Probe (JEP): In situ exploration of the Jovian atmosphere up to 100 bar
- The Jovian System Explorer (JSE): Study of the Jovian magnetosphere and the Jovian System

The main effort in the past years has been focussed on the Jovian Minisat Explorer, to understand the challenges of a remote sensing mission to Europa. The study included two industrial studies (a solar powered and a radioisotope powered spacecraft option) as well as internal reviews using the Agency's Concurrent Design Facility (CDF) that resulted in further evolution of the design. Side studies also looked into the feasibility of a compact ice penetrating radar, highly integrated instrument and avionics suites, low intensity low temperature solar cells in high radiation environment, radiation models and impactors.

The Jovian Entry Probe study, performed by a dedicated CDF team, was initiated to investigate the critical technologies and design issues related to the design of a ballistic Jovian entry probe, with the aim of performing atmospheric measurements during descent and to survive to an ambient atmospheric pressure up to 100 bar.

The third study, the Jovian System Explorer focuses on a cost-efficient and technologically feasible mission architecture for a multi-spacecraft exploration of the Jovian magnetosphere and atmosphere, while providing a preliminary assessment of the logistics and enabling technology development.

Finally an overview is provided of ongoing and planned studies. The following figure provides an overview of the current Jovian activities.

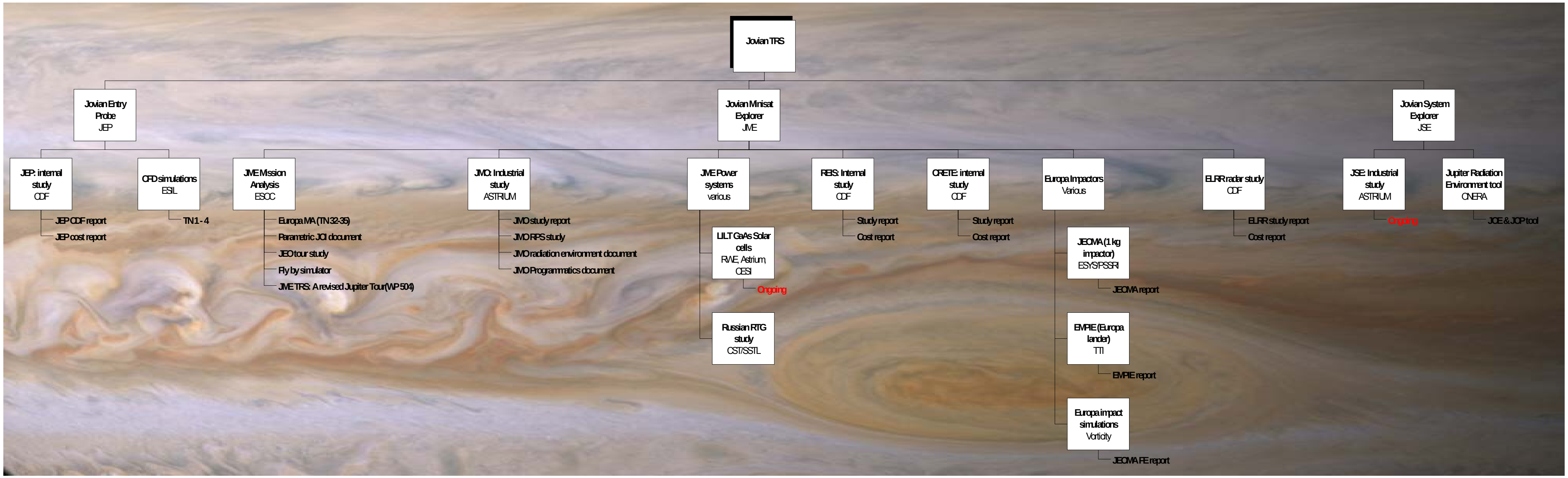

Figure 1: Overview of the Jovian TRS related activities

STUDYING THE JOVIAN SYSTEM

The Jovian System is often compared to a miniature solar system as a result of its dynamism, the massive amount of emitted energy, the huge magnetosphere and its large number of satellites. This combination makes the Jovian System a very interesting destination for scientific missions. In particular Jupiter, its four Galilean moons and the magnetosphere can be considered as high priority targets for future exploration. This is also reflected in the themes identified for the Cosmic Vision 2015-2025: The Jovian System fits well in two of the four themes: The exploration of Europa and mainly the search for its ocean addresses Theme 1 (What are the conditions for planet formation and the emergence of life?), while the exploration of the Jovian System fits well into Theme 2 (How does the solar system work?).

This is also expressed in the following excerpt from ref.1:

A multi-disciplinary investigation of the Jupiter system comprising observations to constrain the Jovian formation mechanism investigations of Europa's physical state and its ability to support life studies of other processes which have bearing on the evolution of the system is a logical next step in our exploration of our Solar System. It complements studies of extra-solar planet formation and astrobiology while at the same time engaging the existing strong communities in planetary and space sciences.

Table 1: Jupiter's main properties [adapted from 2]

Item	Unit	value
Mass	[kg]	1.900e+27
Mass	[Earth = 1]	317.9
Equatorial radius	[km]	71,492
Equatorial radius	[Earth = 1]	11.21
Mean density	[g/cm^3]	1.33
Mean distance from the Sun	[km]	778,330,000
Mean distance from the Sun	[AU]	5.2028
Rotational period	[days]	0.41354
Orbital period	[days	4332.71
Mean orbital velocity	[km/sec]	13.07
Orbital eccentricity	[-]	0.0483
Tilt of axis	[degrees]	3.13
Orbital inclination	[degrees]	1.308
Equatorial surface gravity	[m/sec^2]	22.88
Equatorial escape velocity	[km/sec]	59.56
Visual geometric albedo	[-]	0.52
Mean cloud temperature	[°C]	-121
Atmospheric (volumetric) Composition:		
Hydrogen	[%]	86
Helium	[%]	14
Minor constituents	[%]	< 0.2

Table 2: Main characteristics of Jupiter and its Galilean Moons

	Jupiter	Io	Europa	Ganymede	Callisto
Radius (km)	71492	1820	1565	2634	2403
Distance to Jupiter (R_J)	-	5.9	9.4	15.0	26.3
Inclination (deg)	-	0.04	0.47	0.21	0.51
Period (d/h)	(11.863 y)	1.769 / 42.46	3.551 / 85.23	7.155 / 141.71	16.689 / 400.54
Rotation (d/h)	0.414 / 9.925	Synchronous			

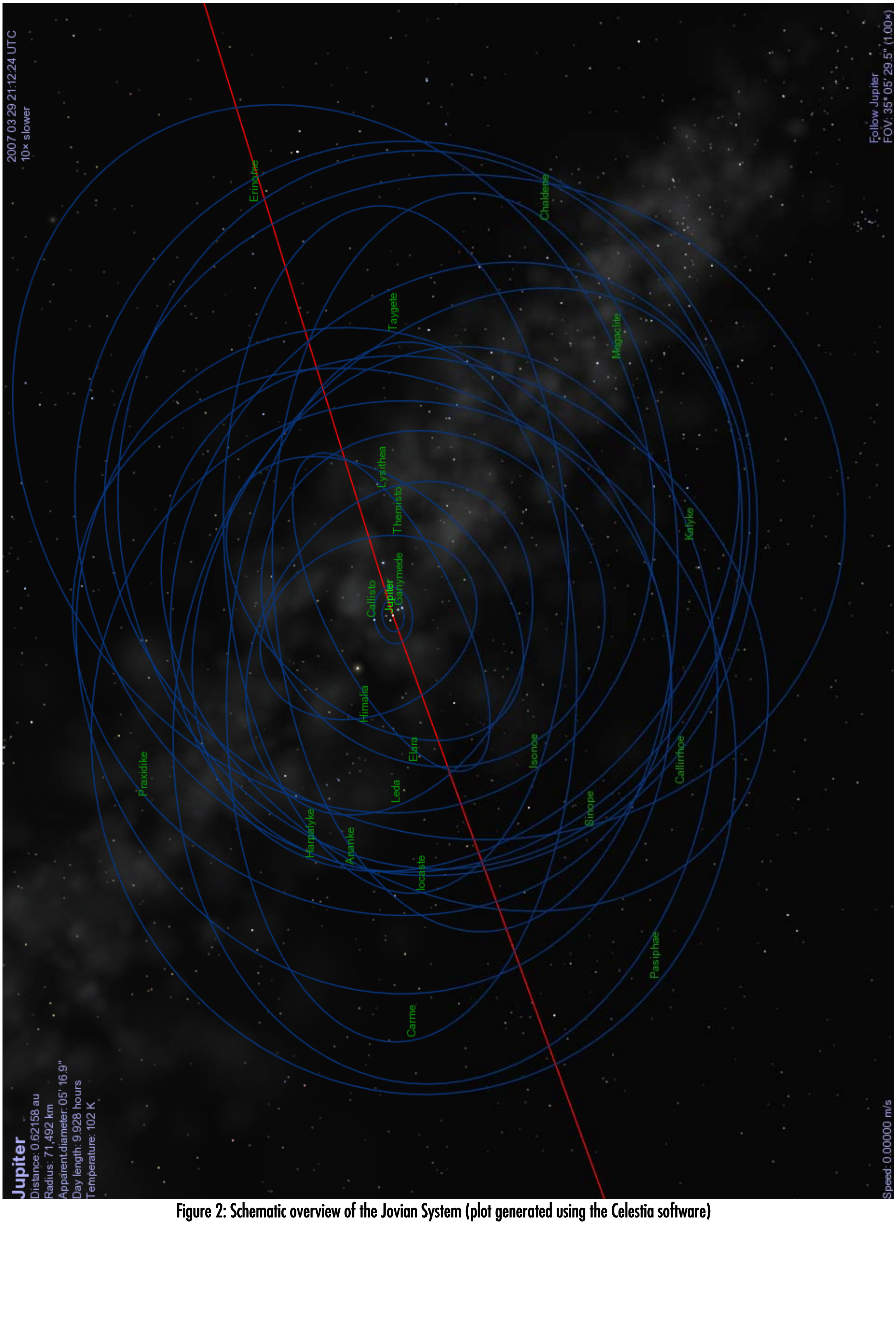

Figure 2: Schematic overview of the Jovian System (plot generated using the Celestia software)

PREVIOUS MISSIONS

Until now, a limited number of missions has studied the Jovian System: Pioneers 10 and 11 were the first, providing information on the Jovian radiation and magnetosphere in the early 1970s, followed by the Voyagers 1 and 2 at the end of the same decade, providing multi-band imaging, as well as radiation and atmospheric observations of Jupiter and the Galilean moons.
Ulysses was the first spacecraft to visit Jupiter (1992) since the Voyager missions in the 1970s, when it used Jupiter for a gravity assist to swing out of the ecliptic plane towards an orbit around the poles of the Sun. Its visit of Jupiter supplied valuable information on the Jovian radiation and magnetic environment. The last mission focussing on Jupiter was Galileo: it was launched in 1989 and has ended its mission on the 21st of September 2003 after being deliberately targeted into the Jupiter atmosphere. This spacecraft provided the most extensive study of the Jovian System until now, including in situ measurements of Jupiter's atmosphere by means of an atmospheric probe [3]. Finally, The Cassini-Huygens flyby of Jupiter in 2000 also provided valuable data on the Jovian radiation environment.

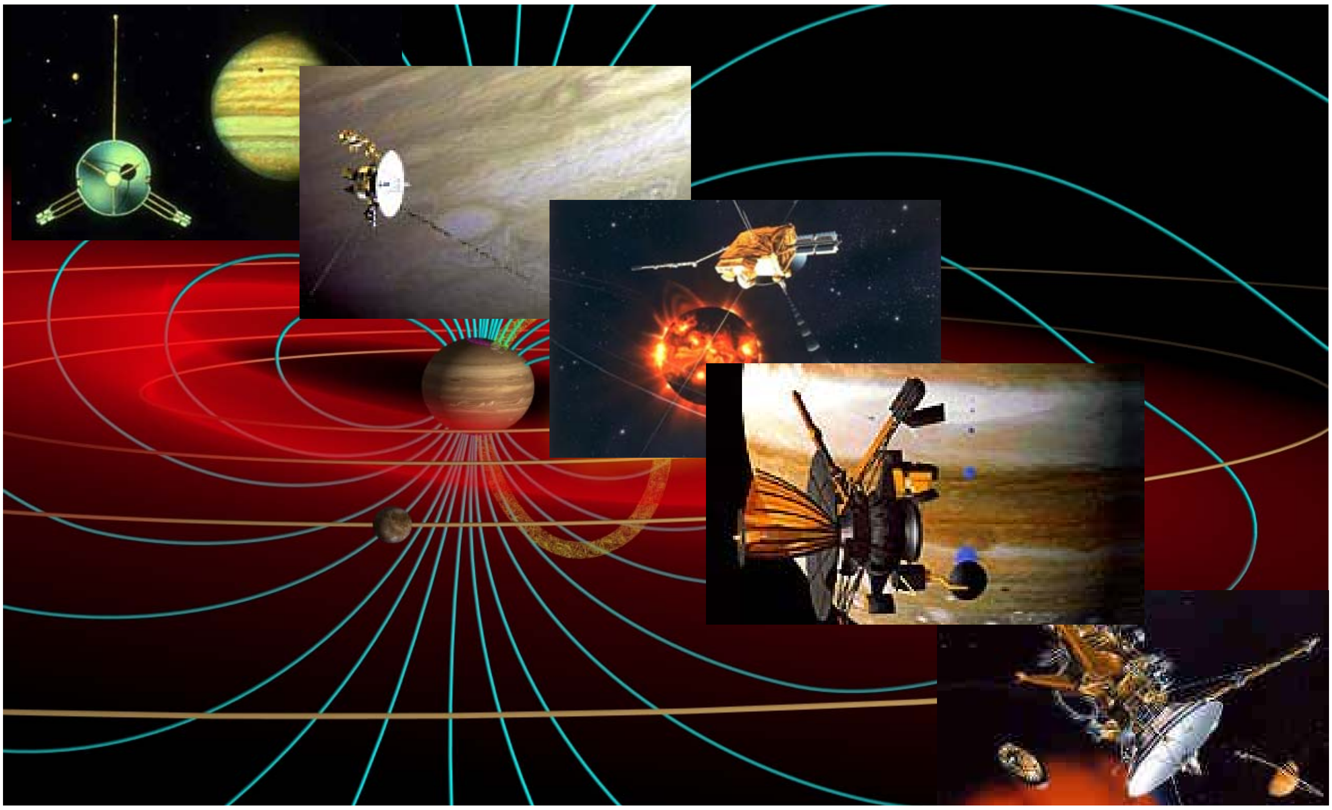

Figure 3: Overview of the spacecraft that visited the Jovian System until now: Pioneers 10&11, Voyagers 1&2, Ulysses, Galileo (including probe) and Cassini-Huygens

Two other missions must be mentioned that will be able to provide inputs for a Jovian mission in the 2020 timeframe:

At the moment of writing, the NASA New Horizons mission to Pluto and the Kuiper Belt performed a gravity assist around Jupiter. All the science instruments have been activated 60 days before and 160 days after closest approach (closest distance to Jupiter: R_J=32) including an instrument which will measure energetic particles and plasma around Jupiter. The results will help to improve our knowledge of the radiation environment of Jupiter. Moreover for the first time a spacecraft will fly through almost the complete tail of Jupiter's magnetosphere, which will help to improve the magnetic field models used in the definition of the radiation environment of Jupiter.

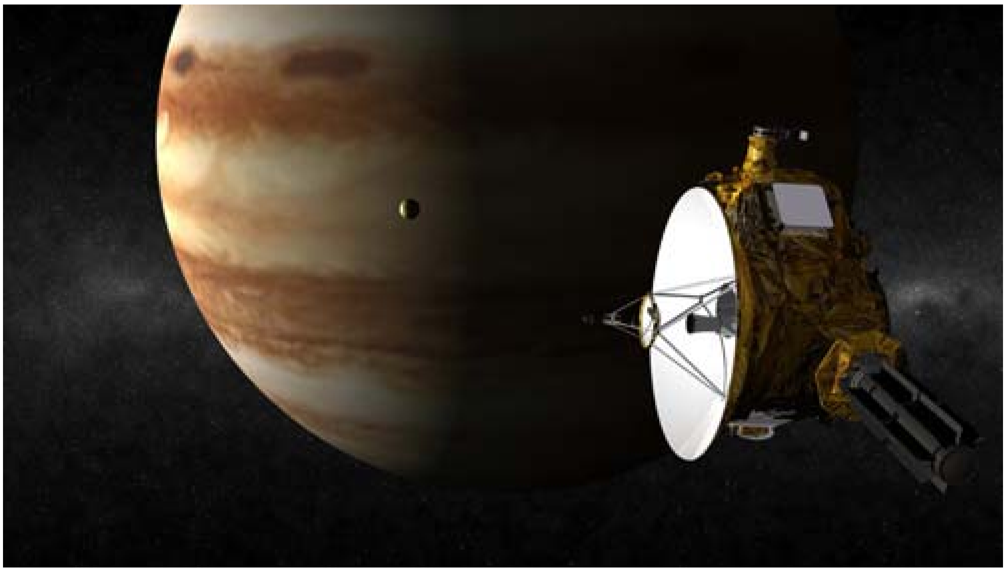

Figure 4: The New Horizon mission fly-by of Jupiter [http://pluto.jhuapl.edu]

Another mission that is dedicated to the Jovian System is currently planned for 2011: the NASA Juno mission, which will study the composition and dynamics of Jupiter's atmosphere, will determine whether Jupiter has an ice-rock core, investigate the origin of the Jovian magnetic field and will explore the planet's magnetosphere in polar regions.

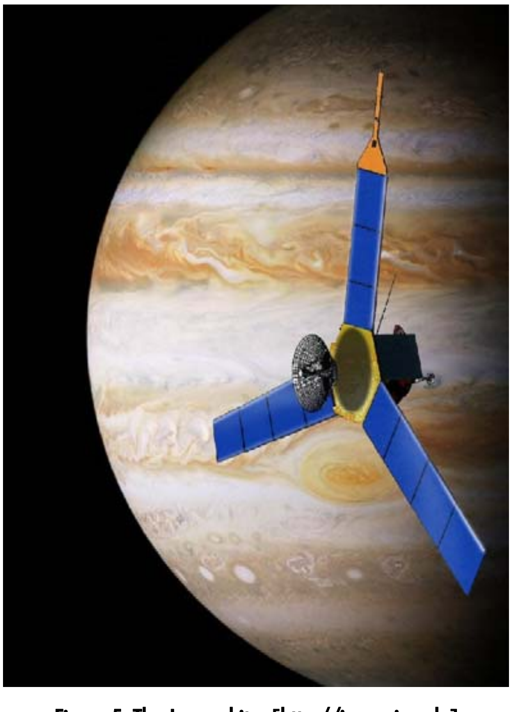

Figure 5: The Juno orbiter [http://juno.wisc.edu]

RADIATION ENVIRONMENT OF JUPITER: MODELS AND MEASUREMENTS

The radiation environment of Jupiter is considered to be one of the most aggressive in terms of ionising radiation in our Solar System. The afore mentioned previous missions have performed in-situ measurements of both inside Io's orbit (Pioneer 1 & 2) and outside Io's orbit. From a much farther range Ulysses and Cassini-Huygens have performed measurements, which have helped to constrain the models for Jupiter's ionising environment. In pre-Galileo times the Divine-Garrett model (D&G) [4] has been used to model the proton and electron fluences in the Jovian System. Recent upgrades of the Divine-Garrett model based on Galileo data were incorporated in the GIRE (Galileo Interim Radiation Environment) model. The latest model developed by ONERA took a different approach: The goal was to develop a theoretical model of the trapped electrons and protons in Jupiter radiation belts. It is based on the Earth's Salammbô code. In this model, all physical processes which act on a trapped electron or proton are taken into account and are only constrained by the local environment. The Salammbo model also incorporates a much more realistic model for the magnetic field of Jupiter. The spatial and spectral coverage of the original Divine & Garrett model, the GIRE and the new Jupiter Salammbô model are given in Figure 6 and Figure 7 [5]

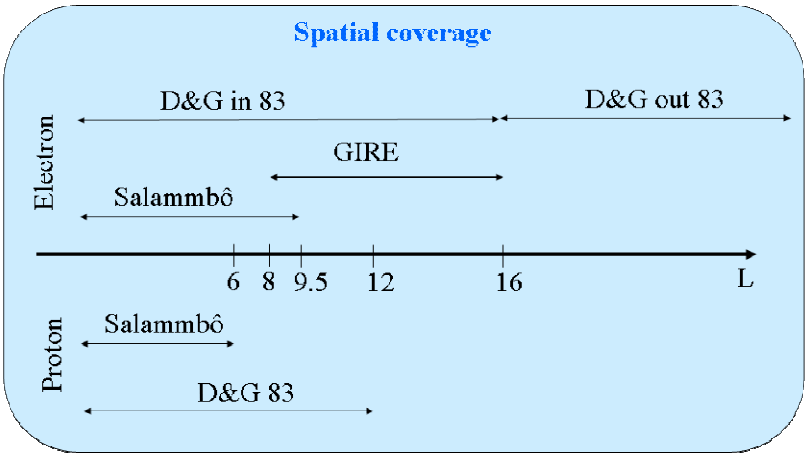

Figure 6 : Summary of spatial coverage for the different Jovian radiation belt models.

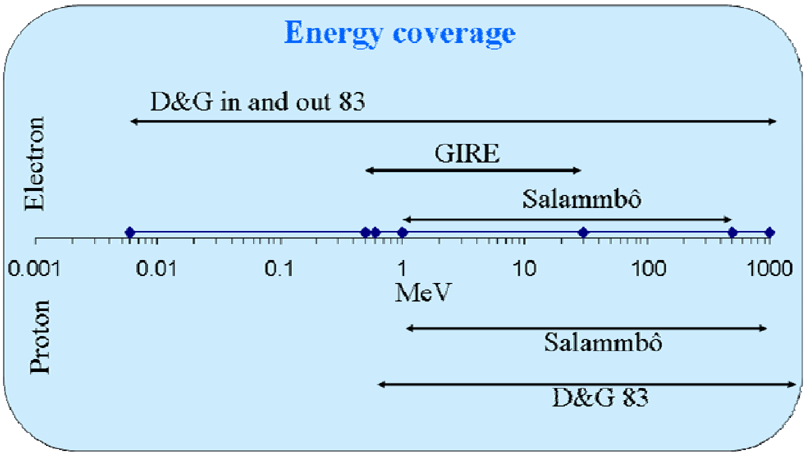

Figure 7: Summary of energy coverage for the different Jovian radiation belt models.

The horizontal scale in Figure 6 is defined as L, which represents the magnetic shell parameter, which in can be regarded as the distance from Jupiter based on the magnetic field model. It corresponds closely to the actual distance from Jupiter. The Salammbô model is valid for ranges up to the orbit of Europa (R_J=9.4) for electrons and up to the orbit of Io (R_J=6) for protons.

In Figure 7 the spectral ranges of the three models are given and it is clear they cover more or less the same spectral range for protons, while for electrons the D&G model also takes into account the low energy electrons (< 1 MeV).

One critical feature is that the more sophisticated model of Salammbô yields far higher fluences of protons inside Io's orbit than the D&G model, which can be seen in Figure 8. This figure shows that the Salammbô model is in better accordance with the actual data taken by the Galileo probe.

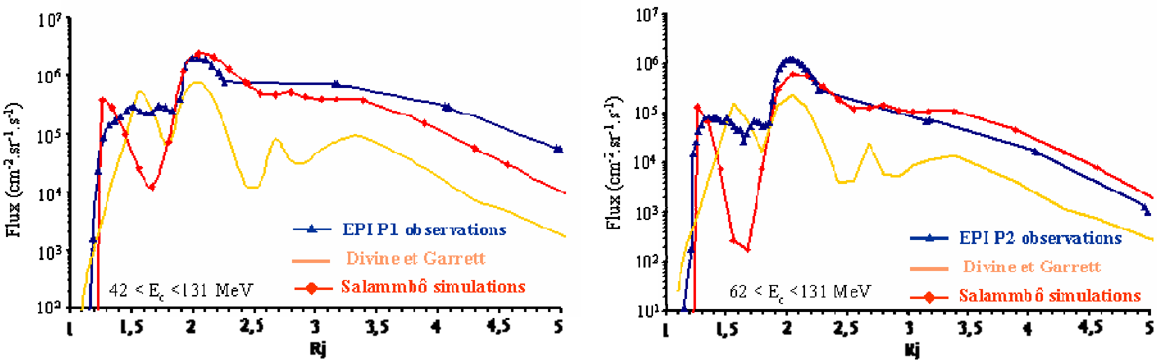

Figure 8: Comparison of proton measurements onboard GALILEO probe and predictions by D&G83 and Salammbô. From A. Sicard [6]

The output of the Salammbo, GIRE and D&G models are fluences of protons and electrons. These results can be used in SHIELDOSE-2 or MULASSIS (two software models) to calculate the total dose as a function of the thickness of the shielding material to define the required amount of shielding. In the most commonly cited calculations, the shielded material is normally taken to be Aluminium. A more sophisticated approach can be used which could lead to shielding mass reductions. In this approach it is taken into account that electrons and secondary radiation are better blocked by a high Z material (where Z is the atomic number) and protons are better blocked by a low Z material. A triple layer of low Z material, high Z material and low Z material could lead to shielding mass reductions.

It is important to note that the current models only describe the trapped particles in Jupiter's radiation belt and exclude solar energetic particles, cosmic rays and heavy ions from the Jovian System. A new activity will be started to model the effect of these particles to have a complete picture of the radiation environment of Jupiter.

It must be noted that **the dose calculations cited in the JME study are based on the Divine&Garrett 83 model**, as the Salammbô model was not yet available at the time of that study. The used D&G 83 model is now briefly described [7].

A simplified model is employed to describe the radiation environment in the inner regions of the Jovian System, where the harshest conditions prevail. The model was taken from [8].It describes five parameters as function of the distance from Jupiter's centre of mass: proton and electron dose rates with 4 and 8 mm of aluminium shielding (to assess the dose absorbed by electronics components) and 1 MeV equivalent electron flux on GaAs solar cells with 509 µm cover glass thickness (to assess the degradation of the solar arrays).

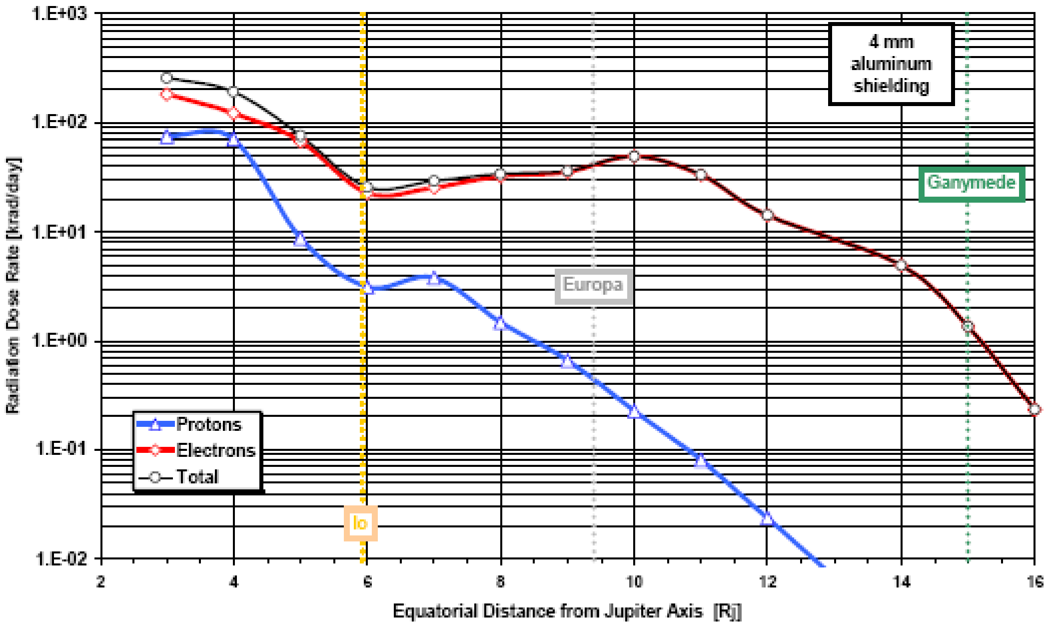

Figure 9: Radiation dose rate with 4 mm Al shielding as function of Range

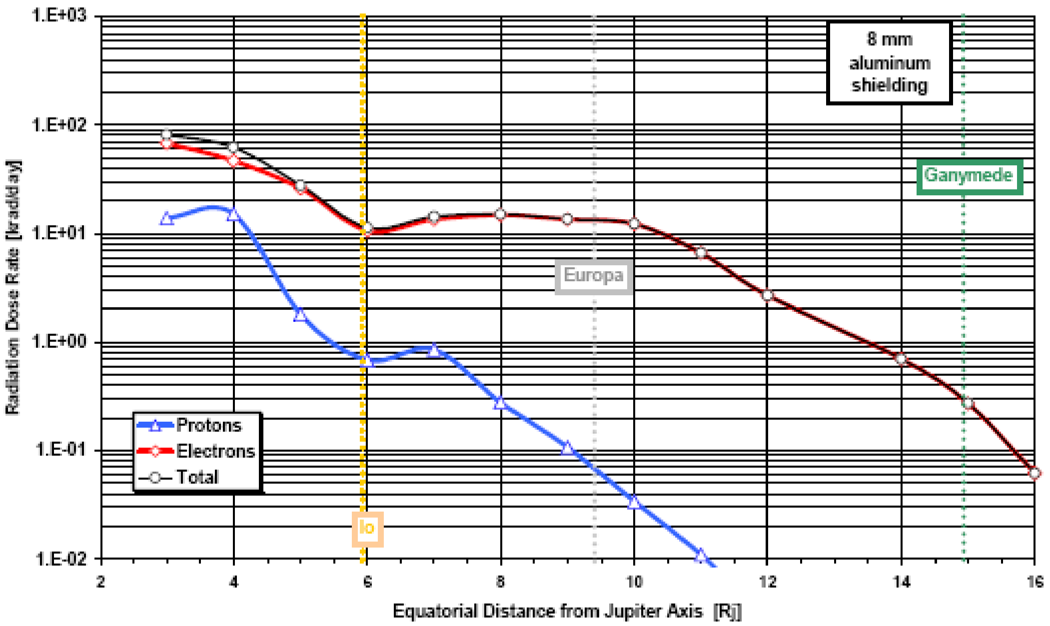

Figure 10: Radiation dose with 8 mm Al shielding as function of range

Figure 9 shows the proton and electron dose rate for a shielding thickness of 4 mm, while Figure 10 shows the same values for a thickness of 8 mm over the distance from the Jupiter axis of rotation. The radiation environment is assumed to be isotropic; variations with latitude and right ascension are disregarded. Figure 11 shows the equivalent 1 MeV electron flux for GaAs solar cells protected by cover glass of 509 µm thickness. Here again, isotropy is implicitly assumed.

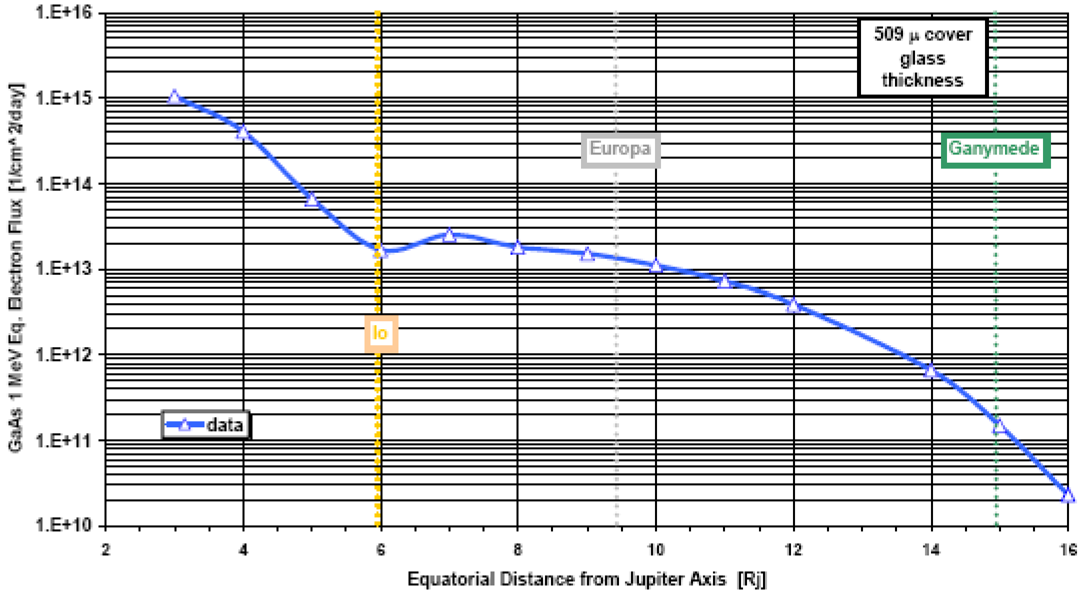

Figure 11 GaAs Solar Cell Equivalent 1 MeV Electron Flux as Function of Jupiter Range

The following Chapters will discuss the results of the main activities performed until now, subdivided in the relevant scenarios.

THE JOVIAN MINISAT EXPLORER – EXPLORING EUROPA

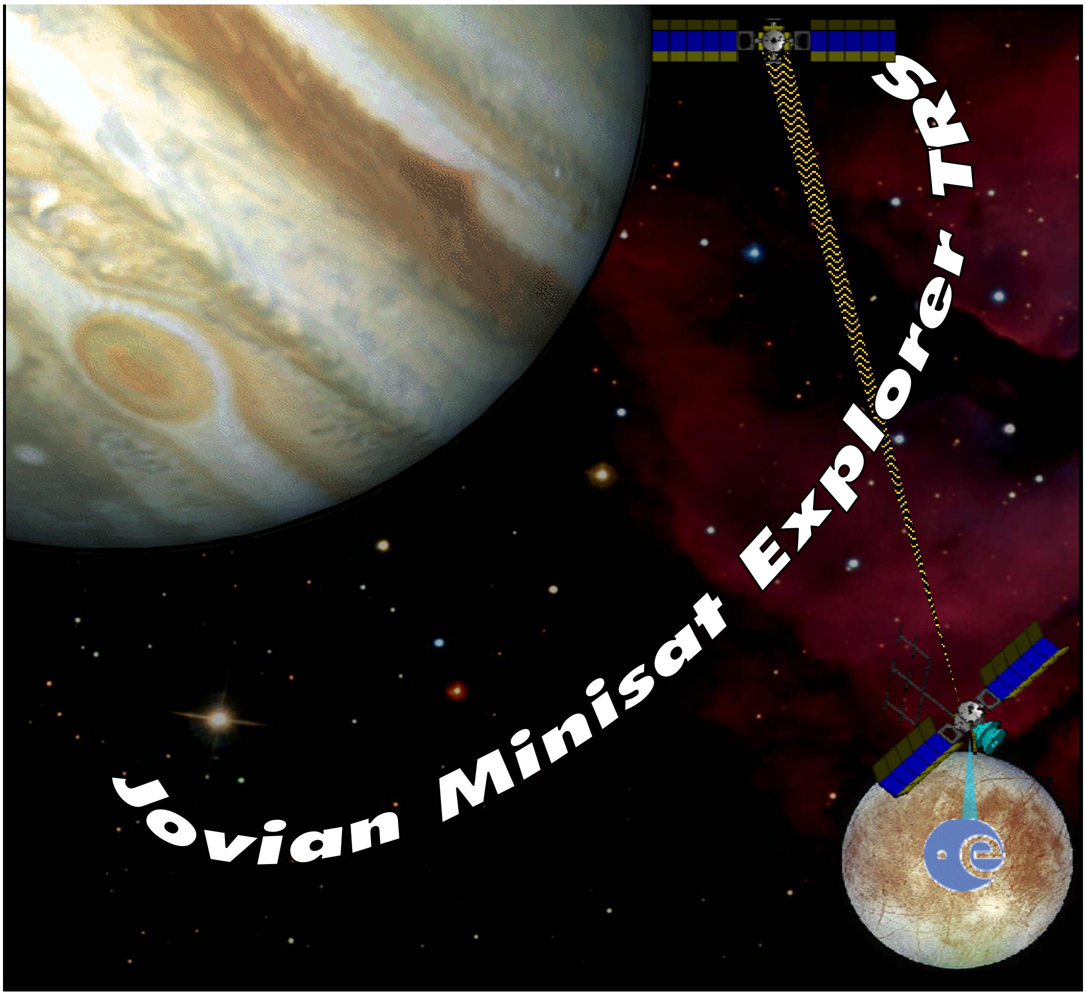

Table 3: Jovian Minisat Explorer Summary (status March 07)

Scientific objectives:		
Primary Objectives	• Determine the presence or absence of a subsurface ocean (includes mapping of the ice thickness) • Measure the global topography and the tidal effects at Europa • Characterise the global geology and surface composition of Europa • Observe Europa's magnetic field • Measure the radiation environment around Europa	
Secondary Objectives	• Measurement of the plasma environment • Imaging of the Jovian System • Characterisation of the Jovian plasma and dust environment • Determine presence and composition of a Europa atmosphere	
Strawman payload	**Jupiter Europa Orbiter**	**Jupiter Relay Satellite**
	• Miniaturised GPR • Stereo micro-camera • VIS-NIR mapping • Altimeter • Magnetometer • γ-ray • UV spectrometer • Radiation monitor • Radiometer	• Plasma Wave Instrument • VIS-NIR camera • Magnetometer • Dust Detector • Radiation monitor
Launch & transfer		
	• Soyuz Fregat 2-1b launch from Kourou (April 2019, 3 week launch window). Capacity into GTO: 3090 kg • 2 S/C composite transfer to Jupiter via an Earth-Venus-Earth-Earth GAM • Transfer duration ~ 7 years • After Jupiter Orbit Insertion the S/C separate and both perform a tour of the Jovian System • JEO will achieve orbit around Europa in 545 days (TBC) • JRS will achieve a highly elliptical orbit around Jupiter (~20 degree inclination w.r.t. equator) 542 days, 3:1 resonance with Europa	
Main mission details:	**JEO**	**JRS**
Operational orbit	200 km circular orbit, period = 2.3 h	11 R_J * 28.1 R_J (R_J=71492 km), period = 10.7 days
Mission Lifetime	• 7.3+2 years until Europa orbit insertion • ~ 60 days of science operations	• 7.3+1.6 years until final orbit insertion • ~ 1.5-2 years of science & relay operations
Delta-V (m/s)	1134	2618
Stabilisation	3-axis	3-axis
Orientation	Nadir / JRS	JEO/Earth
Mass:	Mass figures include 5-20% component margin (depending on maturity) and 20% system margin.	
Payload (kg)	43 kg (=36kg+20%)	20 kg (=17kg+20%)
Dry (incl. P/L) (kg)	432	628
Wet (incl. P/L) (kg)	630	2272 (incl. 40 kg adapter)
Margin w.r.t. launcher (kg)	188 (6 %)	
Dimensions (mm)	HxWxD: 1303 x 1340 x 1340	HxWxD: 2090 x 1340 x 1340
Radiation	~ 5.3 Mrad (4 mm Al shielding) (TBC)	~1 Mrad (4 mm Al shielding), after ~1 year of operation
Max power [peak/avg] (W)	216 / 150	436 / 254
TM band	Ka, X	Ka, X
Antenna	HGA, MGA	HGA, MGA, LGA
Data storage (Gbit)	64	320
Total data volume (Gbit)	~300	Depends on JRS lifetime: up to 300 Gbit/year with current assumptions (1 year necessary for JEO data)
P/L power (W)	24	10.5
Avg P/L data rate (kbps)	23	13
S/C structure material	CFRP	CFRP
Challenges		
	• Radiation: High radiation dose hardened systems (up to 1 Mrad) • Efficient high radiation compatible, low intensity, low temperature solar power generators • Development of solar concentrators: distance from Sun ~5.2 AU, hence solar flux ~1/25th wr.t. Earth flux • Thermal: Hot and cold case drivers (Venus fly-by and Jovian System respectively) • Low resource: Both P/L and S/C subsystems (including comms) must have low resource requirements. Significant effort needs to be invested in the developments of the Highly Integrated Payload Suite • Communications: high efficiency SSPA's, high data rate X-&KA-band transponders • Autonomy: Highly autonomous S/C during entire mission as well as for commissioning and ops around Europa • Planetary protection: JEO needs to be integrated while meeting the highest planetary protection requirements	

JME-1 THE SCIENTIFIC INTEREST IN EUROPA

The main scientific objective of the JME is to perform detailed remote sensing of Europa, with a potential deployment of a microprobe for in-situ analysis.

Table 4: Europa's main properties [2]

Item	Unit	value
Mass	[kg]	4.8e+22
Mass	[Earth = 1]	8.0321e-03
Equatorial radius	[km]	1,569
Equatorial radius	[Earth = 1]	0.246
Mean density	[g/cm^3]	3.01
Mean distance from Jupiter	[km]	670,900
Mean distance from Jupiter	[R_J]	9.4
Rotational period	[days]	3.551181
Orbital period	[days	3.551181
Mean orbital velocity	[km/sec]	13.74
Orbital eccentricity	[-]	0.009
Orbital inclination	[degrees]	0.470
Escape velocity	[km/sec]	2.02
Visual geometric albedo	[-]	0.64

The main issues that need to be addressed in the study of Europa are [9]:

JME-1.1 Looking for a subsurface ocean

Previous observations of Europa confirmed that its surface consists of an icy crust. Observational data suggests the presence of a metallic core and a rocky mantle. However, it's unclear whether the ice layer runs until the rock, or if the outer crust is partly liquid. Models based on gravity measurements provided by the Galileo spacecraft allow for an outer layer of water and ice, with a possible thickness up to 200 km. Galileo's detection of a magnetic field hints at a metallic core, further supporting the three layer model, depicted in Figure 12. Another theory suggests that the magnetic field is induced by the interaction of Jupiter's field and a salty subsurface ocean. Present data is not sufficient to determine the interior structure of the icy moon; more detailed observations will be necessary.

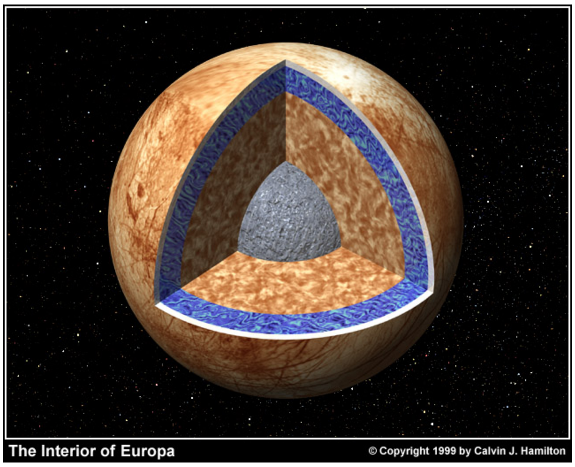

Figure 12: The three-layer model of Europa's interior

JME-1.2 Global topography

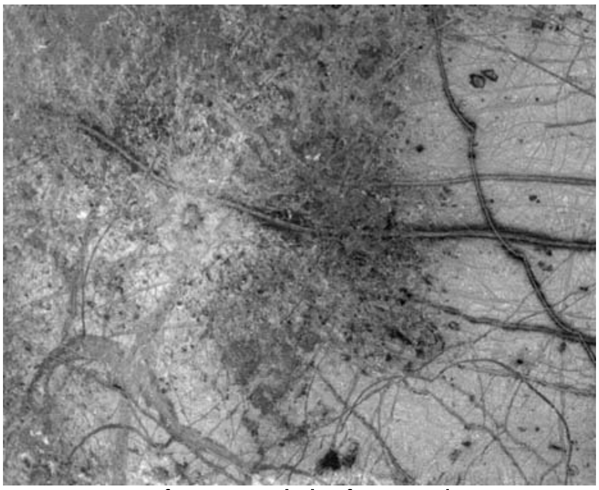

Figure 13: A view of Europa's cracked surface [NASA/JPL]

Striking features have been observed on Europa's surface. These vary from surface cracks filled with darker material to impact craters. The most striking features concern the so-called triple bands: two parallel bands of dark matter separated by a lighter central band (see Figure 13), that can reach lengths up to 1000 km and widths of 20 km. A range of theories exist on how these bands have formed, varying from tectonic activities causing fracturing of the crust, followed by intrusion of subsurface material, to explosive venting by geysers. After establishing the cause of these triple bands, the next step is to observe any possible deformation of the surface features over time.

Observation of the impact craters, the cracks and other formations over time will provide information on composition, surface dynamics, as well as age. This last issue is contentious, since two age-models exist with a massive discrepancy between them. New data is required to shed light on this issue. Furthermore high-resolution images of particular features are needed to determine if large flows and outpourings to the surface take place.

JME-1.3 Composition of the (sub)surface

Europa's surface shows dark bands, spots and mottled zones: the colour and spectral properties change over the surface and cannot only be attributed to a combination of water ice and sulphur. Therefore silicate minerals or other non-ice components are likely to be present. Observations have led to models where a zone of these non-ice materials (e.g. clay and salts) exists a few kilometres under the surface. Analyses of ejecta from impact craters as well as of mottled zones suggest an upwelling of material from the subsurface that could provide information on the materials present on and under the icy surface. The analysis should also focus on presence or absence of compounds that enable organic evolution, one of the necessary building blocks for life, as we know it.

JME-2 THE JOVIAN MINISAT EXPLORER TRS

The Jovian Minisat Explorer TRS concerns the exploration of the Jovian System, and especially Europa, the smallest of the four Galilean moons orbiting Jupiter. This moon has been selected, as it is one of the few places where liquid water may be found in the solar system, making it one of the prime candidates for the search for life outside Earth.

The current scenario foresees two relatively small spacecraft (~ 600/400 kg dry mass): one will act as a relay spacecraft (Jupiter Relay Spacecraft (JRS)) in a highly elliptical orbit around Jupiter, outside the high radiation zones, while the other (Jupiter Europa Orbiter (JEO)) will orbit Europa. The Europa orbiter will include a highly integrated remote sensing payload suite and a communication system for communications with the JRS and Earth.
The relay spacecraft will carry all subsystems that are not directly required for the Europa observation mission, as it will be subjected to less radiation than the Europa orbiter. It will carry the communication system providing the link between Earth and the JEO, data processing and data storage units as well as its own highly integrated scientific payload suite dedicated to the study of the Jovian System. The name “Relay Spacecraft” is actually not the most appropriate name as this spacecraft will be able to provide an important contribution to the science mission: the JRS will be in a very interesting orbit in the Jovian System that will allow for observation of several key aspects of the Jovian System such as the magnetosphere, Jupiter’s atmosphere, the particles in the system, etc.

The presented configurations are a result of a feasibility study performed by EADS Astrium and two successive CDF studies. The design details and spacecraft system budgets reflect the current status (March 07). More details on this study can be found in the extensive final report of this study [10] as well as ensuing CDF study reports [11,12]. The instruments used in this study are part of a strawman payload derived from goals specified in [9], which was necessary to understand the implication of the payload on the spacecraft. This selection is in not intended to preclude further inputs from the scientific community.

This TRS requires sound scientific mission objectives, in order to assess the impact of the payload on the mission. The following objectives are based on the recommendations made by the Committee on Planetary and Lunar Exploration [9], which assessed the required investigations of Europa.

For the purpose of this TRS, the top-level scientific objectives are:

- Determine the presence or absence of a subsurface ocean (includes mapping of the ice thickness)
- Measure the global topography and the tidal effects at Europa
- Characterise the global geology and surface composition (includes measurement of the geochemical environment of the (sub)surface)
- Observe the moon's magnetic field
- Measure the radiation environment

Objectives of secondary importance are:

- Measure the plasma environment of Europa
- Imaging of the Jovian System
- Characterisation of the Jovian plasma and dust environment
- Determine presence and composition of a Europa exosphere

For more details, please refer to the JME Science Requirement Document [13]

JME-3 THE STRAWMAN PAYLOAD

As explained before, it is not the goal of the TRS to fix the payload of an eventual mission to Europa. However the JME needs a representative strawman payload in order to design the mission. Therefore a selection has been made of payloads capable of addressing the scientific objectives stated in paragraph JME-2. This selection is summarised in the following tables and is based on [14]. Additional points that must be considered with this concept are summarised in the payload section on page 74.

Table 5: The strawman JEO payload suite

Instrument	Abbreviation	Goal
Ground Penetrating Radar	EuGPR	Mapping thickness ice layer, structure determination, topography, surface reflectivity
Stereo micro-camera	EuSCam	Topography, geology and surface composition
VIS-NIR mapping	EuVN-IMS	Topography, geology and surface composition
Radiometer	EuRad	Measuring Europa's surface temperature
Altimeter	EuLat	Topographical mapping, study of tidal processes
Magnetometer	EuMAG	Measuring Europa's magnetic field
γ-ray spectrometer	EuGS	Surface composition
UV camera	EuUVcam	Measuring the electron environment of Europa
Radiation monitor	EuREM	Analysis of Jovian radiation environment

Table 6: The strawman JRS payload suite

Instrument	Abbreviation	Goal
Radiation monitor	JuREM	Measurement of the Jovian radiation environment
Plasma Wave Instrument	JuPWI	Investigate plasma waves and radio emissions in controlling the scattering and/or loss of trapped radiation in the Jovian magnetosphere
Narrow angle camera	JuNaCam	Imaging the Jovian System, especially Jupiter
Magnetometer	JuMag	Mapping the electromagnetic field of the Jovian System
Dust Detector	JuDustor	Measure particle size and velocity distribution

These payloads are envisaged as highly integrated payload suites (HIPS): The use of miniaturised components as well as the sharing of common subsystems and functionalities allow for an optimal reduction in resource requirements. This has to be achieved without degrading the scientific return, which calls for the use of state-of-the-art micro- and possibly nano-technologies. This approach is especially appealing for the JME as the HIPS approach is compatible with the low resources available. Furthermore it is also beneficial in view of the high radiation environment: by miniaturising and integrating the payload in a compact volume, it will be easier to protect the payload within a shielded "box", limiting the required shielding mass. The HIPS approach enables a relatively low payload mass, when compared to more conventional instrument suites. The assessment of the HIPS payload suite [14] shows that (provided that the required HIPS technology is developed) the JEO HIPS requirements will be in the order of 32 kg and 25 W, while those of the JRS HIPS will be around 15.5 kg and 10.5 W. The radar, one of the most demanding instruments, has been assessed during a Concurrent Design Facility study at ESTEC, to understand what can be achieved with a low resource radar [15] the results are presented in the next section.

JME-3.1 The JEO HIPS

The following table shows a summary of the strawman payload suite assumed for the JEO.

Table 7: Summary of the JEO HIPS

Instrument	Mass (kg)	Power (W)	Aperture (mm)	Dimension (mm)	FOV (deg)	Pointing direction	Data rate (bps)
EuRR (50 MHz)	9.2	25	-	Stowed: 1340*470x300 Deployed:10 000x2 000	-	nadir	28000
EuSCam	0.6	0.66	35	TBD	4	nadir	5093
EuVN-IMS	1.8	1	38	TBD	4	nadir	13221
EuRad	1.6	0.96	50	60x100x200	2.0 x 4.0	nadir	109
EuLat	2.0	2.5	40	100x100x300	0.2 mrad	nadir	3000
EuMAG	0.7	1.5	-	100x50x100	-	-	248
EuGS	3.6	1	80*	110 (diam) **	92.4	nadir	TBD
EuUVcam	0.7	0.62	20 x 20	40x40x100	0.1x1.0 mrad	nadir	200
EuREM	0.5	0.85	<1	100x50x100	60-100	Limb/nadir	273
Boom	0.4	0.85	-	1500	-	-	-
DPU + CPS	2.0	3.41	-	200x100x50	-	-	
Shielding (20%)	4.6	-	-	-	-	-	
Structure	2.5	-	-	-	-	-	
Margin (20%)[1]	6.0	5.3/2.7	-	-	-	-	
TOTAL	**36.3**	**30/16**	**--**	**--**	**--**	**--**	**28k/22.1k+**

*= diameter of sphere **= dimension of outer shield

Due to the limited resources available, not all instruments can be switched on simultaneously. Because of the resource requirements of the radar, this instrument must be operated independently from the rest of the payload. Therefore two science modes have been identified: one mode for the radar operation, the other for the remaining payloads. The total power and data rate in Table 7 have been subdivided in these two modes. The next figure provides a possible configuration of the JEO HIPS.

The addition of a Radio Science Experiment (RSE) needs to be considered, to ensure accurate orbit determination that is required to correctly interpret the result of e.g. the EuLat. The RSE is not included in this study.

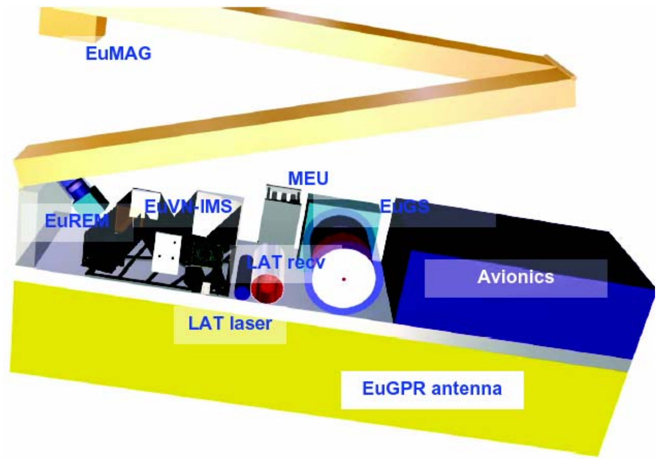

Figure 14: View of a possible configuration of the JEO HIPS [14]

[1] The HIPS study assumed a 10% subsystem margin. However, in view of the new development required for the HIPS payload, 20% is better suited and this has been assumed for the presented JEO design

The CDF study of the low resource radar resulted in the conceptual design of the *Europa Low Resource Radar*, ELRR. A 50 MHz frequency was selected for this ground penetrating radar as a compromise between too much clutter (at higher frequencies) and too much noise generated by the Jovian System at lower frequencies. The selected antenna is a triple three element Yagi, shown in the following figure.

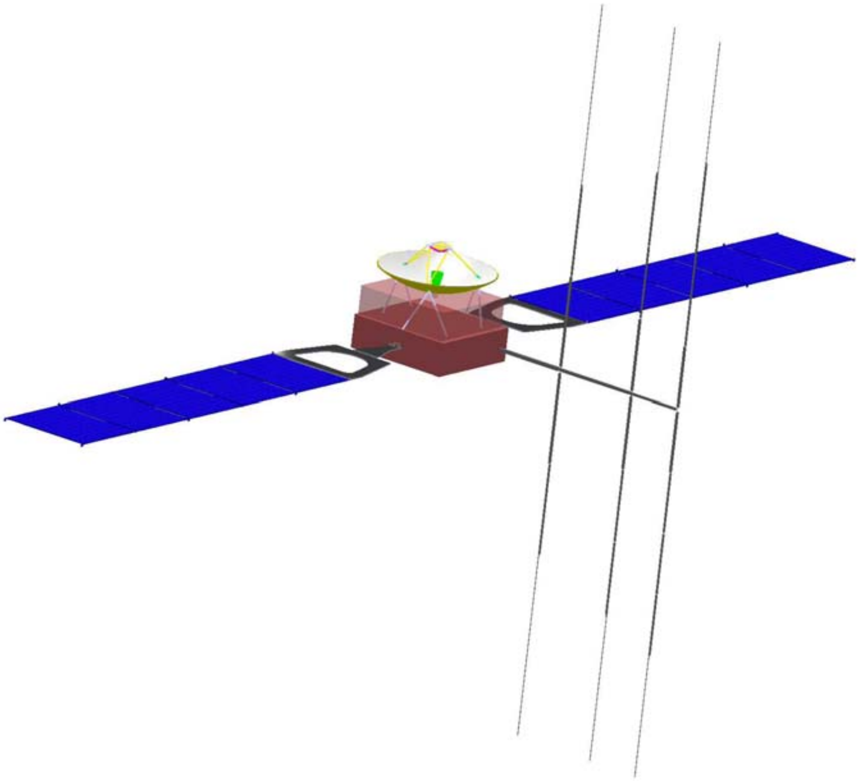

Figure 15: View of the deployed ELRR [15]

One of the main challenges lies in the accommodation of the antenna on the relatively small JEO. As a result of the reduced dimensions, the 12 x 2 m Yagi antenna needs to be folded into a small volume (1.34 x 0.47 x 0.3 m). To accommodate the antenna, a large number of hinges is required (~10), which poses a considerable risk for the deployment, especially after having been folded for seven years. This issue will require significant attention, should such a mission be selected in future. Other critical areas concern the used materials (stiffness, high radiation compatibility) and accurate nadir and attitude pointing control. More details can be found in [15 and 12].

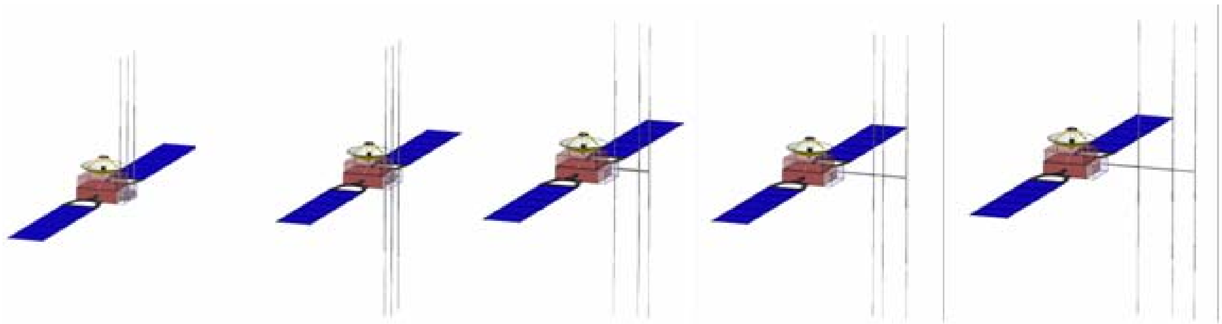

Figure 16: deployment sequence of the ELRR [15]

JME-3.2 The JRS HIPS

The following table shows a summary of the strawman payload suite used for the JRS design.

Table 8: Summary of the JRS HIPS

Instrument	Mass (kg)	Power (W)	Aperture (mm)	Dimension (mm)	FOV (deg)	Pointing direction	Data rate (bps)
JuREM	0.5	0.9	<1	60x40x40	60-100	limb/nadir	273
JuPWI	3.7	2.1	N/A	100x200x120	-	-	3750
JuNaCam	1.7	1	60	300x180x105	2.0	nadir	9128
JuMag	0.7	1.5	N/A	100x50x100	-	-	248
JuDustor	0.7	1	100*	263x177x177	120	limb/nadir	16
Boom	0.5	-	-	1500	-	-	-
DPU + CPS	2	3.1	-	200x100x50	-	-	-
Shielding (20%)	2.0	-	-	-	-	-	-
Structure	2.5	-	-	-	-	-	-
Margin (20%)[2]	2.9	1.9	-	-	-	-	-
TOTAL	**17.1**	**11.5**	**--**	**--**	**--**	**--**	**13.4k**

* entrance grid = 10 mm

The emphasis of this study lies on the science gathered by JEO. However, having a relay spacecraft orbiting Jupiter offers the unique opportunity to study the gas giant. Therefore the JRS has also been equipped with instruments capable of studying the Jovian environment. In view of the limited communication resources, the down link of gathered JRS data will need to be carefully co-ordinated with the JEO data. Data that cannot be sent to Earth directly will need to be stored onboard until all JEO data is sent. Clearly this will impose further requirements on the radiation tolerance of the memory as well as the other electronics. As explained before, if a similar mission scenario will be selected for further study, the scientific potential of the JRS must be carefully studied, improved with respect to this strawman payload, which will have to be adapted accordingly.

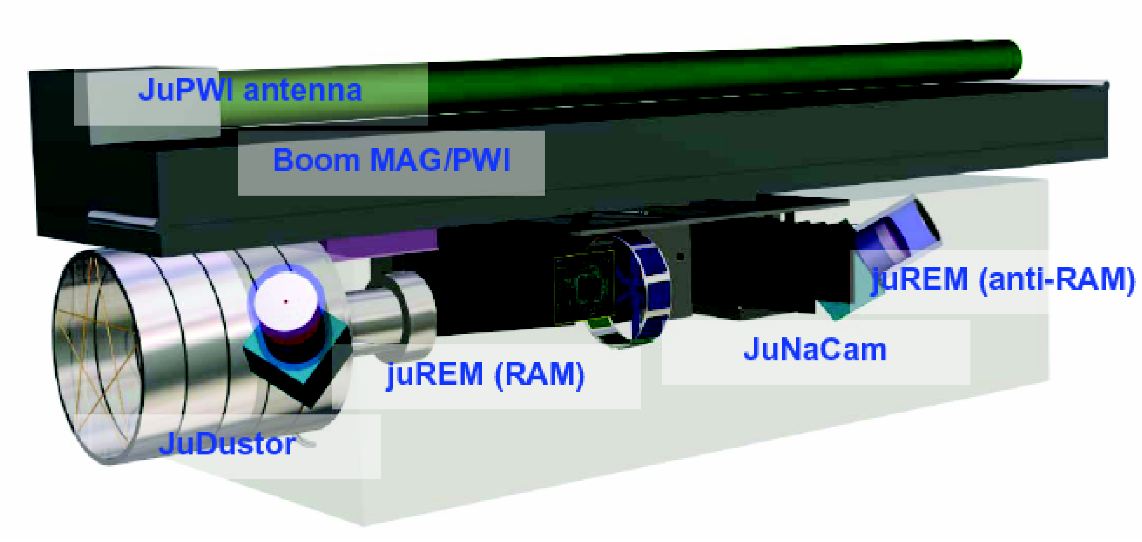

Figure 17: View of a possible configuration of the JRS HIPS [14]

[2] NB: the HIPS study assumed a 10% subsystem margin. However, in view of the new development required for the HIPS payload, 20% is better suited and this has been assumed for the presented JRS design

JME-3.3 Impactors

A considerable interest has been expressed for in situ measurements of Europa's crust. In view of the limited payload resources in this mission concept, two options have been assessed at conceptual level.

JME-3.3.1 JEO Microprobe Analysis (JEOMA)

The feasibility of a microprobe in the 1 kg mass range has been investigated in the JEOMA study [16], performed by ESYS and PSSRI (UK). This mass budget is a severe limit for any probe, but in view of the limited mass availability, a large impactor is unfeasible: it is not only the mass of the impactor itself, but also the snowball effect that the accommodation of such a probe would have on the launch mass.

The probe would need to penetrate the icy surface of Europa to perform basic measurements of the ice crust with a very limited instrument suite. However, as Europa has no appreciable atmosphere, the probe will either need a propulsion system capable of decelerating it to a low impact velocity, or it will have to be able to withstand impact velocities in the order of 1-2 km/s. The first option will most likely require a propulsive system that will exceed the ~1 kg allocation by far. The second option will impose extreme structural requirements, beyond current technology as well as highly complex attitude issues.

The key challenge of the study was therefore to provide a design for the Europa microprobe (EMP) that would undertake useful science while meeting the severe mass budget of 1 kg. Chances of anything surviving impact at 2 km/s is practically zero: not even hardened military projectiles are generally expected to survive such impact velocities. In order to withstand these impact velocities, these designs only function because the projectile impacts along a specific axis. In this way the payload (power, electronics, etc) can be aligned along the direction of deceleration. In terrestrial designs, this is achieved by using atmospheric stabilisation. A probe that lands on an atmosphereless body faces two problems: decelerating to a survivable velocity and aligning itself to impact on a preferred axis so that hardening of the probe is effective. Several options were considered to provide the probe with attitude stabilisation, including gravity and active stabilisation. However, it was concluded that such options will consume so much of the mass budget that little or nothing will be left for the remote sensing payload. An additional complication is that the attitude control systems would require an innovative guidance system (e.g. sun or star sensor), which will need to find its reference star among the Jovian bodies (especially Jupiter itself).

Furthermore, as the knowledge of the Europan surface is very limited and is not sufficient to make a comprehensive selection of the landing site in terms of scientific merit as well as the suitability of the terrain for an impact. If the impactor would hit the surface at a grazing angle (e.g. due to a crevasse or a steep slope), the probe would have completely different impact conditions than the assumed normal impact vector on a flat surface. This would have major consequences for a directional probe, as it is designed for a specific impact attitude. Next to that, an unfortunate landing into a crevasse would greatly reduce communication capabilities with the orbiter.

As no realistic scenario was identified for a 1-2 km/s impact velocity, it was decided to relax the mass constraint by assuming that a propulsive system could be added, capable of reducing the impact velocity to 500-600 m/s. The required mass of a deceleration system without attitude control system is expected to be at least 5 kg (optimistic estimate). As shown in the mass budgets later on, the accommodation on the JEO of such a mass will be a major challenge. Since this is an increase of the dry mass of the JEO, the implication for the wet mass will be much higher due to the so-called snow ball effects: for example, an impactor system mass of ~10 kg would result in a propellant mass increase of ~30 kg, which would stretch the limit of the available propellant tank capacity and would require further strengthening of the spacecraft structure. This snowball effect would lead to an increase of ~50 kg in wet mass (assuming that larger tanks are not required), and would endanger the available mass resources for the remote sensing payload: in all likelihood the mass of the entire JEOMA system (including accommodation on the orbiter) must be detracted from the available payload mass.

Allowing for an impact velocity of ~500-600 m/s still leaves the problem of aligning the probe for impact. An active stabilisation unit would consume the mass budget, so an attitude independent design was suggested: a spherical steel shell. A spherical design allows the EMP to be entirely independent of impact attitude. However, it provides some other major design challenges. First, a hollow sphere is not the ideal shape for a device due to impact solid ice at 500 m/s. Most spheres will erode or collapse on impact. Consultation with weapons experts lead to a design which uses glass microspheres to fill the interior voids. Two key properties of the microspheres are that they are very good at dissipating shocks, and also at accommodating deformation by allowing components to move with respect to one another. The impact of the probe on Europa will cause a shock wave to propagate through the wall of the shell which will be released on the internal shell wall, potentially spraying the interior with high velocity shards of steel. The microspheres reduce this effect, increasing the chances of internal elements surviving. The EMP is also designed to deform to some extent, a critical feature of survivability for the probe.

Figure 18: Illustration of the Europa Microprobe concept [16]

Having identified a potential design for the probe, the study looked into the science that could be achieved using the system. One of the key questions is whether there is liquid water beneath the ice crust. If only a semi-solid slush exists, the chances of life forms existing, no matter how basic, become much smaller. A seismometer or geophone was proposed as the main instrument for this study. This is designed to 'listen' to the surface of Europa at very low frequencies (0-100 Hz). Seismic/acoustic events caused by cracking in the ice (due to tidal forces exerted by Jupiter) or by random meteor impacts, can propagate for thousands of kilometres in ice and water. If there is an ice/water interface then there will be 'echoes' caused by partial reflection of the seismo-acoustic waves at these interfaces. These can be captured and de-convolved to show the layers, their depths and constitution. Such techniques are routinely used terrestrially in fields such as oil prospecting and underwater warfare. To enhance the scientific package of the probe, other instruments were proposed that would meet the mass, power and volume budgets, including temperature sensors that could measure the thermal flow in the crust of Europa, accelerometers that could measure the deceleration and hence the surface hardness of the ice and strain gauges that could measure the deformation of the probe. Figure 18 provides an illustration of the payload concept. The impactor would require a considerable life time in order to measure the surface conditions, partly to allow for the dissipation of the thermal energy released at impact, but also because the tidal effects are a function of Europa's orbit around Jupiter (orbital period = 3.55 days) and finally the communication windows with the orbiter, which are limited by the ground track of the JEO and the limited power of the impactor's communication system.

Some improvements can be expected by better battery performance and the use of a Radioisotope Heating Unit (RHU). In this way the duration of the probe's lifetime could be significantly increased. Of course the AIV and launcher implications of the presence of RHUs need to be taken into account.

The indicative parameters for this probe are shown in Table 9:

Table 9: Indicative parameters for JEOMA [16]

Item	Unit	Value
Radius of payload cavity	cm	2.5
Remaining volume for payload	cm^3	65.5
Mass of payload components	g	133.2
Volume of payload components	cm^3	36.1
Remaining volume for micro-sphere packing	cm^3	29.3
Mass of packing	g	51.4
Mass of steel shell (1kg – payload – packing)	g	815.5
Volume of steel shell	cm^3	101.9
Diameter of probe	cm	6.8
Thickness of shell walls	cm	0.9
Mean power consumption of probe	W	4
Power density of battery	Wh/Kg	377
Power available	Wh	26.4
Probe lifetime	h	6.6

As explained, the microprobe study is at conceptual level: the presented figures are based on the allowable mass of 1 kg, and do not pretend to be a fully tested and feasible design, as much more analysis and testing would be required to understand the exact behaviour of the impactor. Furthermore it is unclear if the payload and communication system will survive the still very high deceleration loads experienced during impact.

As the study did not include any structural analysis (in the study the structural mass has not been based on strength calculations) a separate Finite Element (FE) analysis has been performed to obtain a first indication of the impact conditions of the sphere. Even if the results need further validation, as the computations must be cross checked with actual tests, the analysis -performed by Vorticity Ltd. (UK)- provided a clear indication of the extreme loads on the payload: the decelerations are in excess of 100 000 g. The following results are taken from [17].

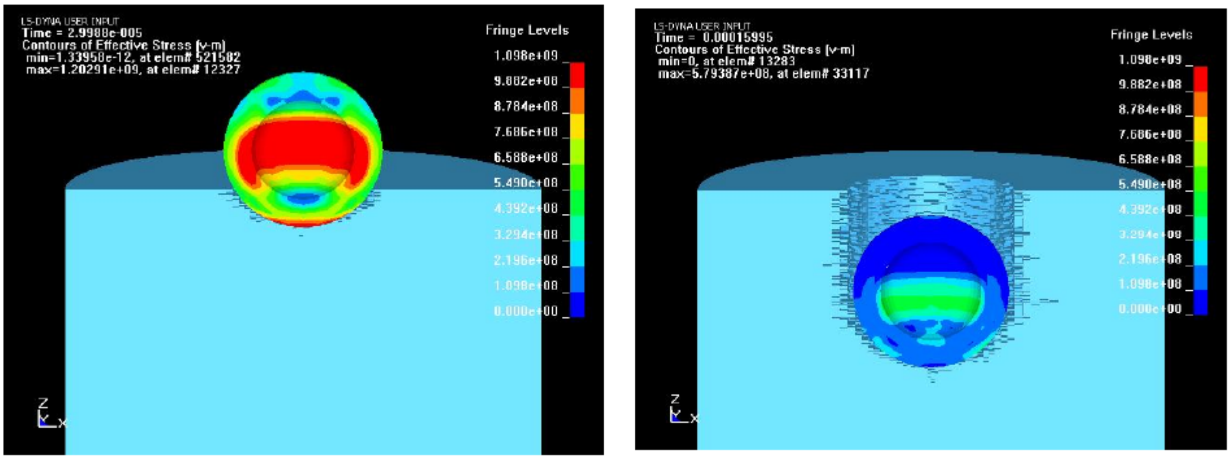

Figure 19: Typical result of FE simulation: two timesteps [17]

Acceleration profiles have been obtained for impact cases with varying angles of impact. Acceleration is reduced as impact angle becomes shallower. These acceleration profiles could be used for better definition of instrumentation requirements until more detailed investigations can be made. The study also revealed that a titanium shell impactor, sized to attain the same mass as a steel shell, produces higher deceleration loads than the steel probe as a direct result of having a larger surface area. A titanium shell sphere only serves to increase the survivability of the payload cavity volume during higher impact velocities– not reduce the loads into the structure.

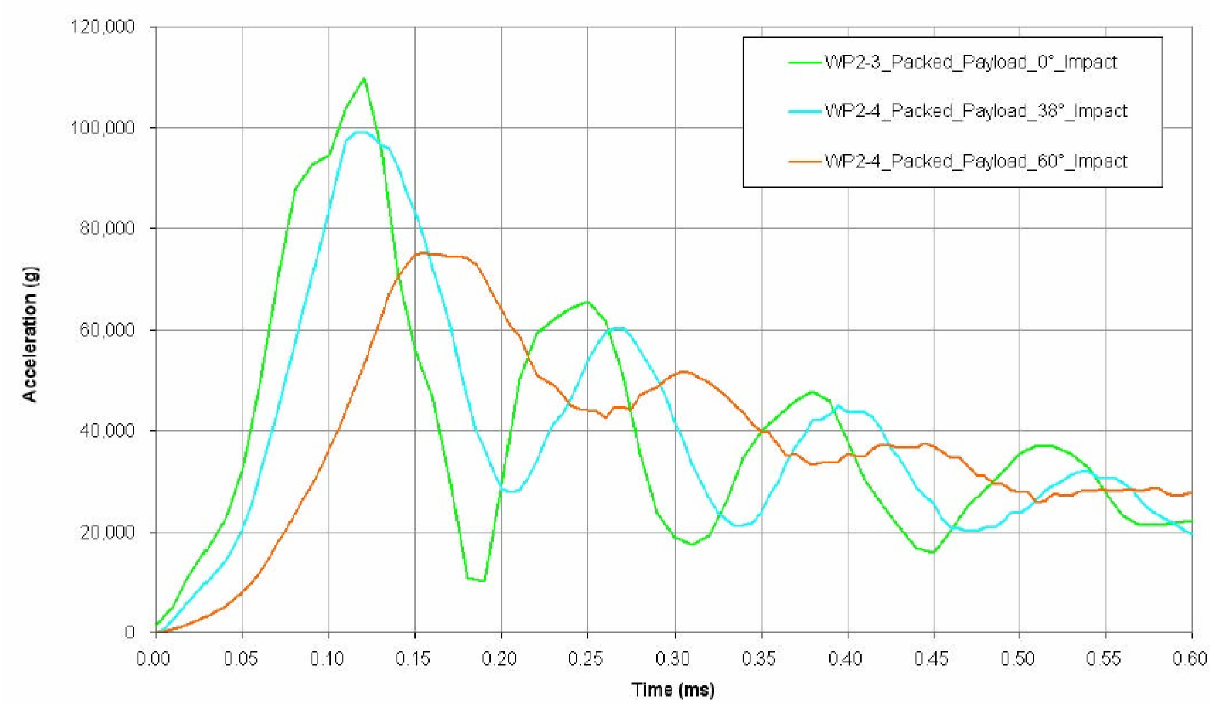

Figure 20: Payload acceleration profiles for different impact angles [17]

The shell is clearly unable to provide a sufficient level of energy attenuation to the payload. Indeed, the acceleration profile shows that the filler material (microspheres) fractionally delays the impact of the payload into the steel shell. This delay increases the differential velocity between the two bodies such that the acceleration at impact is increased. The limited stroke (~4 mm) available is insufficient for the microspheres to attenuate the payload energy significantly. It would require a more tailored combination of material properties and available stroke to attenuate the payload acceleration. Inappropriate combinations can actually lead to amplification: Preliminary dynamic analysis suggests that a stroke in the order of centimetres -with the current mass and velocity parameters- would be required. The microspheres cannot resist the motion of the payload sufficiently prior to deforming. It is clear from the elementary analysis conducted that the stroke / failure properties of the filler material need to be tailored to the precise impact acceleration conditions and mass of the payload. It can therefore not be said that "microspheres are a good attenuator for high velocity impacts"; it is only possible to say that they are appropriate for certain conditions.

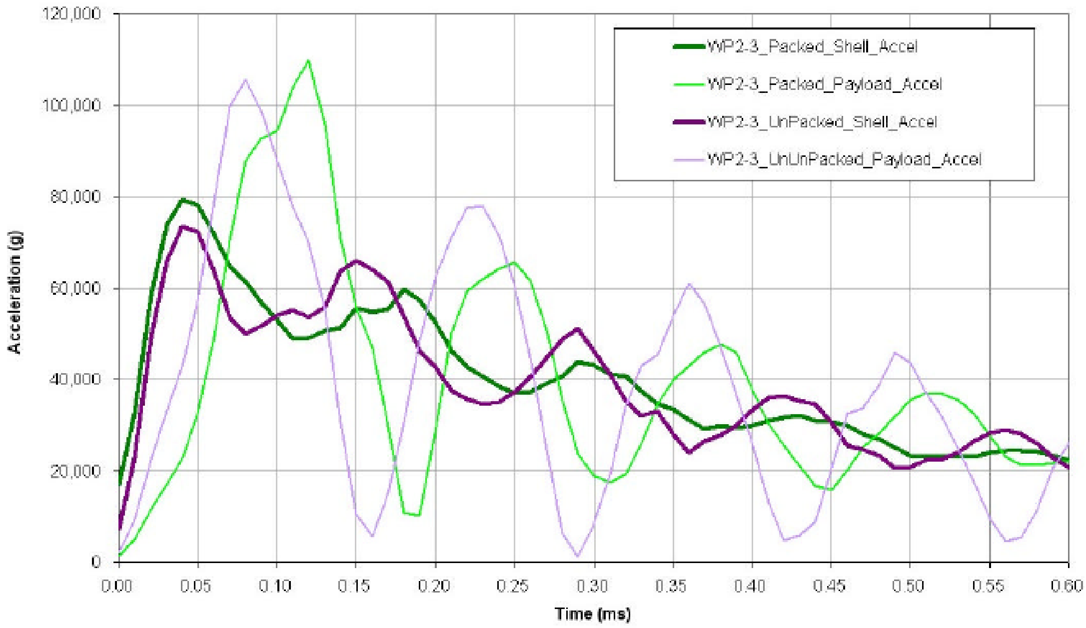

Figure 21: Acceleration profiles with and without packing for a normal (i.e. 90) impact at 500 m/s [18]

Next to these excessive loads on the sphere itself, the motion of the payload during an impact also raises questions on the survivability of any antennae structure connections to the payload from the outside surface of the sphere through the steel shell. It is believed that, even with the provision of longer flexible connections, the shearing action caused by the flow of the current selected packing material would limit the survival of any connection cable from the outer shell.

Despite the interesting results of the JEOMA study, it must be concluded from these inputs that a spherical impactor with low mass is not feasible with current technology and even the current impactor configuration would exceed the 1 kg mass allocation goal by far. A different concept must be considered to further assess whether a low resource Europa impactor can be considered at an acceptable reliability and risk.

JME-3.3.2 The Europa Microprobe In-Situ Explorer (EMPIE)

In view of the identified problem with the "1 kg" impactor, a second study was initiated to understand the requirements of a dedicated microprobe payload, including a descent platform. This led to the EMPIE study, performed by TTI (E). The results are now summarised and taken from [18]. Due to the severe payload mass constraints, in this concept the JEO payload will be replaced by the EMPIE system, with the exception of the imaging systems and some key instruments required for the mission.

The EMPIE concept is essentially divided into two parts: the Common Module (CME), a carrier vehicle to perform the common functions and manoeuvres of the descent; and the EMP Microprobes, which carry the science equipment and the related subsystems necessary to perform the scientific investigations on the surface.

The study baseline included 4 microprobes in the CME. The baseline configuration of the system is driven by the requirements of the Primary Propulsion, which is based on a solid propellant rocket and accounts for 50% of the EMPIE mass. The following figure shows this baseline configuration.

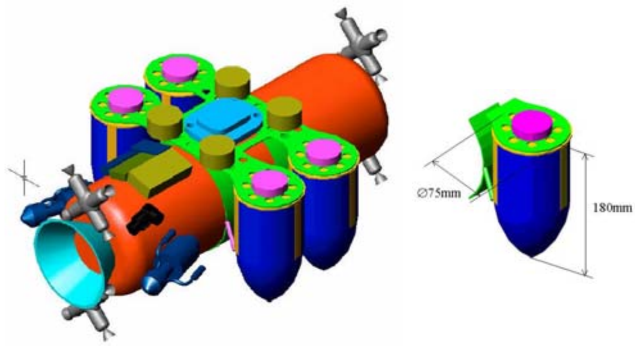

Figure 22: EMPIE baseline configuration [18]

The next figure shows the individual subsystems, which are now briefly summarised:

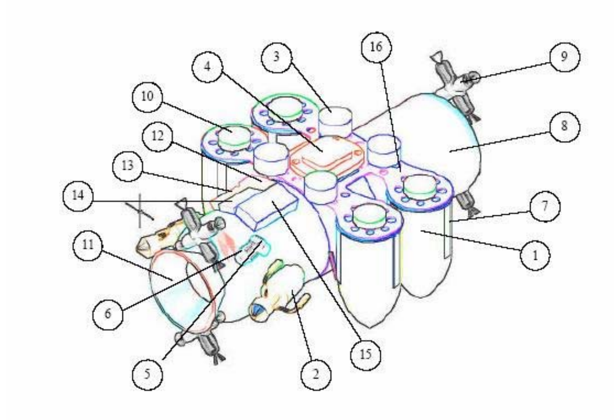

Figure 23. EMPIE subsystems [18]

1. EMP Probes (x4) 2. De-orbit Thrusters (x2) 3. EMPIE Ejection Mechanism (Spring x4) 4. Inertial Measurement Unit (IMU) 5. Jupiter Sensor 6. Europa Horizon Sensor 7. EMP Probe Ejection (Guiding System x4) 8. Primary Propulsion System (Tank) 9. ACS Thrusters (x4) 10. EMP Probe Ejection (Spring x4) 11. Primary Propulsion System (Nozzle) 12. Power Subsystem (Microbatteries) 13. Power subsystem (PMAD) 14. OBDH 15. Communications Subsystem 16. EMPIE structural support for the EMP probes

The Primary Propulsion system has the objective of performing the retro-thrust braking manoeuvring, which is used to cancel the horizontal velocity of the EMPIE over the Europa surface. It is based on a solid propellant rocket and provides a Delta−V of 1442m/s, with an average thrust of 221N and an I_{sp} of 280 s for the solid propellant. It was assumed that this approach will result in the complete cancellation of the horizontal velocity and will define the incident angle of the free fall trajectory. This incident angle will be largely responsible for the survival of the EMP at impact, therefore the global performance of the thrust is the most critical parameter for the system, and it must be within +/-1% of the optimum impact angle (i.e. perpendicular to the surface). Considerations in the JEOMA section have already shown that this is not straightforward.

The Secondary Propulsion system performs the de-orbit of EMPIE from the initial circular orbit to the descent elliptical orbit. The Delta-V=31m/s impulse is provided by two hydrazine thrusters with a 1N thrust. Extra propellant in this system is then available for correction manoeuvres after the retro-thrust operation.

Attitude control is performed with the use of a Europa Horizon Sensor, a Jupiter Sensor, and a miniaturised Inertial Measurement Unit (IMU). The design of the optical sensors is complicated by the high radiation levels which will likely lead to necessary reductions in the FOVs (10-15deg) and moderate accuracies (1-2 degrees). The attitude corrections are performed by 4 pairs of cold gas micro thrusters.

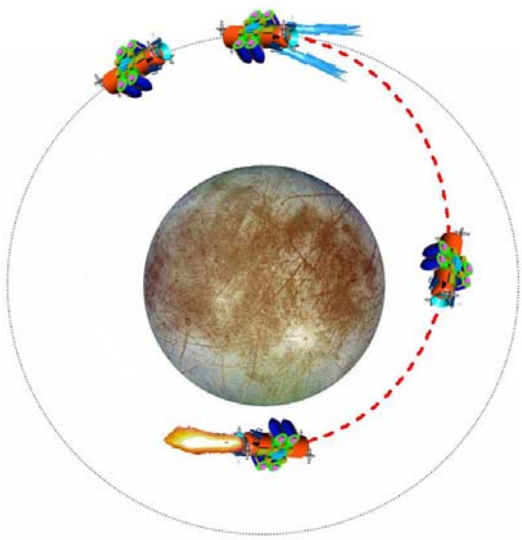

Figure 24. EMPIE Descent sequence: de-orbit, free descent and start of the retro-thrust [18]

Ejection of the EMPIE system from the Orbiter is performed by a mechanical ejection mechanism consisting of four springs. The four EMP probes are located in the CME within a structural frame, which includes the probes' ejection mechanism and guiding system.

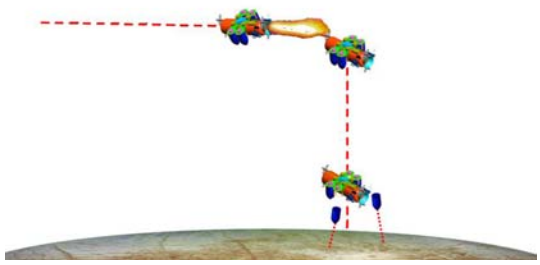

Figure 25. EMPIE Descent sequence: retro-thrust manoeuvre, free fall and ejection of the microprobes [18]

The structure of the probes is based on a Ti Alloy structure 2mm thick, with a reinforced nose, able to withstand the loads at impact. The total mass of the structure accounts for 35% of the total mass of the EMP. The study assumed an impact velocity of 364m/s, and concluded that this would lead to a deceleration load of 8600 g. The much lower deceleration is due to the pointed shape, provided that the impactor hits the surface at the correct point and that the assumed deceleration profile in the ice is maintained. No FE analyses have been performed in this case to corroborate the predicted g-levels, hence the g-levels and the required structural mass are likely to increase.

Thermal control for the probes, under the critical conditions of the science phase, is based on the use of a Radioisotope Heat unit for the heat generation, as well as passive and active control elements. As a first estimate a value of 2.5W of heat was estimated to be necessary to keep the system in operation at the conditions of the surrounding ice layer (~100K) during the science phase. The inclusion of RHU will incur significantly higher cost due to limitations in AIV procedures and measures required by the launch authority. Also the availability of RHU will have to be assessed.

Both the CME and the EMP probes are provided with miniaturised communications subsystems for the descent tracking and the science phase respectively. An analysis of the link budgets shows the viability of the communication in all cases, with a 1.7kbps data transmission rate for the probes during the science phase. A polar landing is recommended due to the link visibility (Orbiter to the probe) of 6%. The Equatorial landing has a visibility of less than 1%, which is a great limitation in view of the limited lifetime of the orbiter (~60 days).

OBDH subsystems for the CME and the EMP probes require on highly miniaturised, radiation hard, failure tolerant electronics.

The power supply for the CME and EMP probes will be provided by state-of-the-art micro-batteries. Maximum power needs for the EMPs vary between 1.7W and 2.7W, depending on the payload. The microprobes will have their lifetime limited by the total energy of the batteries, to a duration of about 4 to 5 days. The alternative of the use of a small RTG for power generation has been studied to increase the mission lifetime. However, next to the previously mentioned problems, the immature development of high efficiency (~20%), low power thermal to electric converters, together with the complexities for the thermal management, make this option less favourable.

The EMPIE mission scientific interest relies on two aspects: the possibility of performing in-situ subsurface measurements and the advantages from the microprobe approach to distribute the Payload on different locations. The three main science objectives for EMPIE are: a Compositional Analysis of the surface, a geophysical and geological survey, and a biological analysis. However, the limitations of the mission will severely drive and constrain the science capabilities and performances of the instruments on board the probe.

These limitations come mainly from two factors:

- The *low level of resources* available. The System design of EMPIE allows a mass of approximately 350grams for the Payload, while the size and power/energy are also limited. Miniaturisation and high level of integration (e.g. MEMS or 3D-MCM) are necessary for the instruments and associated electronics. For the science data rates, a maximum of 9Mbits per day of operation is possible, for a polar landing.

- The *impact conditions* and *access to the surrounding ice*: Hard impact conditions will limit the performances of the instrumentation, and the capacity to access the surrounding ice to study. Compact instruments with indirect access to the ice samples are preferred, in order to simplify the operations of the science measurements.

The current baseline foresees the implementation of two different Payload models, to distribute in the four EMP probes. The first Payload model will perform a Compositional Analysis and is based on a miniaturised ATR Spectrometer with the inclusion of specific environmental and biological sensors, to measure pH and redox status and identify specific biogenic compounds or biomarkers. The second Payload model will perform a Geological Survey, based on a micro seismometer with the inclusion of Physical Sensors to measure the characteristics of the surrounding ice (the instruments are based on current developments for miniaturised instruments and previous missions, see [18] for references).

The total mass of the CME and 4 EMP probes, including 20% subsystem margin, would result in a mass of approximately 25 kg, not including the support system on the orbiter. An additional 20% system margin would lead to a total mass of 30 kg, assuming the deceleration loads will be in line with the assumptions of this feasibility study. This mass range equals the payload slot for the entire Europa orbiting spacecraft and would therefore not allow other instruments on board the spacecraft. This mass is likely to increase further due to required support systems on the orbiter, including instrumentation (comms, release mechanisms, manoeuvres for targeting, imaging and tracking systems, etc.). This approach is not recommended in the reference frame of JME, as the global investigation of Europa would not be possible.

JME-3.3.3 General comments on impactors

The two presented studies have shown that an impactor for an atmosphereless body as Europa is very complex, due to the high impact velocity and the problem in attitude control. This is complicated further by:

- **The insufficient knowledge of the actual impact site**, due to the limited imaging resolution of Europa (the highest resolution image is obtained by Galileo and is 1.8km wide with a resolution of 6m/pixel)

- **The Orbiter survives only 60 days**. Therefore, the lander/impactor will not have a mapped landing site prior to release: most science data will be available long after the JEO mission has finished, due to the required storage of JEO data on the JRS: it can take more than a year before that data is received. What is needed is surface and subsurface mapping before the release of any Lander to a precise location

- **Environment constraints:**
    - Radiation Protection: shielding, Hi-Reliability components and redundancy are mass drivers
    - Radiation environment: risk of Single Event Upsets and Electrostatic Discharge
    - Limited lifetime due to radiation as well as low temperatures
    - Power needed to heat probe after impact: RHU may be required, complicating AIV and launcher procedures. These points will drive cost and schedule
    - Planetary Protection of Category-IV is applicable to Europa. This imposes severe limitations on AIV as well as on the materials

- **Technical constraints:**
    - Limited power available for heating, payload and communications
    - Impact at high velocity will result in deformation of the impactor, complicating payload access to the outside
    - Impactor localisation is difficult due to the uneven terrain, the short life time of the impactor and the ground coverage limitation of the orbiter
    - In case of high velocity impact, the (often sensitive) payload and communication system will suffer. Strengthening of the payload will require more mass and is likely to reduce sensitivity of the instruments, hence reducing the science output
    - Significant technology development is required to make such systems feasible, irrespective of high or low velocity impact
    - Additional engineering margins are required when the properties and topography of the surface unknown
    - Lander/impactor system mass will be directly deducted from the orbiter payload. In view of the mass limitations of this concept, an impactor system will in the best case use a large part of the JEO payload (typically more than 60%). In view of the high uncertainty, it is also possible that the entire payload must be dedicated to the impactor, as shown by the EMPIE study.

- A soft lander has not been presented here, but using the study performed for the Bepi Colombo lander concept, a mass in excess of 500 kg is expected. This clearly is not at all compatible with the JME concept

- **Instrumentation constraints:**
  - Access to subsurface required to measure areas unaffected by radiation
    - This is a mass and energy driver, technically complex
    - Melting is potentially difficult due to unknown impurities that can accumulate underneath the probe

  - Contamination of ice by propulsion system due to the controlled landing:
    - This can result in non representative measurements of exobiology instrumentation

To conclude, a controlled landing is difficult and expensive. This would require a large and expensive mission, far beyond the JME concept.

Irrespective of the landing method, a clear understanding of landing zone is required. High resolution data must be analysed first, which cannot be done in real time in the JME concept.

A lander, especially for an atmosphereless body, incurs very high risk for both a controlled landing as a high velocity impact. The risk of failure must be properly assessed and the result must be used in the mission design trade off: in other words, is the risk acceptable and can the required resources be justified. Finally the risk for such a mission must be made clear to all parties involved, including the public: it shall be made clear that such a mission is very challenging. High risk is inherent to the concept and success is not at all guaranteed.

JME-4 MISSION ANALYSIS

This chapter gives a summary of the performed mission analysis for this mission. After taking various options into consideration (all-chemical and a hybrid option consisting of electrical and chemical propulsion), the all-chemical solution was selected, mainly because of the lower spacecraft complexity lower cost and higher reliability.

JME-4.1 Launch

The spacecraft composite is assumed to be launched from Kourou with a Soyuz-Fregat 2-1b in the 2015-2025 timeframe. To maximise the payload in orbit, the Soyuz will place the composite in a highly elliptical GTO orbit, rather than performing a direct escape. This allows for a significant increase in payload compared to performing the escape manoeuvre with the Fregat upper stage. With this strategy the launcher will be able to deliver a payload in excess of 3000 kg, after which the spacecraft composite will perform the Earth escape manoeuvre using its own propulsion system. The following scenarios refer to a fully chemical propulsion approach with gravity assists.

JME-4.2 The Transfer Phase

The transfer from Earth to Jupiter will be achieved by performing a series of Gravity Assist Manoeuvres (GAM). Mission analysis revealed that in general for an all-chemical propulsion system, the best performance is obtained with a Venus-Earth-Earth Gravity Assist (VEEGA) sequence. The best performance in this case is a trade-off between the minimal Delta-V manoeuvres and the shortest transfer time.

Previous versions of this document showed a reference launch date of 2017. However, this trajectory has been modified during reassessments of the mission concept, due to unexpected increases in the propulsive manoeuvres during the Jovian tour. By revisiting the launch dates new solutions were found to further decrease the Delta-V's of the reference mission scenario. An overview of these refined opportunities is found in Table 10, which provides a summary of transfers between 2017 and 2024 and shows the effect of the launch date on the total Delta-V and the transfer duration

Table 10: Summary of Transfer characteristics [19]

Launch Dates	Jan 2017	Jan 2017	Mar 2019	Mar 2020	May 2021	Jun 2023	Aug 2024
Transfer duration (years)	5.75	6.75	7.1	6	7.2	6.6	7
Departure declination	16°	18°	-3°	4°	-14°	-20°	-20°
Departure Vinfinity	3.05 km/s	3.32 km/s	3.114 km/s	3.449 km/s	3.569 km/s	3.3 km/s	3.481 km/s
LEOP correction	~35 m/s						
Apogee raising ΔV	692 m/s (400000km apogee)						
Gravity Losses (~2%)	~15 m/s						
Inclination change	0 m/s (all departure declinations within +/-20°)						
Escape ΔV	495 m/s	570 m/s	512 m/s	608 m/s	645 m/s	564 m/s	618 m/s
Gravity Losses (~5%)	~25 m/s	~29 m/s	~26 m/s	~30 m/s	~32 m/s	~28 m/s	~31 m/s
Deep Space ΔV	262 m/s	88 m/s	20 m/s	0 m/s	0 m/s	0 m/s	0 m/s
GA Navigation ΔV (15m/s per GA)	45 m/s (VEE)	45 m/s (VEE)	60 m/s (EVEE)	45 m/s (VEE)	45 m/s (VEE)	45 m/s (VEE)	45 m/s (VEE)
Jupiter approach navigation ΔV	20 m/s						
Approach Vinf	6.09 km/s	5.97 km/s	5.57 km/s	5.58 km/s	5.48 km/s	5.6 km/s	5.85 km/s
Capture ΔV via Io GA. Perijove = 3.5Rj, Apojove = 420Rj	~530 m/s	~525 m/s	~445 m/s	~445 m/s	~440 m/s	~450 m/s	~500 m/s
Gravity Losses (~3%)	~16 m/s	~16 m/s	~13 m/s	~13 m/s	~13 m/s	~13 m/s	~15 m/s
Perijove Raising Manoeuvre	~375 m/s						
TOTAL	~2510 m/s	~2410 m/s	~2213 m/s	~2278 m/s	~2312 m/s	~2237 m/s	~2346 m/s
Mass in Jupiter initial orbit	~1295kg	~1340kg	~1430kg	~1400kg	~1385kg	~1420kg	~1370kg

The following overview shows the current baseline for the JME reference scenario, for which the 2019 launch opportunity is selected. The used Delta-V budgets shown for this baseline are different from those shown in Table 10, because the reference transfer scenario assumed the worst case in the three-week launch window. The reference transfer scenario was established during the CRETE study performed by SCI-AP and CDF. The following overview is taken from this study [11]. The 2019 launch case is not the best opportunity (see June 2023), but this allows for more flexibility in the launch date (selecting the best case would leave no alternative launch opportunities, which is not a good design policy).

The 2020 launch with the subsequent VEEGA transfer has a moderate departure and arrival velocity and a low Deep Space Manoeuvre (DSM) budget. The single major disadvantage is the high departure declination, which penalizes the launcher capability, as the Soyuz rocket has to launch into an equally high initial inclination. The workaround is to launch exactly one year earlier, which allows leaving the Earth at a much lower declination, performing an Earth GAM one year after launch, and using the GAM to deflect the direction of the outgoing asymptote outside of the ecliptic plane. This strategy was already effectively used in the Rosetta mission. The strategy adds one year to the mission duration and advances the launch by one year, but does not require a plane change manoeuvre and increases the payload delivered to Jupiter.

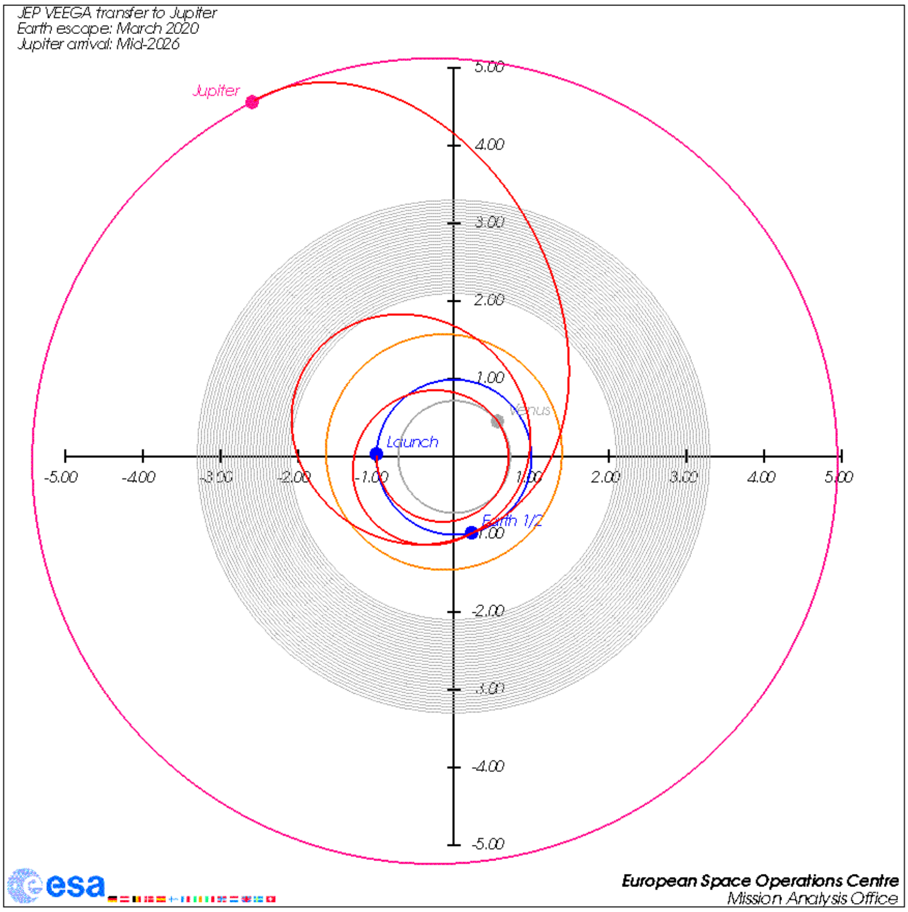

Figure 26: VEEGA 2020 Transfer

Figure 26 shows the original transfer from April 2020 to mid-2026, when Jupiter is reached. The added one year heliocentric portion is shown separately in Figure 27. Launch and EGA are exactly at the same location because they are separated by exactly one year. The departure hyperbolic velocity is the same in 2019 as it would have been for launch and escape in 2020, but the declination is far more moderate. This allows a higher mass to be launched, avoiding the penalty of launching into a high initial inclination.

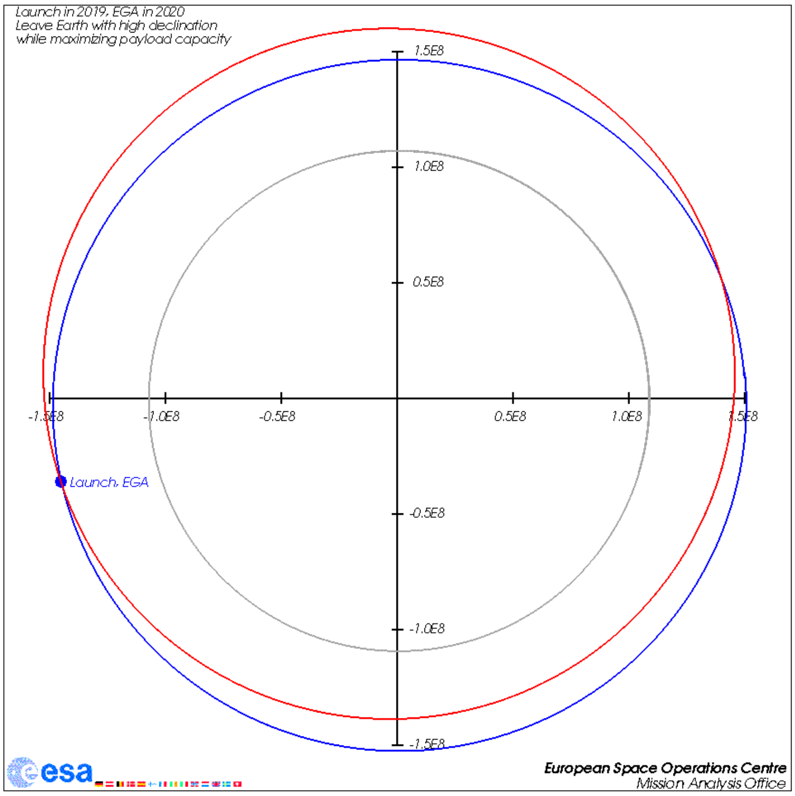

Figure 27: Preceding 1-Year Heliocentric Orbit Following Launch in 2019

Table 11: summarizes the entire transfer, using the worst case in the three-week launch window.

Table 11: Summary of worst-case EVEEGA transfer to Jupiter

Earth Escape	4/4/2019
Hyperbolic departure velocity [km]	3.17
Departure declination [deg]	-22
Earth Swing-by 1	4/4/2020
Hyperbolic velocity [km/s]	3.17
Outgoing declination [deg]	43
Venus Swing-by	11/9/2020
Earth Swing-by 1	18/7/2021
Earth Swing-by 2	19/7/2023
DSM budget [m/s]	32
Jupiter arrival date	28/7/2026
Hyperbolic arrival velocity [km/s]	5.704
Arrival declination [deg]	8

With launch into a GTO-like orbit inclined by 22 deg and a four- or five burn escape strategy using the onboard propulsion system to increase the apogee to a maximum of 300,000 km (this constraint is imposed to avoid excessive detrimental perturbations through the lunar gravitational attraction) the payload mass into escape is around 1985 kg, starting out from a launch mass of 3090 kg, from which an assumed adapter mass of 45 kg is subtracted.

The next table provides the details on the Gravity Assist conditions during the EVEEGA transfer

Table 12: Minimum fly-by altitudes of Earth and Venus during EVEE GAM

Planet	Date	Closest approach at fly by (km)
Earth (1)	4/4/2020	13 679
Venus	11/9/2020	14 766
Earth (2)	18/7/2021	1 427
Earth (3)	19/7/2023	478

Finally Table 13 provides a summary of the required manoeuvre budget for the transfer phase:

Table 13: Manoeuvre budget for the transfer phase

Manoeuvre	Delta-V [m/s]
Apogee Raise	706
Inclination change	0
Earth escape	733
VEEGA+DSM	122
Total	1561

JME-4.3 The Jovian Tour

After the transfer from Earth to Jupiter, the spacecraft must leave its elliptical orbit around the Sun, to achieve an orbit around Jupiter. Once the spacecraft composite arrives in the vicinity of Jupiter, a capture manoeuvre must be performed. To do this, a considerable change in velocity magnitude and direction is required. Two fundamental strategies can be applied:

- Acceleration in deep space to reduce approach speed at Jupiter
- Reliance on (several) manoeuvres and GAM within the Jovian System to capture from high-speed approach.

In general, direct manoeuvres require large amounts of propellant and tend to take less time when compared to gravity assist manoeuvres and vice versa.

Once captured, the two spacecraft will tour of the Jovian System, performing a series of GAM at the Galilean moons (see Figure 28 for the capture manoeuvre principle using gravity assist manoeuvres or GAM). Finally, one or more impulsive manoeuvres will insert the spacecraft into their respective final orbits.

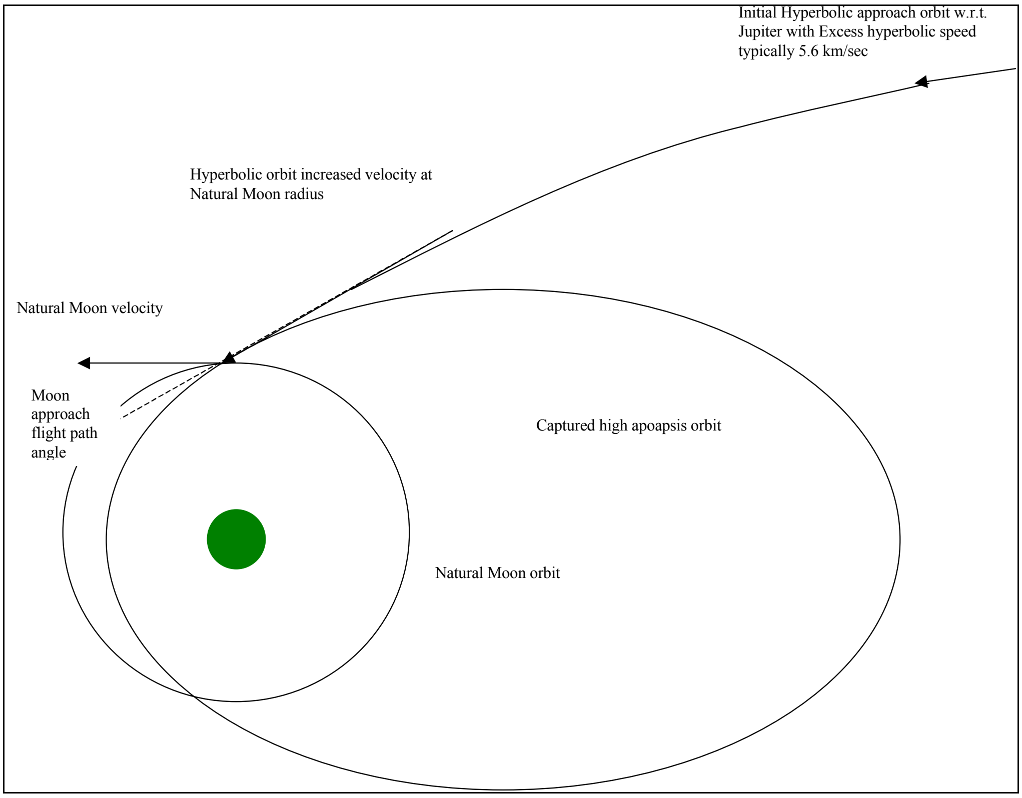

Figure 28: Principle of planetary capture using GAM at a natural moon [20]

The tour strategies differ considerably for the two spacecraft. Although both spacecraft stay close by during the initial part of their Jovian tours, the final part will be completely different: The JEO will target a 200km circular orbit around Europa, while the JRS will target a highly elliptical Jupiter orbit, outside the main radiation belts, to reduce radiation exposure. The choice of the final orbit strategy is influenced by required Delta-V, eclipses, communication links with JEO and radiation dose.

JME-4.3.1 ORBIT INSERTION AND INITIAL ORBIT CAPTURE

The initial conditions correspond to those derived in the CRETE CDF study [11]. A short summary is given in Table 14. The resulting mission analysis can be found in [7]

Table 14: Summary of arrival conditions

Arrival date	**[dd/mm/yyyy]**	28/07/2026
Incoming infinite velocity magnitude	**[km/s]**	5.71
Declination w.r.t. Jupiter equator-fixed frame	**[deg]**	8
Azimuth w.r.t. Jupiter equator-fixed frame	**[deg]**	21
Io fly-by altitude	**[km]**	300
Capture Orbit (1 R_J = 71 492 km)	**[R_J]**	5 x 400

The Jupiter tour phase begins with an Io GAM prior to the actual Jovian Orbit Insertion (JOI) burn to reduce the required delta-V for the insertion. The previous version of this document assumed a Ganymede GAM. However, in view of the required reduction in propellant mass, an Io GAM has been base-lined: When compared with a Ganymede swing-by, it is beneficial in terms of Delta-V, however the drawback is that radiation dose will increase significantly. At arrival both spacecraft fly stacked together until the Perijove Raising Manoeuvre (PRM). In the meantime, all manoeuvres are performed by the JRS. Then the separation is performed and each spacecraft can follow its own route to the final orbit.

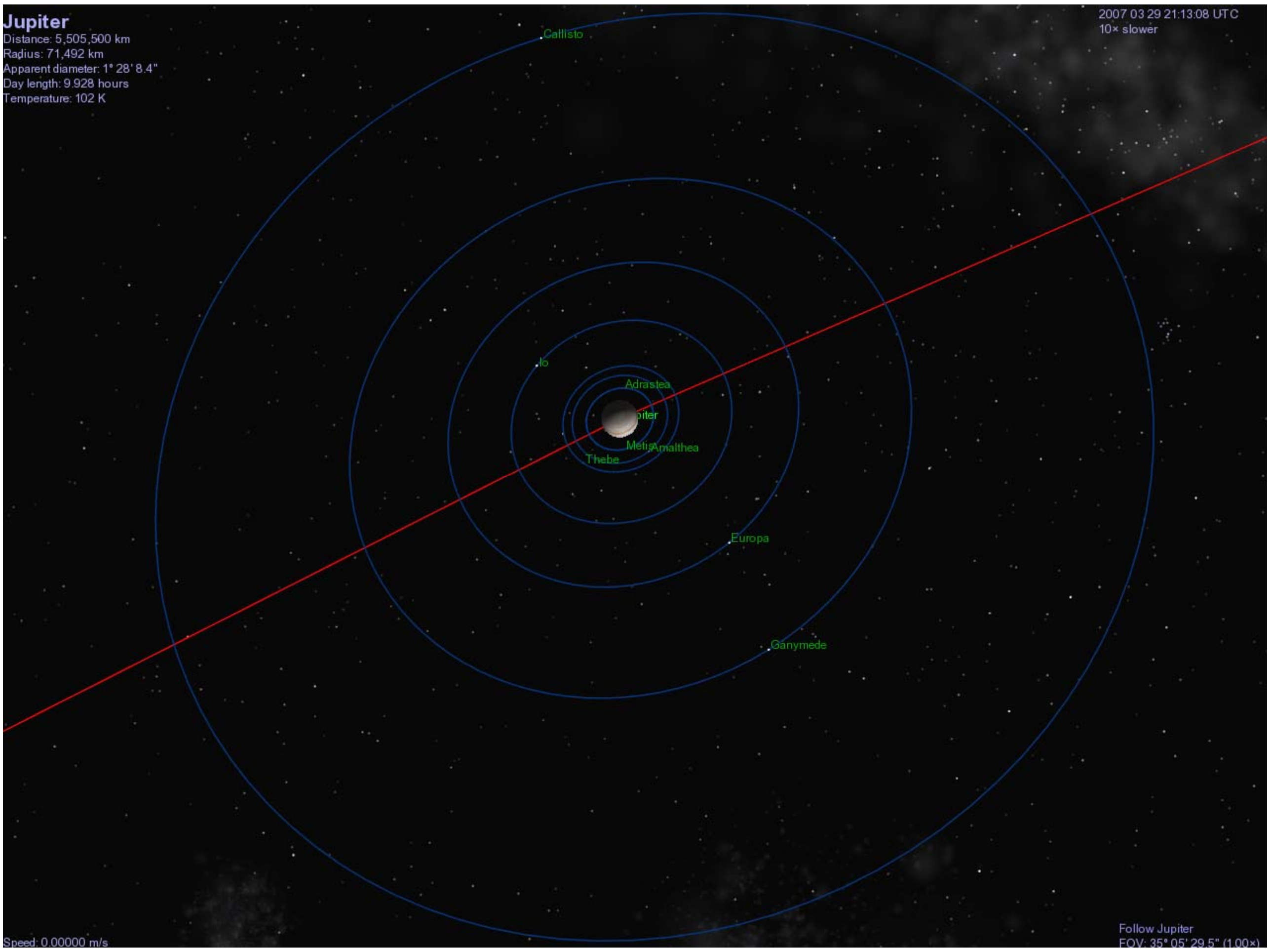

Figure 29: The Galilean Moons [Image generated using Celestia software]

Io swing-by – Enhanced JOI

This swing-by is performed to reduce the JOI. It is executed at the minimum altitude, i.e. 300 km, to maximise its efficiency. The minimum perijove radius for JOI is set to 5 R_J. At this distance, the perijove radius is already well below the Io orbit and thus in a region of extremely high radiation exposure. Any further reduction would slightly reduce the JOI size but further increase the inevitable charged particle dose, despite the high velocity and relatively short time spent in this low region. A better alternative is that the JOI can be minimized by choosing a high apojove radius. However increasing the apojove also increases the PSM: a trade-off between these variables led to choose an apojove of 400 R_J.

Perijove Raising Manoeuvre (PRM)

After the JOI, the spacecraft will be in a highly eccentric orbit with perijove and apojove radii of 5 and 400 R_J. This orbit must be changed to avoid that the spacecraft will again enter the very high radiation belts at pericentre: the pericentre must be raised to reduce the encountered radiation dose. As a result the JRS will retarget the composite before apojove in such a way that the two spacecraft will perform a Ganymede GAM at the pericentre (see Figure 30). As Ganymede is the most massive of the Galilean moons, it will provide the maximum benefit when performing gravity assist manoeuvres. As Ganymede is situated at a distance of 15 R_J from Jupiter, it will be outside the main radiation belts, which end in the vicinity of Europa.

Perijove Stabilization Manoeuvre (PSM)

The objective of the PRM is to raise the perijove to limit the radiation dose for the next revolutions. The solar perturbation tends to decrease the perijove, further necessitating the Perijove Stabilisation Manoeuvre at the apojove. The size of this manoeuvre for the 2026 case is 200 m/s [21]. After this manoeuvre the two spacecraft are finally separated.

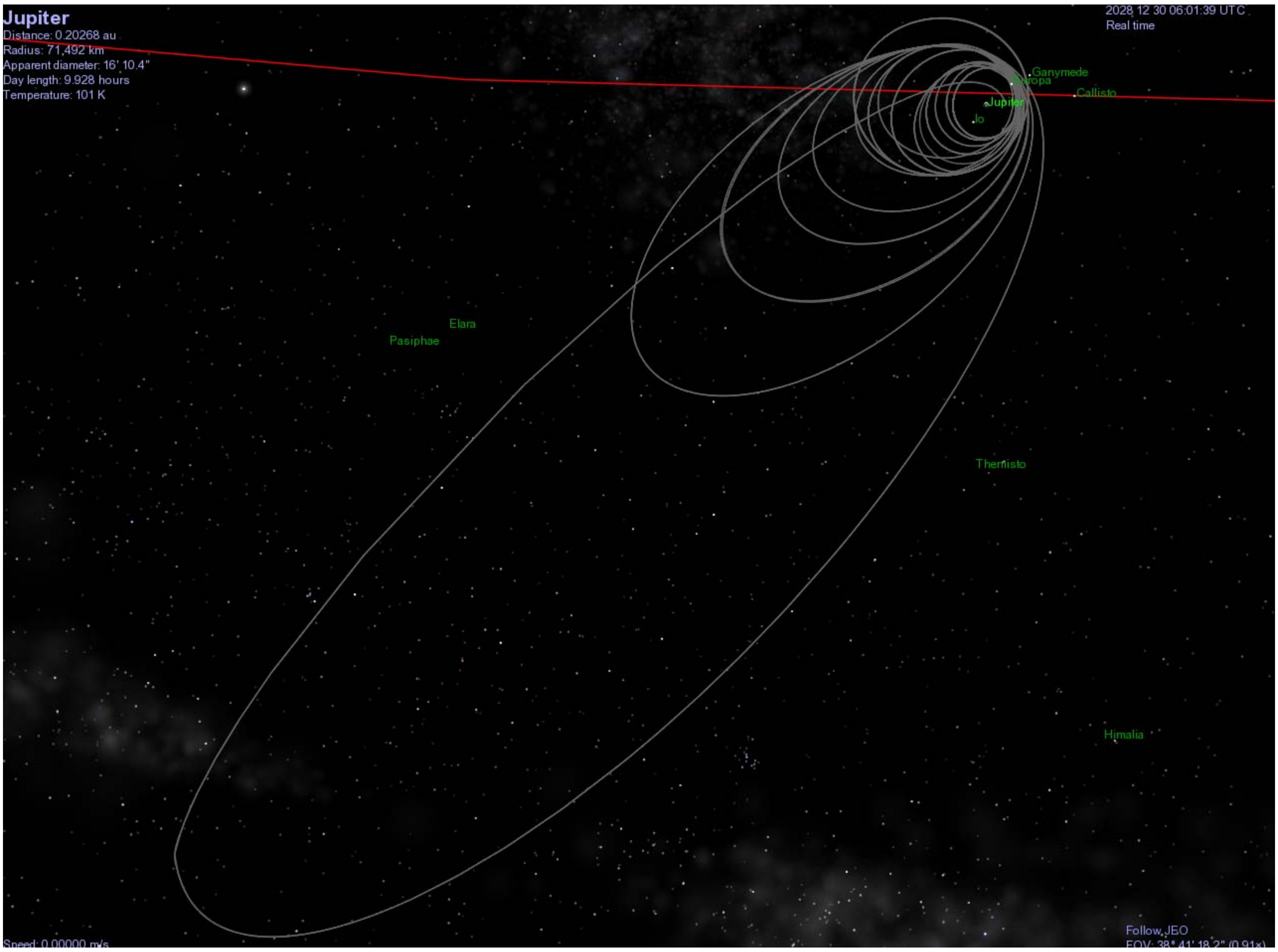

Figure 30: Overview of the Jupiter capture orbit and ensuing GAMs of the Galilean moons [image generated using Celestia software]

The following sections on the JRS and JEO tours are a summary of the mission analysis work performed by ESOC and are based on [7].

JME-4.3.2 THE JRS TOUR

The aim of the JRS tour is to achieve the target orbit, which is 3:1 resonant with Europa, near equatorial and with a minimum perijove of 11 R_J. This has to be done with a minimum expenditure of propellant within acceptable radiation dose limits. In the end the orbit is a compromise between minimisation of the Delta-V and the encountered radiation dose.

Tour Design

The following approach led to the final JRS tour design:

- Minimizing the radiation exposure precludes swing-by's at Europa. The tour will therefore use Ganymede and Callisto swing-by's exclusively
- JOI takes place at a low altitude of 5 R_J. The PRM is performed four months later and raises the perijove radius almost up to the Ganymede orbit. The Perijove Stabilisation Manoeuvre takes place at the apojove
- Following the initial capture orbit, the spacecraft performs six Ganymede swing-by's, each reducing the orbital period and decreasing the apojove considerably
- At the end of the tour there are two consecutive Callisto flybys. In this phase, the perijove radius is raised to meet the constraint on the minimum perijove radius (11 R_J)
- The final Ganymede flyby is used to meet the resonance constraint. The resulting orbit period is 10.5 days

Design Outcome

Figure 31 shows the initial decrease in period. In the right-hand diagram, the peri-/apojove radii (blue and red respectively) are displayed. The decrease in the apojove in the initial series of Ganymede swing-by's is clearly visible. Note that the axis of the radii is logarithmic.

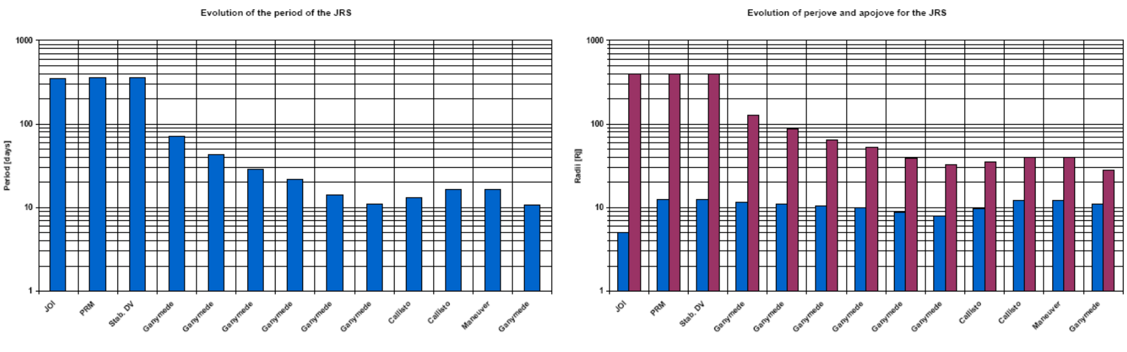

Figure 31: Orbital Period and Apo-/Perijove Radii for JRS Tour

The top plot of Figure 32, shows the electron dose (assuming 4 mm aluminium shielding) absorbed by the spacecraft in the different phases of the tour. The initial dose of 43 krad is received up to the first Ganymede swing-by, mostly during Jupiter approach, the Io swing-by and the first perijove with the JOI. During the following phases most of the additional radiation is absorbed every time the spacecraft dips below the Ganymede orbit. During the final Callisto flybys, the perijove radius is close to 10 R_J where there is a local maximum of radiation dose. This explains the high dose received while the perijove increases. The bottom plots provide the proton dose as well as the electron fluence. The main difference with the electron dose is that JOI strongly contributes to the received radiation dose. The radiation colleted during the operational phase is not yet included.

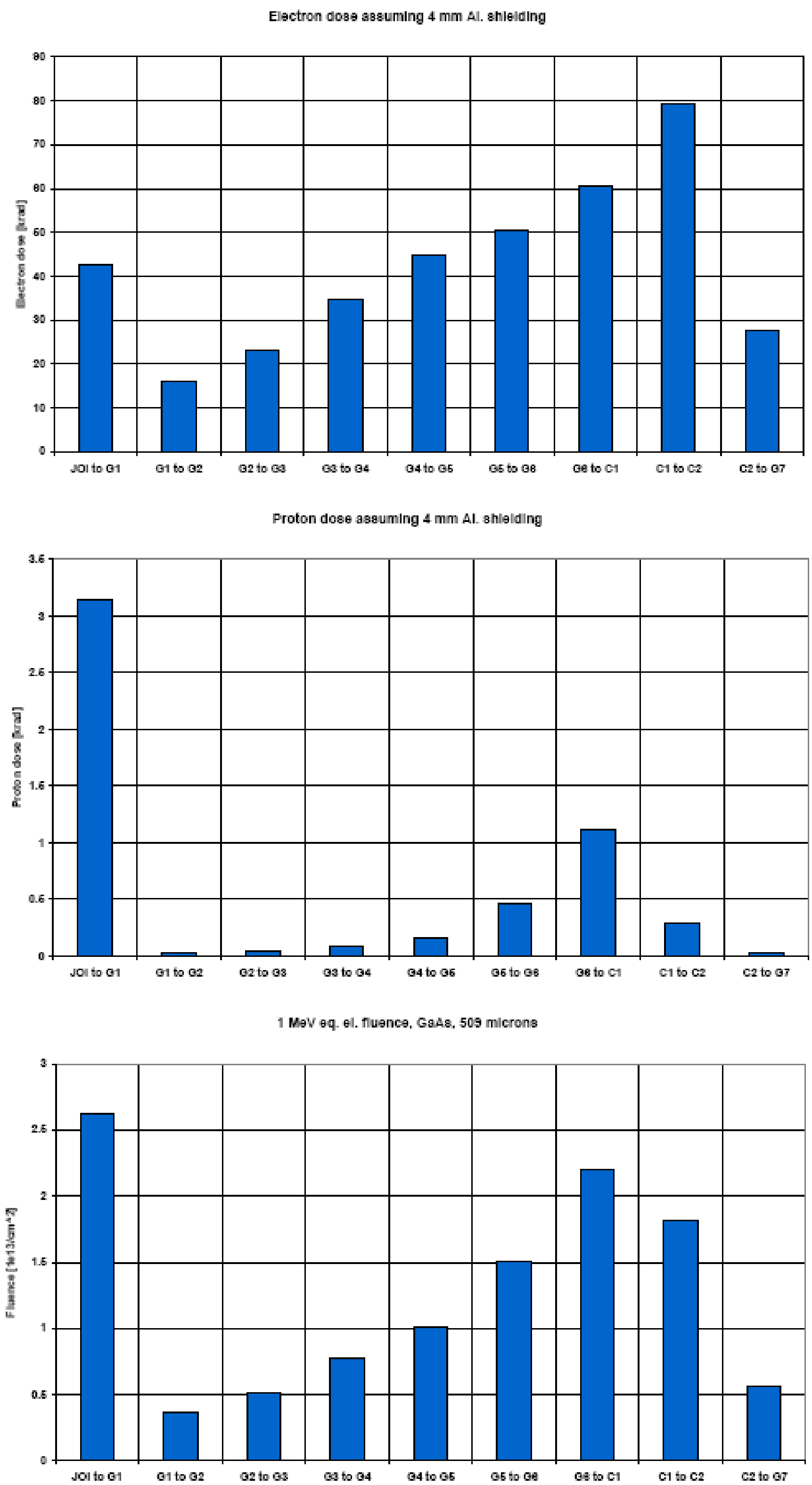

Figure 32: Radiation Doses and Fluence Received During JRS Tour

Final Orbit

The orbital elements of the final orbit are given in Table 15. This is inclined by 3.5º with respect to Jupiter equator plane.

Table 15: Target Orbit for JRS

Semi-major axis [km/R_J]	1.4 million /19.6
Eccentricity	0.44
Apojove radius [km/R_J]	2.0 million / 28.1
Perijove radius [km/R_J]	785 000 / 11
Orbital period [d]	10.7
Inclination w.r.t Jupiter equator [º]	3.5
Synchronicity relative to Europa	3
Daily electron dose for 4mm Al. shielding [krad/day]	3.15
Daily proton dose for 4mm Al. shielding [krad/day]	0.006
Daily 1 MeV eq. el. fluence, GaAs, 509 µ [e-/cm2/day]	7E11

The radiation dose rates could be reduced by increasing the perijove: as an example, if the perijove could be increased to 13 R_J, the daily electron dose would be 0.94 krad/day instead of 3.15 krad/day. This could be achieved via further Callisto and Ganymede swing-by's. Alternatively, a completely different tour could lead to an inclined final orbit incurring a lower dose, as mentioned before. This would need more analysis and is not taken as the current baseline. However this option can be studied if the current radiation dose proves to be too high.

Results

The event timeline for the JRS tour is shown in Table 16. The conditions after each event are listed. An event can be a moon encounter or a thruster manoeuvre. The third column contains the hyperbolic approach velocity for swing-by's or the velocity increment for manoeuvres. The flyby altitude applies only to swing-by's.

Table 16: Summary of the JRS tour

Event	Time [d]	v_{inf} / ΔV [km/s]	h_{flyby} [km]	T [d]	r_p [R_J]
Io	0	*11.6*	300	hyperbolic	5
JOI	0.14	*4.25*	357000	355	5
PRM	130	0.233	-	358	12.3
PSM	171	0.2	-	358	12.3
G/1	351	*7*	344	72	11.6
G/2	422	*7*	4108	43	11.1
G/3	465	*7*	342	28.6	10.4
G/4	494	*7*	2768	21.5	9.8
G/5	515	*7*	520	14.3	8.8
G/6	530	*7*	1608	11.1	7.9
C/1	542	*4.499*	2135	13.1	9.8
C/2	563	*4.499*	1618	16.5	12.3
Man	564	0.033	-	16.4	12.3
G/7	598	*4.632*	304	10.7	11

Table 17 summarizes the results of the tour design for the JRS. Note that the tour design is based on linked conics rather than full numerical integration: The achieved accuracy is sufficient at this point of the analysis. It must be noted that the dose and fluence numbers do not include the operational phase. This is discussed in JME-4.7

Table 17: JRS Tour Summary (up and until final orbit insertion)

Jovian Orbit Insertion (JOI)	[m/s]	503
Perijove Raising Manoeuvre (PRM)	[m/s]	233
Perijove Stabilisation Manoeuvre (PSM)	[m/s]	200
Delta-V cost between JOI and final orbit	[m/s]	33
Total tour budget	[m/s]	969
Total tour duration (JOI to final swing-by)	[d]	598
Io swing-by's		1
Europa swing-by's		0
Ganymede swing-by's		7
Callisto swing-by's		2
Total proton dose (4 mm Al)	[krad]	5.35
Total electron dose (4 mm Al)	[krad]	379
Total 1 MeV equivalent electron fluence	[e-/cm^2]	1.13E+14

The major part of the manoeuvre budget for the tour is made up by the PRM and the PSM. This raises the perijove from 5 R_J to over 12.3 R_J, slightly lower than the Ganymede orbit. After the PSM, the tour is nearly ballistic. Note that the PSM cannot be reduced. The PRM can only be reduced if one accepts a lower perijove for the JOI that would include higher radiation doses. The Delta-V budget of 969 m/s should be close to the minimum cost within the given constraints. A slight overall budget improvement could be achieved by splitting the PRM into two manoeuvres: because of a minor out-of-plane component, the PRM is not exactly at apojove. Two separated manoeuvres would permit to raise the perijove altitude and adjust the orbital plane.

Trajectory Plots

The JRS trajectory in the Jovian System from JOI to the point of reaching the final orbit is shown in Figure 33. The plot to the left shows the entire tour including the very large initial orbit. Clearly visible are the asymmetry in the orbit introduced by the PRM and the gradual apojove reduction. The orbits of the four Galilean moons are also drawn.

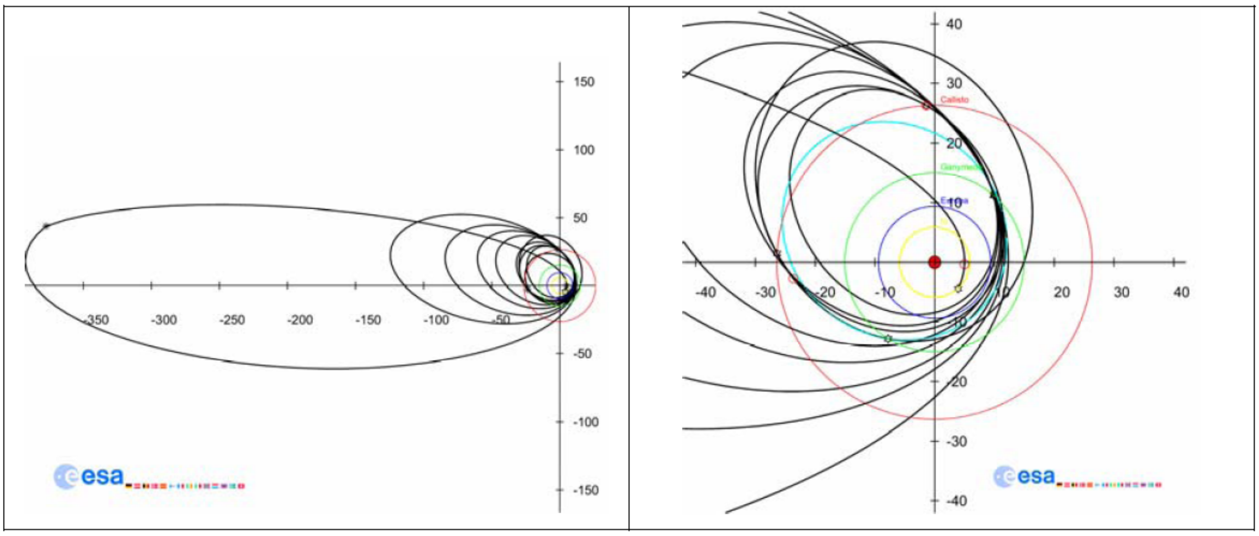

Figure 33: JRS Tour Trajectory Plots

In the plot on the right, the inner region of the Jovian System is enlarged. Again, the spacecraft trajectory is superimposed over the orbits of the Galilean moons. The plot shows that only the pass with the JOI occurs close to the central planet. The trajectory plot starts right after the Io swing-by and ends when the final orbit is reached (in blue).

The trajectories are shown from a viewpoint over the Jupiter North Pole. Therefore, inclination changes during the tour are difficult to make out. The initial inclination of JRS orbit is 8 deg. This value is reduced close to zero during the tour and will end up at 3.5 deg after the last swing-by.

JME-4.3.3 THE JEO TOUR

The JEO is stacked with the JRS up to the PSM. Before the separation, all manoeuvres (namely JOI, PRM and PSM) are performed by the JRS. Hence the aim is to minimize the manoeuvre budget from separation to Europa Orbit Insertion (EOI), including all intermediate manoeuvres while keeping the radiation dose as low as possible.

Tour Design

In comparison to the JRS tour, the orbiter is subjected to a considerably higher radiation dose. The following approach led to the final tour design:

- Europa swing-by's are permitted.
- The beginning of the tour up to the Ganymede swing-by coming after the second Callisto
- swing-by, named G/7 in the previous chapter, is similar to the JRS tour
- The swing-by G/7, together with G/8 is used to bring the perijove below Europa orbit.
- When Europa is reached, the EOI is not immediately performed: indeed the infinite velocity w.r.t. Europa can be further reduced by performing an endgame. It consists of a series of near resonant swing-by's that reduce the period of the orbit. As a consequence, the perijove is also reduced. A small manoeuvre applied at apojove allows raising the perijove back to Europa orbit, thus reducing the infinite velocity at the next encounter. Increasing the number of swing-by's decreases the Delta-V budget because the incoming infinite velocity, and ultimately the EOI, is reduced. However, each swing-by adds an apojove manoeuvre. As a compromise 4 swing-by's were chosen. The endgame design is also a trade-off between Delta-V cost and radiation exposure
- The tour ends with the EOI. The last swing-by allows reducing the EOI around 650 m/s, compared to a minimum of 2.113 km/s without an endgame. Conversely 642 m/s of apojove manoeuvres are added by the endgame, so the net gain is 1.471 km/s.

Design Outcome

Figure 34 shows how the JEO reaches the target orbit: the first part of the tour is similar to that of the JRS. After a slight increase in the period to reach Callisto orbit, the spacecraft reaches Europa with a period of one week. The period is further reduced to 3.55 days, i.e. Europa period. When the endgame starts, it can be seen that the perijove remains constant (close to Europa radius) while the apojove regularly decreases.

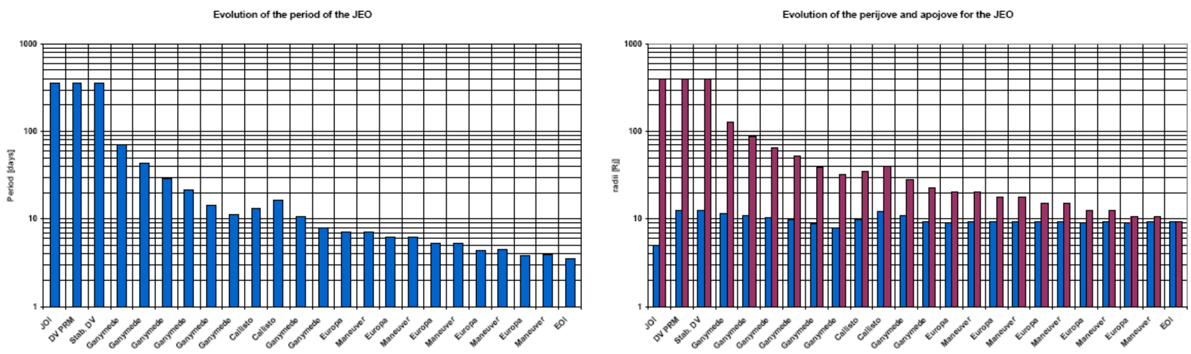

Figure 34: Orbital Period and Apo-/Perijove Radii for JEO

The incurred radiation dose increases drastically during this phase, as shown in Figure 35. It is obvious that most of the radiation dose is picked up during the endgame (i.e. a sequence of GAMs that minimises the required chemical manoeuvre at EOI) as the spacecraft spends an extended time in a high radiation environment.

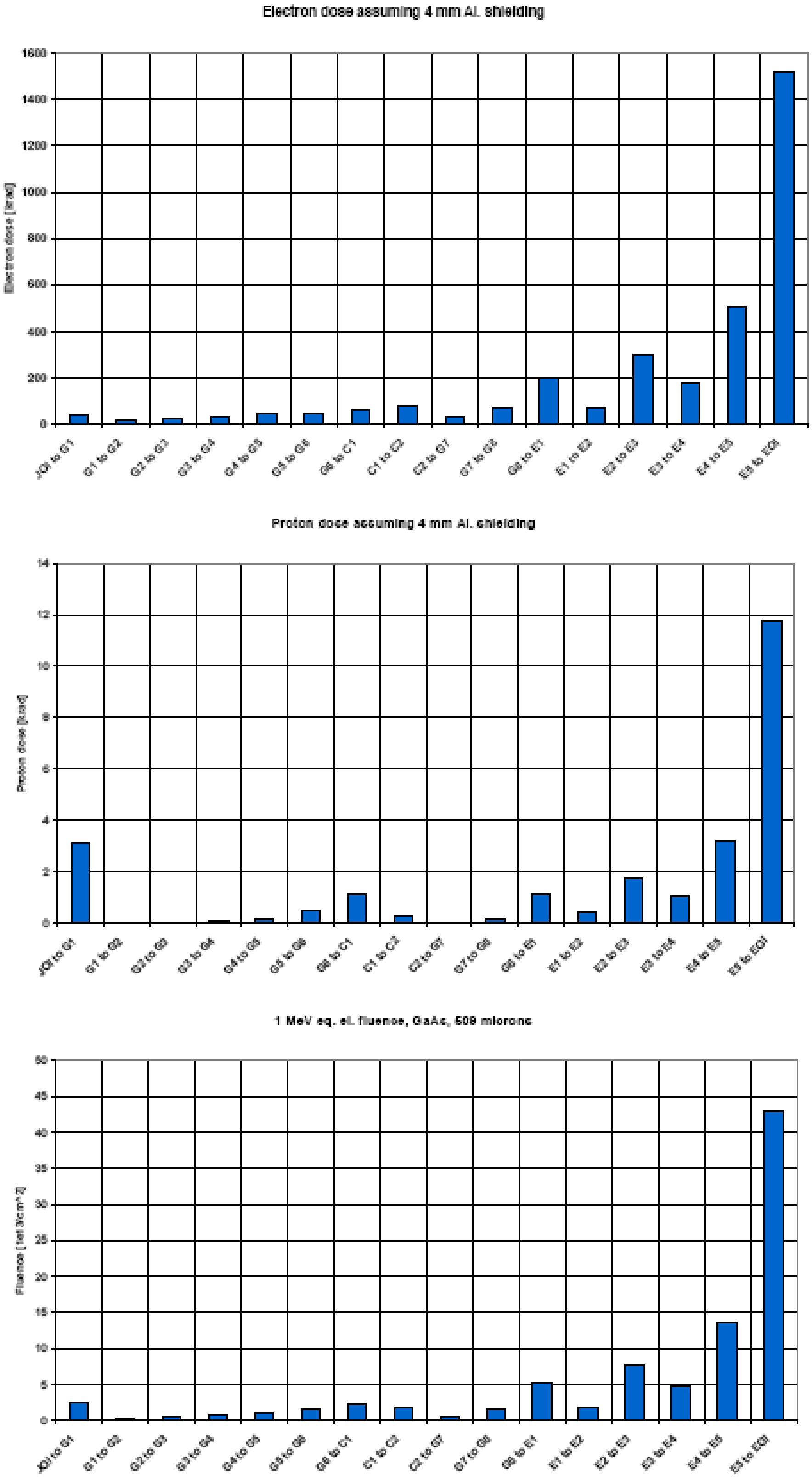

Figure 35: Radiation Dose and Fluence Received During JEO Tour

Europa Orbit Insertion and final orbit

The final hyperbolic arrival velocity at Europa is 568 m/s in the obtained tour. This leads to a velocity on the hyperbolic arc of 2.02 km/s at an altitude of 200 km over the Europa surface. As the circular velocity at 200 km altitude is 1.38 km/s, the required impulsive EOI burn size as one or several burns is 642 m/s (splitting the EOI into parts reduces the gravity losses). After this the initial manoeuvre injects the spacecraft into an eccentric orbit, which will be circularised by subsequent burns. The sum of these multiple manoeuvres approaches that of the impulsive single burn. Once EOI is performed, the JEO is captured in the target orbit described in Table 18.

Table 18: Target orbit around Europa

Altitude w.r.t. Europa	[km]	200
Orbital period w.r.t. Europa	[hours]	2.2
Daily electron dose for 4mm Al. shielding	[krad/day]	41.6
Daily proton dose for 4mm Al. shielding	[krad/day]	0.45
Daily 1 MeV eq. el. fluence, GaAs, 509 µ	[1/cm2/day]	1.32E+13
Science phase duration	[days]	60
Electron dose for 4mm Al. shielding for 60 days	[krad]	2496
Proton dose for 4mm Al. shielding for 60 days	[krad]	27
1 MeV eq. el. fluence, GaAs, 509 µ for 60 days	[e-/cm2]	7.90E+14

Results

Table 19 gives the event timeline for the orbiter tour. Events can be either manoeuvres or moon swing-by's. The main difference with the JRS is the introduction of the endgame with Europa.

Table 19: Event Timeline for JEO Tour

Event	Time [d]	v_{inf} / ΔV [km/s]	h_{flyby} [km]	T [d]	r_p [R_J]
Io	0	11.6	300	hyperbolic	5
JOI	0.14	4.25	357000	355	5
PRM	130	0.23	-	358	12.3
PSM	171	0.2	-	358	12.3
G/1	351	7	344	72	11.6
G/2	422	7	4108	43	11.1
G/3	465	7	342	28.6	10.4
G/4	494	7	2768	21.5	9.8
G/5	515	7	520	14.3	8.8
G/6	530	7	1608	11.1	7.9
C/1	542	4.5	2135	13.1	9.8
C/2	563	4.5	1618	16.5	12.3
G/7	598	4.72	970	10.7	10.8
G/8	619	4.72	1360	7.9	9.3
E/1	644	2.89	1720	7	9.2
Man	648	0.06	-	7.1	9.4
E/2	651	2.38	446	6.2	9.2
Man	654	0.04	-	6.2	9.4
E/3	676	2.05	514	5.3	9.2
Man	678	0.07	-	5.3	9.4
E/4	687	1.57	309	4.3	9.1
Man	688	0.1	-	4.4	9.4
E/5	704	0.95	2426	3.8	9.1
Man	705	0.09	-	3.8	9.1
EOI	743	0.64	200	3.55	9.4

Table 20 summarizes the results for the orbiter tour, which, as for the JRS, were obtained using the linked conics method rather than numerical integration. It must be noted that the dose and fluence numbers do not include the operational phase. This is discussed in JME-4.7.

Table 20: JEO Tour Summary (up and until final orbit insertion)

JOI	[m/s]	0
PRM	[m/s]	0
PSM	[m/s]	0
Delta-V cost between stab. manoeuvre and EOI	[m/s]	372
Delta-V cost of GAMs	[m/s]	120
EOI	[m/s]	642
Total tour budget	[m/s]	1134
Total tour duration (JOI to final swing-by)	[d]	743
Io swing-by’s		1
Europa swing-by’s		5
Ganymede swing-by’s		8
Callisto swing-by’s		2
Total proton dose (4 mm Al)	[krad]	24.86
Total electron dose (4 mm Al)	[krad]	3220
Total 1 MeV equivalent electron fluence	[e-/cm^2]	8.91E+14

Manoeuvres of the stack are performed by the JRS only. Therefore the Delta-V budget is zero for the JOI, PRM and PSM. The total impulsive manoeuvre budget (excluding the gravity losses, navigation and contingencies) is the sum of intermediate manoeuvres and the EOI. The JEO reaches its final orbit 145 days after the JRS has achieved its target orbit. As can be expected, the radiation dose is considerably larger than for the JRS tour: especially the electron doses are more than one order of magnitude larger, as is the 1 MeV equivalent electron fluence for the solar arrays. In total, the electron dose and solar array fluence have the same order of magnitude as the amount expected for the science phase in orbit around Europa.

The last stage of the endgame is the main contributor to the radiation dose. An alternative option consists in removing the last Europa swing-by and is presented in Table 21. The first consequence is the reduction of the transfer, about 40 days. Hence the radiation doses are also significantly reduced: -46.8 % for the proton dose, -47.0% for the electron dose and -48.1 % for the fluence.

Table 21: JEO Tour Summary Excluding the Last Stage of the Endgame (up and until final orbit insertion)

JOI	[m/s]	0
PRM	[m/s]	0
PSM	[m/s]	0
Delta-V cost between stab. manoeuvre and EOI	[m/s]	280
Delta-V cost of GAMs	[m/s]	112
EOI	[m/s]	783
Total tour budget	[m/s]	1175
Total tour duration (JOI to final swing-by)	[d]	704
Io swing-by’s		1
Europa swing-by’s		4
Ganymede swing-by’s		8
Callisto swing-by’s		2
Total proton dose (4 mm Al)	[krad]	13.07
Total electron dose (4 mm Al)	[krad]	1705
Total 1 MeV equivalent electron fluence	[e-/cm^2]	4.62E+14

However this improvement has a cost in terms of Delta-V: the last manoeuvre of 92 m/s is removed (see Table 19), but the EOI increases because the incoming infinite velocity rises from 568 m/s to 950 m/s. In total, 41 m/s are added to the final tour. This option is very interesting from a radiation point of view and the additional propellant required for this option (~20 kg) must be traded against the lower shielding mass required due to the lower dose.

Trajectory Plots

The JEO tour trajectory is shown in Figure 36, with the entire tour in the left-hand diagram and an enlarged view of the inner system on the right. The viewing direction is from the north pole of Jupiter. The orbits of the Galilean moons are included in addition to the space track of the JEO.

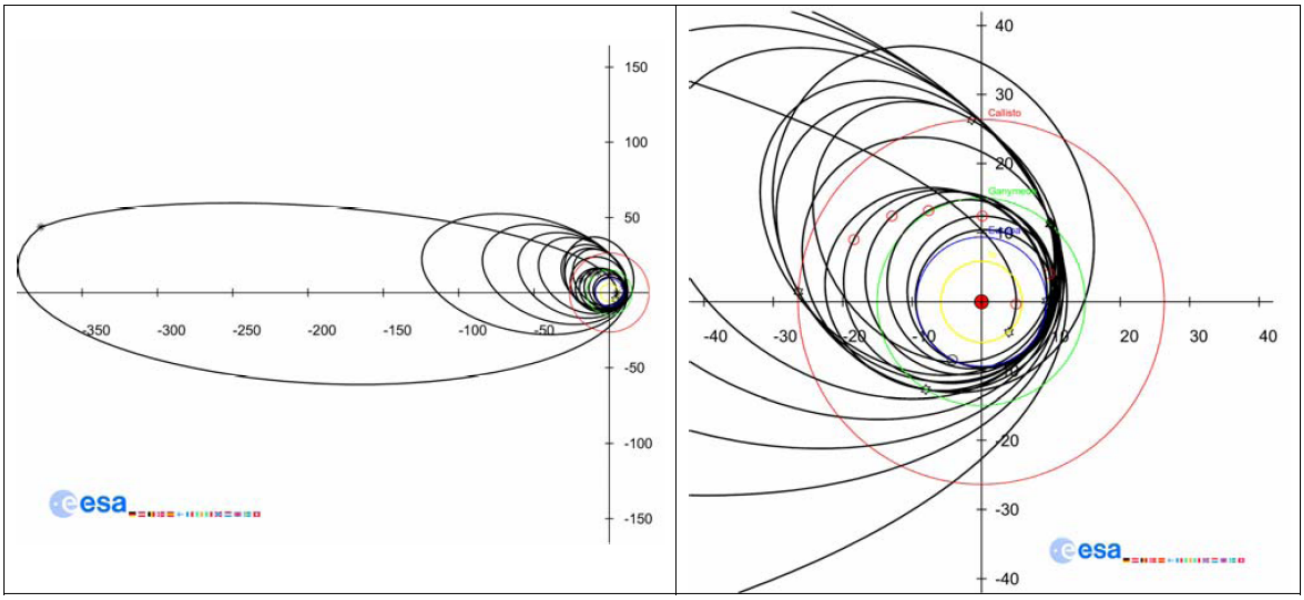

Figure 36:JEO Tour Trajectory Plots

A comparison with Figure 33 shows the additional part corresponding to the endgame: the apojove radius is reduced step by step with the help of near resonant Europa swing-by's (whereby the position for the swing-by slowly drifts) and apojove manoeuvres (red circles).

Summary

Table 22 provides an overview of the Delta-V budgets for the reference scenario, while

Table 23 shows the alternative scenario that reduces the radiation exposure by dispensing with the final Europa GAM at the cost of 49 m/s minus 8 m/s (one less GAM) = 41 m/s, resulting in an increase of approximately 20 kg on the launch mass. This must be offset against a **decrease** of approximately 1.5 Mrad (4 mm Al) in total dose (proton + electron) and 4.29 E+14 e-/cm^2.

Table 22: Trajectory and radiation overview for the selected mission profile

Parameter		JRS	JEO
Launcher		Soyuz Fregat 2-1b from Kourou	
Launch date		April 2019	
Apogee Raise	[m/s]	706	-
Inclination change	[m/s]	0	-
Earth escape	[m/s]	733	-
VEEGA+DSM (assumed 30 m/s per planetary GAM)	[m/s]	90+32=122	-
Jovian Orbit Insertion (including Jupiter and Io GAM)	[m/s]	503+30+8 = 541	-
Perijove Raising Manoeuvre	[m/s]	203	-
Perijove Stabilisation Manoeuvre	[m/s]	200	-
Delta-V cost between JOI and final orbit	[m/s]	33	-
Delta-V cost between stabilisation manoeuvre and EOI	[m/s]	-	372
Europa Orbit Insertion	[m/s]	-	642
GAM correction manoeuvres (assumed 8 m/s per GAM)	[m/s]	72	120
Total	[m/s]	**2610**	**1134**
Transfer duration	[years]	7.3	
Tour duration	[days/years]	598 / 1.6	743 / 2.0
Total radiation dose (electron + proton) until final orbit insertion	[krad]	385	3 245
Total 1 MeV equivalent fluence until final orbit insertion	[e-/cm^2]	1.13E+14	8.91E+14

Table 23: Trajectory and radiation overview for the alternative mission profile

Parameter		JRS	JEO
Launcher		Soyuz Fregat 2-1b from Kourou	
Launch date		April 2019	
Apogee Raise	[m/s]	706	-
Inclination change	[m/s]	0	-
Earth escape	[m/s]	733	-
VEEGA+DSM	[m/s]	122	-
Jovian Orbit Insertion (including Io GAM)	[m/s]	503+30+8 = 541	-
Perijove Raising Manoeuvre	[m/s]	203	-
Perijove Stabilisation Manoeuvre	[m/s]	200	-
Delta-V cost between JOI and final orbit	[m/s]	33	-
Delta-V cost between stabilisation manoeuvre and EOI	[m/s]	-	372
Europa Orbit Insertion	[m/s]	-	642+49 =691
GAM correction manoeuvres (assumed 8 m/s per GAM)	[m/s]	80	112
Total	[m/s]	**2618**	**1175**
Transfer duration	[years]	7.3	
Tour duration	[days/years]	598 / 1.6	704 / 1.9
Total radiation dose (electron + proton) until final orbit insertion	[krad]	385	1718
Total 1 MeV equivalent fluence until final orbit insertion	[e-/cm^2]	1.13E+14	4.62E+14

JME-4.4 The Operational Phase

After the approximately 7-year mission to reach the final orbits, the main science phase of the mission begins. The JRS will remain in its highly elliptical orbit for approximately two years, mainly acting as a data transfer station but also performing measurements of the Jovian System. Both spacecraft will also perform science operations during their tours, to maximise the science return (e.g. measurements of the Jovian radiation environment and possibly of Ganymede and Callisto during the flyby's).

The JEO operational phase will be completely different. As a result of the strong orbit perturbations caused by Jupiter and the limited propellant available, the JEO will have a highly time constrained operation phase, before the orbiter will crash on the icy surface of Europa. Analysis [22] has shown that a circular orbit between 50 km and 300 km provides the maximum lifetime of 60 to 66 days (maximum around 150 km), as shown in the next figure.

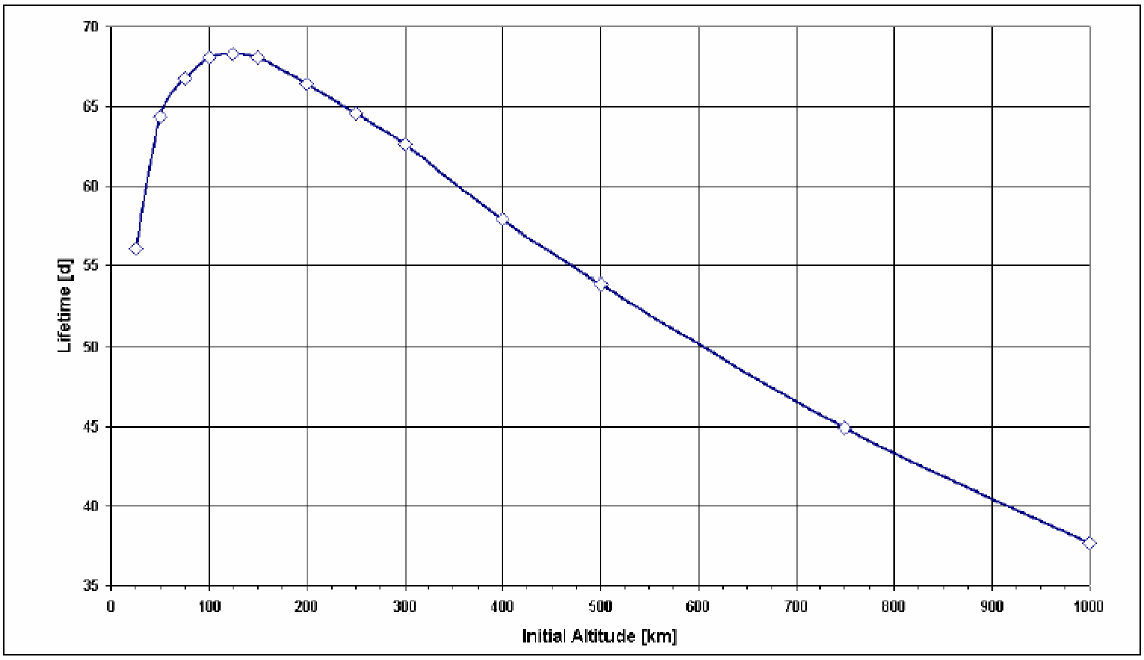

Figure 37: Variation of lifetime as a function of the initial altitude [22]

However, the mission duration is not only limited by orbital mechanics: the received radiation dose in 60 days will nearly reach 3 Mrad (assuming 4 mm of Al shielding), raising the total radiation dose to more than 6 Mrad (4 mm Al). Therefore the 60-day mission must be seen as the maximum mission duration, even if the orbital lifetime could be increased with station keeping manoeuvres. For this study a 60-day observation period of Europa is taken as the design case.

A further limitation lies in the illumination of the spacecraft: the lower the orbit, the longer the eclipse periods of the spacecraft and the larger the battery capacity requirement. This additional mass will directly impact the payload mass and therefore requires a careful trade-off. At this point an orbit altitude of 200 km has been selected as a compromise between science and spacecraft requirements. Clearly the JRS orbit must be chosen in such a way that JEO can uplink all its science data before crashing onto Europa or succumbing to the lethal radiation dose.

JME-4.5 Europa Coverage

The following picture [22] shows the coverage quality for the nominal, circular, polar 200 km orbit. The Europa surface area is subdivided into a grid of 4º width in longitude and 3º width in latitude. For each element in the surface grid, the number of times the satellite passes directly overhead is counted, subject to the condition that the respective part of the Europa surface must be sunlit at the time of the pass.

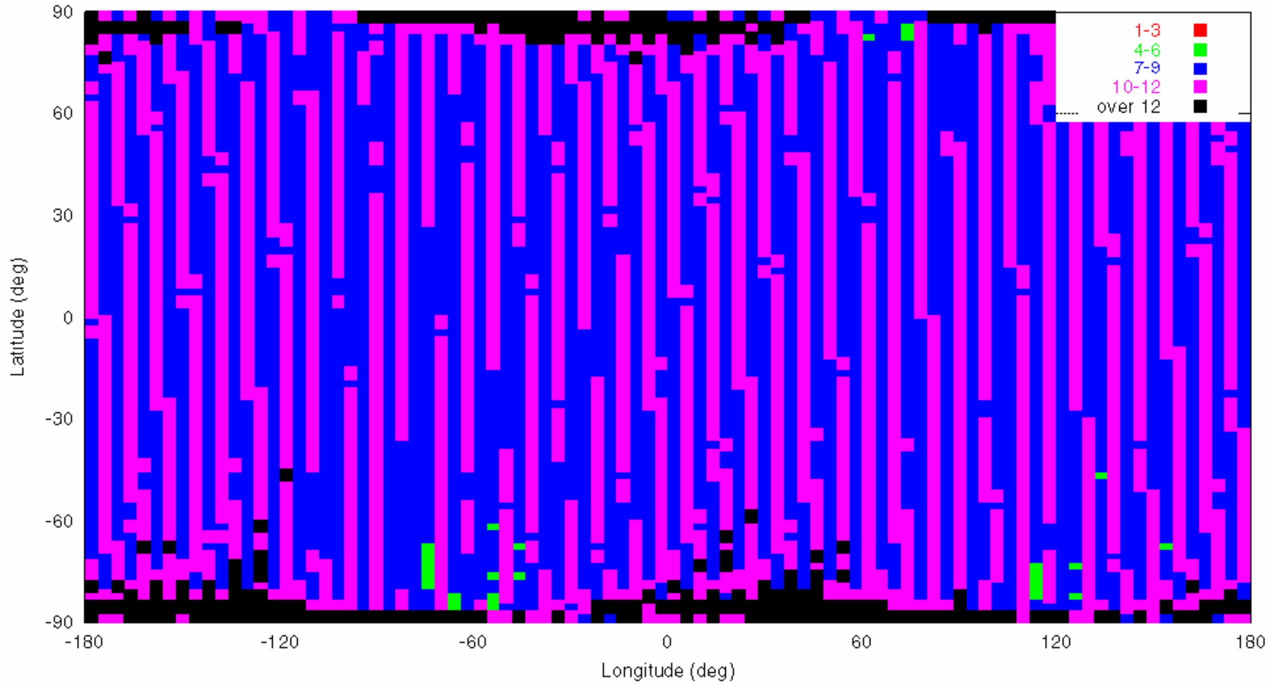

Figure 38: Surface coverage quality for the nominal case [22]

The spacecraft passes every point of the sunlit surface at least 4 times during the 66 days until it hits Europa. Most of the surface can be imaged more frequently than that, especially the polar regions.

JME-4.6 Decommissioning

In view of the planetary protection constraints on Europa, the decommissioning of the spacecraft must be taken into account. The JEO will crash on Europa and must therefore comply with the stringent Class IV COSPAR cleanliness requirements. A class IV approach would also be an option for the JRS if no decommissioning is planned. However, alternatives do exist for the JRS and would make the AIV process much less complicated. Two main options will be shortly discussed in this section.

JME-4.6.1 Endgame crash

Once of the options is to perform a decommissioning tour after the JRS finishes its mission. Preliminary analysis has shown that targeting the JRS into Jupiter (as for Galileo) is not an option, since it would expose the JRS to a tour in the high exposure zones of the Jovian radiation belts. After the high dose to which JRS will have already been subjected, this additional dose would exceed the design limit before completing the tour.

However the GAM sequences have shown that significant orbital changes can be achieved for orbits that intersect the Galilean moons. Since Ganymede and Callisto (and obviously Europa) are subject to the high planetary protection restrictions, these cannot be used as a crash site. However, Io does not have the same restrictions due to its volcanic surface. First order analysis has show that an endgame can be devised to target an impact into Io. The benefit would be that a near zero propulsive manoeuvre strategy can be devised, which would provide additional science opportunities due to the additional fly-bys. The downside is that the received radiation dose will still be high. This must therefore be taken into account should this option be considered further.

Preliminary analysis has resulted in the following:

Table 24: First order analysis of possible Io end game for JRS

Event	Ganymede revolutions	JRS revolutions	Perijove radius [R_J]	Time [days]
Initial orbit	-	-	11	10.7
Ganymede GAM 1	1	1	8.8	7.2
Ganymede GAM 2	3	4	6.6	5.4
Ganymede GAM 3	-	-	4.9	4.3

This table shows that after one JRS orbit, JRS will encounter Ganymede, after which the perijove is lowered by ~2 R_J. Then after 1 revolution of both JRS and Ganymede the next GAM takes place reducing the perijove further. After this the JRS will perform 4 revolutions and Ganymede 3. After this the next GAM takes place, which will place the JRS in an orbit below the Io orbit (i.e. 5.9 R_J). Even if this scenario is purely ballistic, a few tens of m/s will be required for this manoeuvre to fine tune the trajectory. This quick analysis did not include a radiation dose analysis, so the feasibility of this scenario must still be confirmed. An alternative would be to perform the Io targeting manoeuvre with both Ganymede GAMs and a propulsive manoeuvre. This would reduce the radiation exposure, but could require a manoeuvre in the order of 200 m/s [23], which will severely impact the launch margin. Also this option would require more analysis.

As an alternative to Io, one could consider investigating the position of smaller Jovian moons, provided they are large enough in view of targeting inaccuracies.

JME-4.6.2 Spacecraft Graveyard Option

Another option could be to target an orbit that does not intersect the orbits of any Galilean moon. This would require the orbit's peri- and apocentre to lie between moons or alternatively to reach a higher inclination orbit to reduce the probability of collision with a moon. The first option would require a high Delta-V, making it unattractive. The second option would be performed by means of GAMs, however this would require an extensive and detailed analysis over a long time interval and the chance of impact cannot excluded [23]. This option is therefore not recommended.

JME-4.6.3 Escape option

The last mentioned option provides another interesting scenario. Analysis performed by Astrium in the context of the Jovian System Explorer (JSE) [23] has shown that a combination of Callisto and Ganymede GAMs could be used to raise the apocentre and potentially escape the Jovian System. This option was studied for the JSE orbits, which are considerably different than the JRS orbit (see the chapter on The Jovian System Explorer on page 117 and further), namely a 15 R_J x 70 R_J orbit. Further analysis is required to understand the implications for the 11 R_J x 28 R_J orbit of the JRS.

Provided this option is viable with the predicted low Delta-V and the spacecraft will still be functional (i.e. not at end of life), this option could allow an interesting add on to the mission. Again this option needs further analysis to assess its viability for the JRS case.

JME-4.7 Radiation environment and electrostatic discharge

The radiation environment of the Jovian System is very severe. This is especially the case for the JEO spacecraft. The next tables provide an insight in the radiation doses that are expected, based on the Divine-Garrett radiation models [4]. It is important to note that these results are not based on the new model incorporating Divine & Garrett, GIRE and Salammbô [5] (see page 13).

Table 25 shows the expected radiation dose as a function of (Aluminium) shielding thickness for the main phases of the JME.

Table 25: JEO total radiation dose assessment as function of shielding thickness

Environment	4 mm shielding (krad)	8 mm shielding (krad)*	10mm shielding (krad)*
Jupiter Tour	3245	824	358
dose per day around Europa	41.6	14.3	8.3
66 days around Europa	2775	944	548
Total	6020	1768	906

*= The total dose behind 8 and 10 mm shielding were extrapolated from earlier results and are therefore indicative.

Table 26 shows the expected radiation dose for the two JME spacecraft.

Table 26: JEO and JRS total radiation dose assessment

Environment	JEO	JRS
Jupiter Tour	3245krad (4mm Al)	385krad (4mm Al)
	8.9E14 e-/cm² on solar cells*	1.1E14 e-/cm²on solar cells*
Alternative JEO Tour	1718 krad (4mm Al)	-
	4.6E14 e-/cm² on solar cells*	
Operational orbit	2 775 krad (4mm Al)	1 150krads/year (4mm Al)
	8.7E14 e-/cm² on solar cells*	2.6 E14 e-/cm²/year on solar cells*

*= The presented figures are likely to underestimate the damage caused by protons to GaAs solar cells.. The fluences given here are equivalent 1 MeV electron fluences after 509 micron cover glass.

The aluminium shielding thickness is used to assess the amount of shielding that is required. However, mass gains can be achieved by using different materials, such as Tantalum, as shown in the next table. Tantalum-based shielding is therefore recommended when the required aluminium shielding thickness exceeds 4mm (it must be noted that Tantalum is difficult to machine). In the end a layering of different materials will be required to mitigate both primary and secondary radiation damage effects (including Bremsstrahlung).

Table 27: Mass gain by replacing Aluminium shielding by Tantalum shielding [10]

Environment	4 mm shielding (krad)	8 mm shielding (krad)	10mm shielding (krad)
Dose	5.3 Mrad	1.5Mrad	770Krad
Equivalent Ta shielding	0.651mm	1.1mm	1.3mm
Mass gain in percents	**~0%**	**> 12 %**	**20%**

For comparison: the Galileo equivalent Al shielding was 7.5mm up to 10 mm specific places.

In the future more detailed calculations will be performed using the more recent Salammbô model. Moreover SEU/SLU rates will also be computed as more information on the heavy ion concentration in JEO/JRS orbits will become available.

To further understand the effect of shielding on the total dose, Figure 39 shows the total dose for the JEO as a function of the shielding thickness (assuming Aluminium). To generate this graph, an approximation of the JEO tour is used; therefore the graph is to be used as an indication. This figure clearly shows that an equivalent Al thickness in excess of 10mm significantly reduces the total dose. The effect of Bremsstrahlung is not included in this graph.

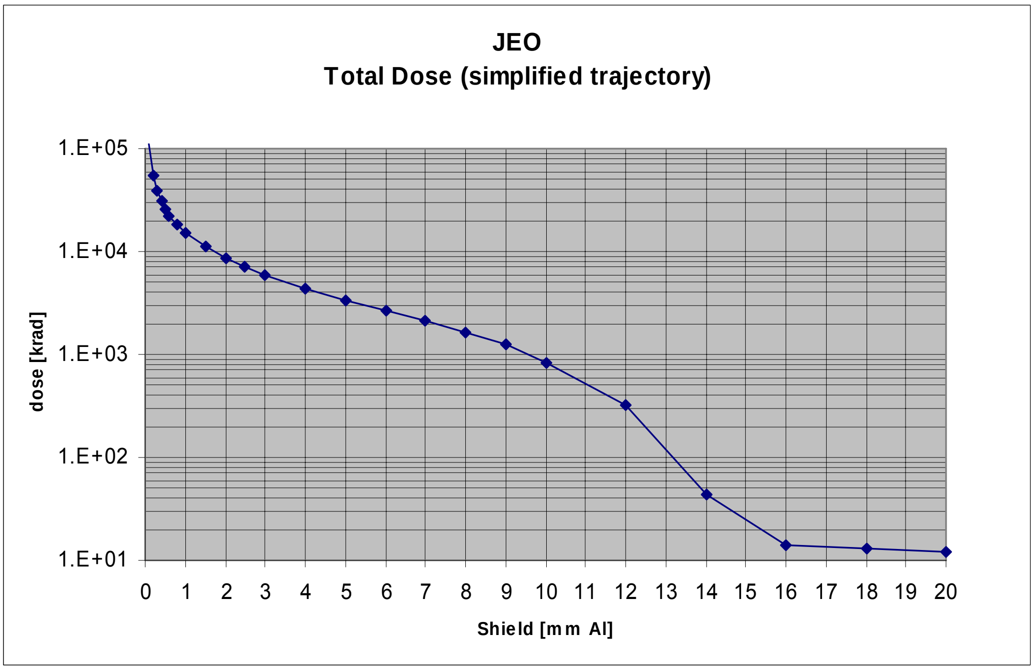

Figure 39: JEO total dose vs. shielding thickness

The size of the HIPS also drives the mass of the shielding required for the electronics box (i.e. the back end of the payload). The next figure shows the shielding mass required to protect the electronics for a certain dose as a function of a (theoretical) box size of the HIPS electronics. Again this figure is based on a simplified trajectory and should therefore be taken for informative purposes only.

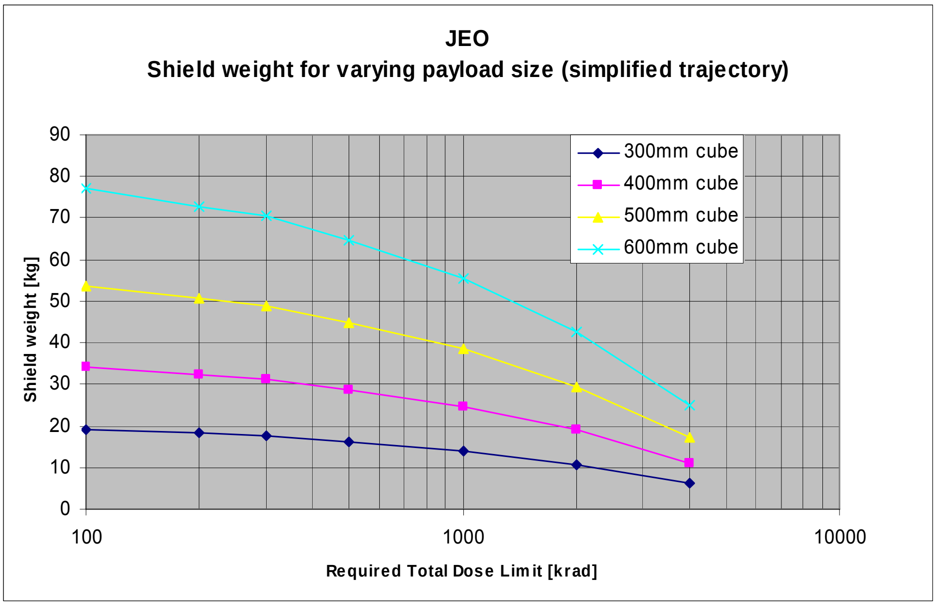

Figure 40: Shield mass as function of total dose and P/L volume

Another issue that needs attention is the internal discharging. Internal charging refers to the accumulation of electrical charge on interior, ungrounded metals or on or in dielectrics inside the spacecraft. Internal discharge can occur close to electronics equipments, causing significant upset or damage to satellite electronics.

The following figure shows the energy electrons must have in order to penetrate aluminium (10 mils ~0.25mm). Note also that 10^{10} to 10^{11} electrons/cm² are needed on the interior of a spacecraft to possibly cause internal discharges.

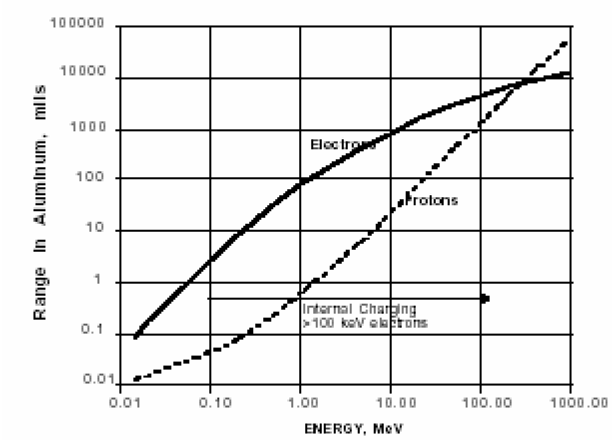

Figure 41: Approximate average electron and ion penetration ranges in aluminium [10]

Anomalies can be avoided by limiting isolated conductors on the inside of the spacecraft radiation shield. This solution was successfully implemented for the Galileo programme: isolated conductors were limited to <3cm², ungrounded conductors with length greater than 25cm were not allowed.

Note that both internal and external electrostatic discharge can occur due to differential charging of ungrounded spacecraft surfaces. In particular all external surfaces must be conducting and be grounded to the spacecraft platform.

JME-5 THE SPACECRAFT

This chapter provides the main characteristics of the spacecraft. After giving the rationale for the two spacecraft configuration, their current configuration and tentative payloads will be briefly discussed.

JME-5.1 The composite

The composite consists of two spacecraft: the Jupiter Relay Satellite and the Jupiter Europa Orbiter. The propulsive manoeuvres of the composite will be performed by the JRS until and including the Jovian Orbit Insertion (JOI) burn, the Perijove Raising Manoeuvre (PRM) and the Perijove Stabilisation Manoeuvre (PSM). The propulsion system for the JME is all-chemical, to limit the spacecraft complexity and cost. Once the PSM has been performed, the composite will be separated into the JRS and JEO, which will perform their separate tours. As soon as the composite is separated from the Fregat upper stage, the JEO and JRS solar arrays will be deployed, to provide the power required during the transfer. The Venus fly-by will bring the composite to its closest approach to the Sun and in combination with the high albedo of Venus, the spacecraft will be exposed to high temperatures. Since the GaAs LILT solar cells are not designed for the very high temperature, the solar arrays will need to be off-pointed to reduce the thermal load. If this measure is inadequate, the solar concentrators, required in the Jovian System, must be deployed after the Venus GAM. This would require a more complex solar panel design, but it could be the only way to avoid damage. Analysis indicated that restrictions on thermal emission through the cells should not be a problem for the panels as a result of the use of OSR (Optical Solar Reflectors).

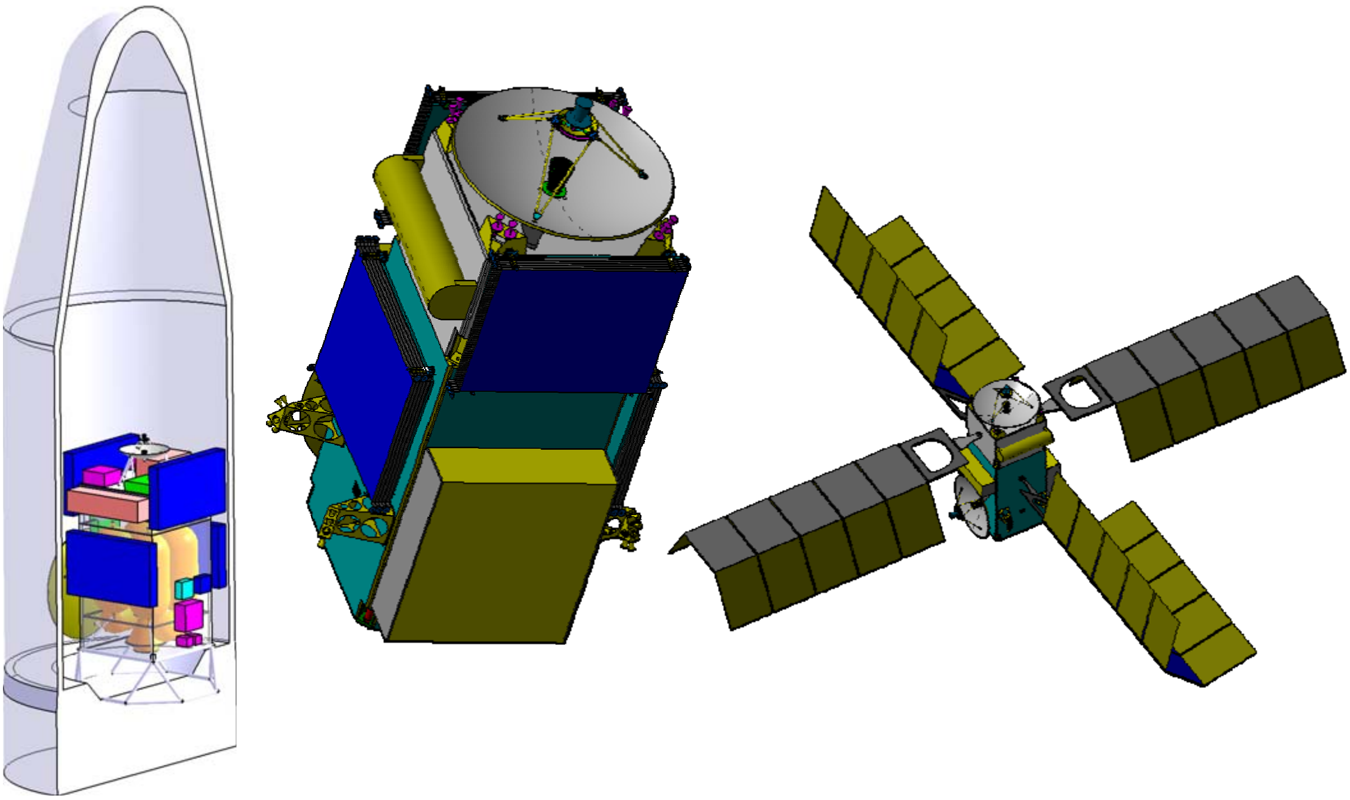

Figure 42: The JME spacecraft composite under the Soyuz Fregat 2-1b fairing (L), in launch (M) and transfer (R) configuration

The composite will also use the high gain antennas of both spacecraft, to be able to operate during all orientations, without the need of a steerable HGA by switching from one antenna to the other depending on the direction of the Earth with respect to the composite. This will require a master-slave operation system capable of using systems on both spacecraft.

JME-5.2 The Jovian Europa Orbiter

The JEO will carry all payloads required for the remote sensing mission of Europa (36 kg, 30 W, see JME-3.1). The operational mission duration of the JEO is approximately 60 days.

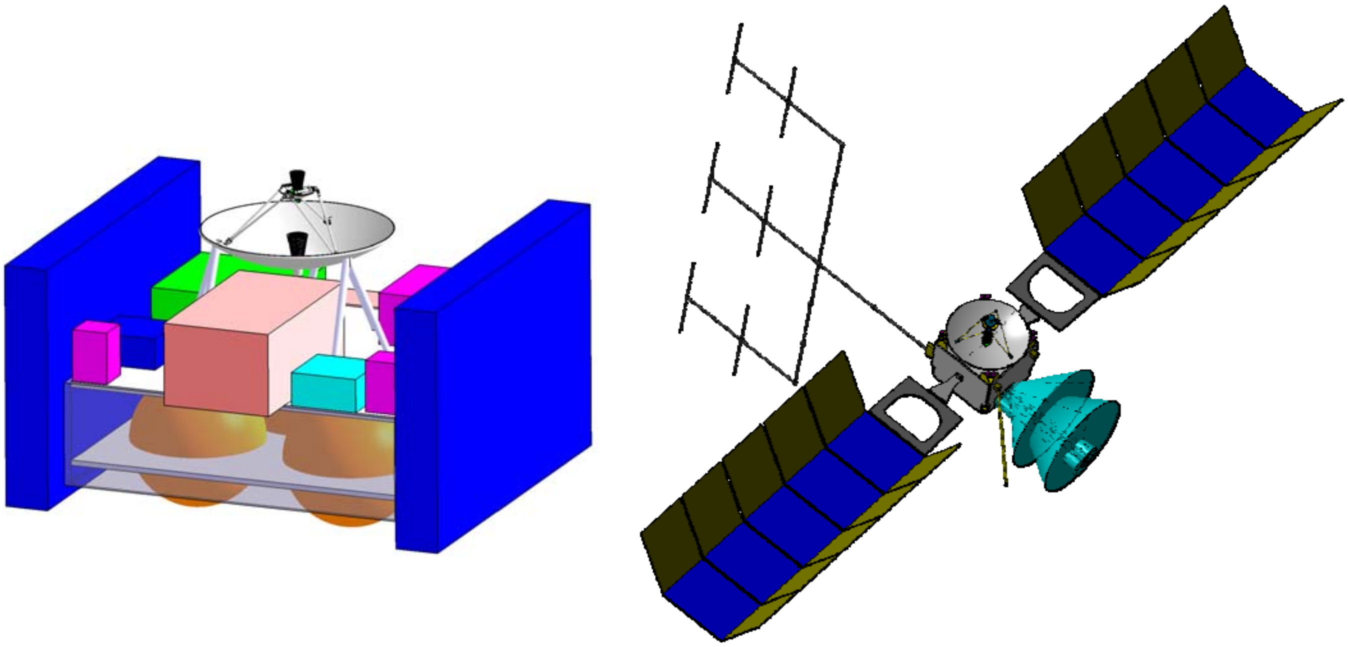

Figure 43: The Jovian Europa Orbiter cutaway and deployed view (radar does not reflect the more up to date EuGPR design)

To limit the mass of the spacecraft, the main structure is composed of Carbon Fibre Reinforced Plastics (CFRP), rather than the traditional aluminium honeycomb. The drawback of this method is that the spacecraft will provide less shielding (up to 50% less) than the traditional Al structure. This must be accounted for in the radiation dose calculations when a more detailed sector analysis is performed, which will show how much shielding is provided by the spacecraft itself. The behaviour of CFRP in the high radiation environment must be studied, especially the behaviour of the used resin.

As explained before, the main challenges of the JEO are the very high radiation levels (close to 6 Mrad (4 mm Al) or 4 Mrad (4mm Al) for the alternative JEO tour and a 1 MeV equivalent fluence of 1E15 e-/cm^2), the low solar flux (1/25th of the flux at Earth), the strong orbit perturbations caused by Jupiter and the long distance between the Jovian System and Earth. The high radiation exposure calls for a combination of radiation hardened electronics and shielding: Radiation hardened electronics up to 1 Mrad are currently foreseen. The additional radiation will be attenuated by placing the spacecraft electronic components in a shielded box whenever possible. Spot shielding will be used for the remaining electronics.

The following figure shows the functional architecture of the JEO spacecraft:

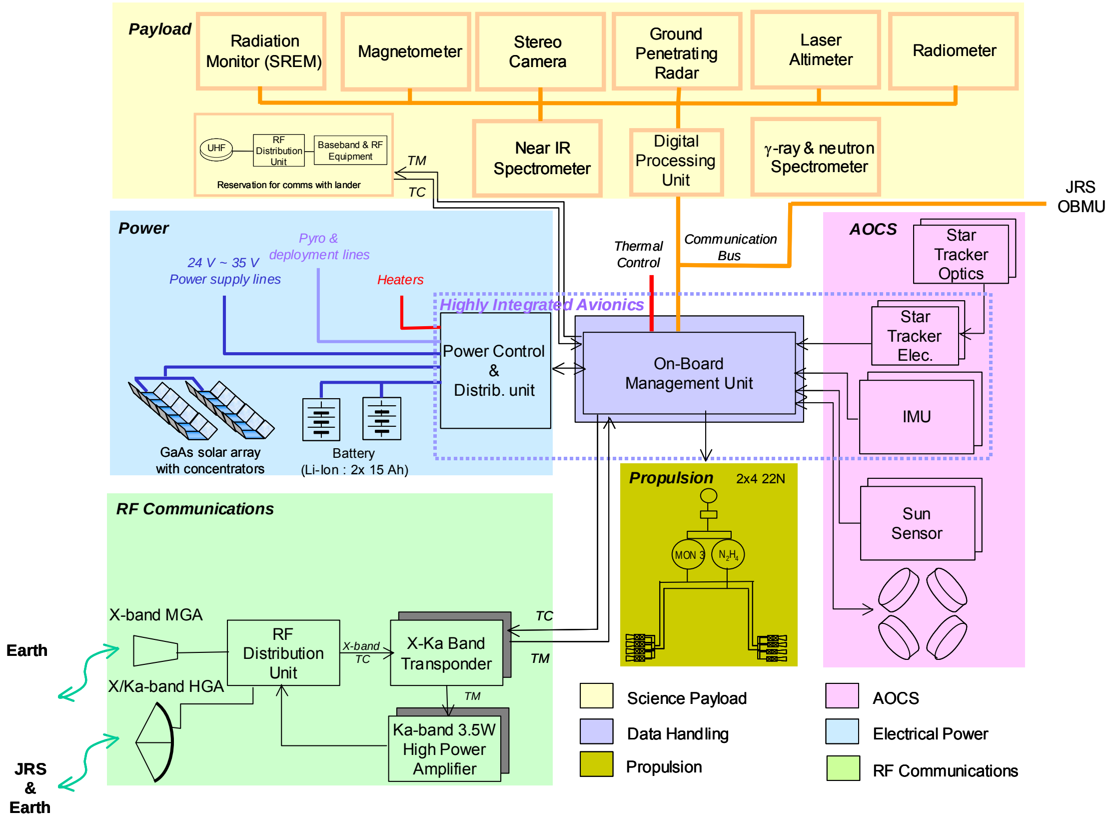

Figure 44: Schematic of the JEO system [10]

The use of solar concentrators has been foreseen to mitigate the low solar flux: highly reflective panels at the edge of the solar array that increase the total flux on the photovoltaic cells, to obtain an acceptable solar array size (~15 m^2 for 232 W EOL). Figure 43 shows the resulting solar array design. These figures are based on current triple junction GaAs cells, assuming that LILT GaAs solar cells can be developed, capable of withstanding the encountered electron and proton fluence. This is currently being investigated in a dedicated study (see page 134). The exact design of the solar concentrators must also be studied in more detail. Should the development of the solar cells prove to be unfeasible, an alternative power source, such as RTGs, will have to be considered (see next chapter on RPS).
The solar array is pointed towards the Sun by a Solar Array Drive Mechanism and around Europa, by a complementary satellite yaw manoeuvre for guaranteeing an optimal capture of the solar flux, provided it doesn't affect science operations. During eclipses, the required power is provided by a Li-Ion battery module. This module has a 571 Wh BOL capacity and includes redundant strings. Such sizing is driven by the requirement for science during eclipses by Jupiter (3.5h). Battery charge management is performed by the Power Control and Distribution Unit by shunting non-needed solar array sections.

The Data Handling architecture is organised around an integrated, internally redundant On-Board Management Unit (OBMU). The JEO data handling mainly works in slave mode when stacked on top of the JRS and in independent mode once the composite is separated. In slave mode the JEO OBMU directly receives its instructions from the JRS OBMU through the communication bus linking the two spacecraft. In independent state, the command and control functional chain is in charge of all activities, ranging from telecommands and telemetry to memory management, health monitoring and all AOCS functions. The On-Board Management Unit, which is part of the Highly

Integrated Avionics package, is based on a powerful microprocessor and includes 64 Gbits of RAM memory.

The limited power and the short lifetime dictate the use of a relay spacecraft. Only in this way will the gathered data be secured, before JEO impacts on Europa. As will be explained in the next section, it is impossible to send all data to Earth in real-time. The Europa orbiter will require a data storage capability in excess of 50 Gbit to store data that can not be uplinked to the JRS directly. A 0.75 m high gain antenna (HGA) is used to communicate with the relay spacecraft and even Earth, when the spacecraft-Earth alignment allows for it. The communication links with Earth will only allow for very limited telemetry rates, just enough to send (and receive) housekeeping telemetry or very limited science data. RF communications are performed in X- and Ka-Bands. X-band is dedicated to ground telecommands either by a direct link (7 GHz) or via JRS (8 GHz). Ka-band (32 GHz) is dedicated to both housekeeping and science telemetry, in accordance with ESA standards. An additional MGA is used as backup. On one hand TM/TC with Earth is performed at a data rate of 2kbps. On the other hand, science telemetry is performed with JRS at a variable data rate between 0.9 Mbps and 2.3 Mbps. Data is only transferred once every 10.7 days (period of the JRS orbit), at the minimum distance between the two spacecraft. New technology developments are required on the JEO communications system on high efficiency Solid State Power Amplifiers (SSPA) (30% targeted) and high data rate capability (up to 2.3 Mbps).

The current design foresees the use of 2x4 22 N thrusters (I_{SP} = 308 s) rather than a high thrust main engine. This approach has been selected to facilitate the accommodation of the JEO on the JRS for the composite configuration, resulting in a very compact JEO design, saving structural mass. Furthermore the lower thrust level will also impose lower stresses on the deployed JEO structure. This approach also offers an operational advantage: the thrusters can be tested during the cruise phase, which would be impossible for a main engine encased in the composite.

Attitude and orbit control is performed by a chemical dual-mode propulsion system using 4 redundant 22N thrusters and operating with MON-3 and N2H4 propellants. During transfer and mission operational phases, the satellite is three-axis controlled according to a classic gyro-stellar architecture. The only actuator used for orientation and pointing of the satellite is a set of four 1 Nms reaction wheels mounted in a tetrahedral configuration (one ensuring redundancy). Chemical propulsion is used for wheels off-loading. Attitude sensing is performed by a self-redundant inertial measurement unit (IMU) and a redundant narrow-field star tracker. The IMU and the electronic part of the star tracker are part of the highly integrated avionics package. A coarse sun sensor is used to detect exit from eclipse without any a priori knowledge of the position along the orbit being needed, and to ensure solar array pointing towards the Sun. An innovative horizon sensor for accurate positioning around Europa will be required. Considerable developments are required to ensure compatibility of the star tracker and horizon sensor with the high radiation environment. Some star tracker designs include an optical head design that provides a shielding equivalent of tens of mm Al around the detector, which would significantly reduce the radiation dose of the sensor. A small optical navigation camera is required (approximately 24 micro-rad per pixel resolution is needed) for the navigation of the spacecraft. 5 kg of propellant have been included for the AOCS manoeuvres.

As mentioned before, an additional complication applies to JEO: as the orbiter will impact on Europa, the JME will have to comply with the COSPAR requirements on planetary protection for Europa, imposing limitations on the usable materials as well as complex assembly and integration operations, which will be a design and cost driver.

JME-5.3 The Jupiter Relay Satellite

The JRS has several tasks:

- Provide the propulsive manoeuvres from the highly elliptical Earth orbit to the JOI for the composite and subsequent manoeuvres to place the JRS in its final orbit around Jupiter
- Relay science and telemetry data from JEO and JRS to Earth and back
- Carry a payload suite to study the Jovian System
- Carry all subsystems that are not required on JEO, as it is exposed to much less radiation

The JRS will perform most of the spacecraft composite's propulsive manoeuvres and will finally assume a highly elliptical equatorial orbit around Jupiter, outside its main radiation belt. From here its payload suite (17 kg/11.5 W, see JME-3.2) will perform science observations of the Jovian System. This will allow the JRS to survive up to two years once it reaches its final orbit. As a result of the large propulsive manoeuvres, the JRS is considerably larger than the JEO, as can be seen in Figure 42 (JEO is the small cube placed on top of the elongated JRS). Figure 45 shows a cutaway view of the JRS.

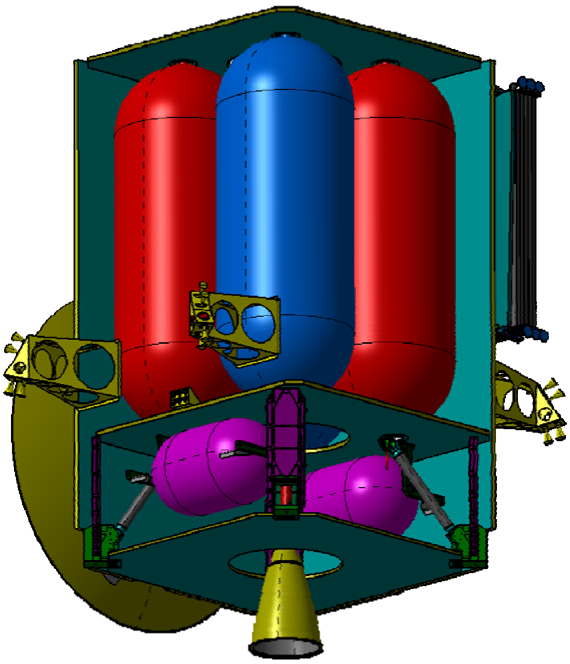

Figure 45: Cutaway view of the Jovian Relay Spacecraft

The JRS will require less shielding to survive for two years when compared to JEO, which survives for about 60 days. Even if the JRS requires more power than the JEO due to the higher performance required for the comms system, the same solar panel surface will suffice (~15 m^2 for approximately 300 W EOL), as the panels will suffer less degradation in 2 years, than the JEO in 60 days. Again, this is valid assuming that the required LILT solar cells can be developed.

The following figure shows the functional architecture of the JRS spacecraft:

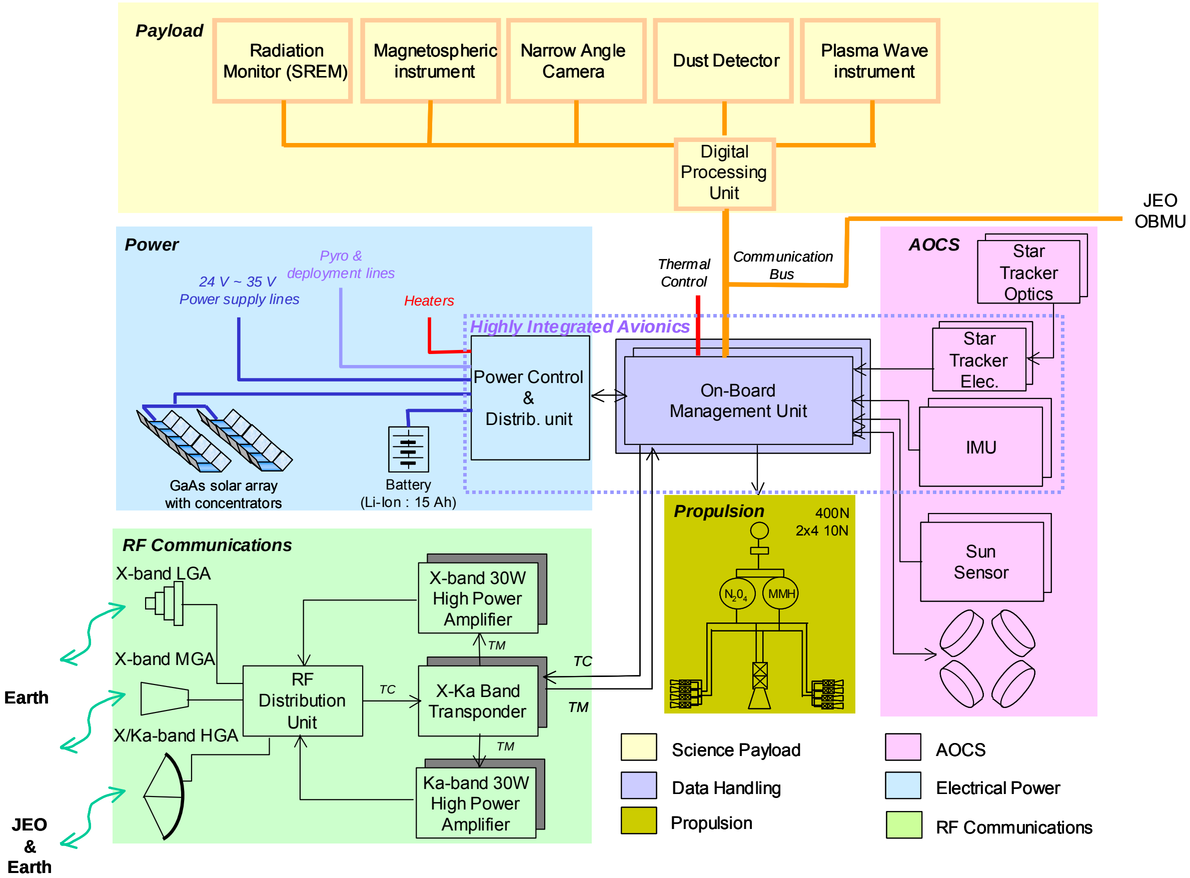

Figure 46: Schematic of the JRS system [10]

The solar arrays require to be pointed towards the Sun by a Solar Array Drive Mechanism during interplanetary cruise and while performing Jupiter science operations while pointing towards Jupiter. During eclipses, which are only expected in the vicinity of Earth and possibly during gravity assists, the required power is also supplied by 15 Ah Li-Ion batteries. Due to the eclipse free orbit of JRS in the Jovian System, a smaller battery module is required (around 400 Wh BOL, mainly for LEOP eclipses). Battery charge management is performed by the same PCDU used on JEO. The resulting 28 V regulated power bus is distributed to all spacecraft users by solid state switches located in this unit. The PCDU is also responsible for pyro commands generation. The PCDU is integrated in the Highly Integrated Avionics package.

The JRS data handling architecture is identical to the JEO architecture, without the dual mode configuration and is also part of the highly integrated avionics package.

RF communications are performed in X-and Ka-Bands. X-band is dedicated to ground telecommands (7 GHz), to telecommands sent to JEO (8 GHz) and for telemetry in early operations (LEOP). Ka-band (32 GHz) is dedicated to both housekeeping and science telemetry, in accordance with ESA standards. A 1.5m diameter HGA is used for communication, whereas an LGA and an MGA are used during LEOP operations. The MGA is also intended for backup telecommand reception at large distances from Earth. TM/TC with Earth is performed at a data rate of 2kbps, while science telemetry with JEO and Earth is achieved at data rates of 2.3Mbps and 30kbps respectively. As for the JEO, new technology developments are required on the JRS communications system on high data rate receiver. A down converter from 32GHz to 7GHz on the telemetry receiving channel permits to use the same transponder on JRS as on JEO.

The propulsion system of the JRS (and for the entire composite) is based on a 400 N main engine operating with N_2O_4 and MMH propellants (I_{sp}= 320 s), equivalent to the performance of a European S400 series engine. It would be advantageous to target a performance of >323 seconds. This would increase mass margin by at least 24kg. The baseline architecture uses 8 pairs of 10 N thrusters to provide thrust in all axes and pure torques.

During transfer and mission operational phases, the satellite is three-axis controlled according to a classic gyro-stellar architecture. The only actuator used for orientation and pointing of the satellite is a set of four 1 Nms reaction wheels mounted in a tetrahedral configuration (one ensuring redundancy). 8 redundant 10N thrusters are used for wheels off-loading. This configuration permits pure torque generation in order to minimize the perturbations on the interplanetary trajectory (most interplanetary spacecraft use only 8 thrusters (4 pairs) for manoeuvres and attitude control, but this does produce cross-couplings, and there is only one axis for pure thrust.). Attitude sensing is performed by a self-redundant IMU and a redundant narrow-field star tracker. As for JEO, the IMU and the electronics part of the star tracker belong to the highly integrated avionics package. A coarse sun sensor is used to detect exit from eclipse without any a priori knowledge of the position along the orbit being needed, and to ensure solar array pointing towards the Sun. The same challenges apply for the JRS star tracker as for the JEO system. Also as for the JME, a small optical navigation camera is required (also here, approximately 24 micro-rad per pixel resolution is needed) for the navigation of the spacecraft. 5 kg of propellant have been included for the AOCS manoeuvres.

The relay spacecraft will have a memory storage capability of approximately 320 Gbit. This will allow for the storage of JEO and JRS data obtained during the operational phase of the mission. Unfavourable alignment with Earth will preclude continuous data transmission: the foreseen data rate of 30kbps to Earth through a 1.5 m HGA will require up to 290 days to send the JEO data gathered during its 60 day mission. Since the JRS will have its own payload suite, additional storage is required: as the scientific interest will be focussed on the Europa science data, the JRS science data transmission will have lower priority than the JEO data. Multiple scenarios can be envisaged for relaying the data to Earth: either all JEO data is first sent before sending any JRS measurements, or a priority ratio is established: e.g. 80% of JEO data and 20% of JRS data. In this way it would take approximately 20% more time to send down all JEO data, but this would give the JRS payload PI's data to work with in the first year.

The mass budgets for both JEO and JRS is summarised in Table 28. More characteristics can be found in Table 3.

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JME-5.4 Overall mass budget

The present analysis has shown that the budgets are very sensitive to variations in the mission variables, such as the actual radiation dose, orbital parameters, science operation duration and especially payload mass. The current launcher margin includes a system margin of 20% along with a component margin of 5%-20%, depending on the technology readiness level of the component. The current mass budget is shown in Table 28. The Soyuz-Fregat Performance from Kourou into the GTO like orbit is taken from [11]. This will have to be verified with the official manual once it is available.

Table 28: The JME mass budget including margins (status November 04)

Item	Mass including margin (kg)
JEO platform mass	389
JEO science instruments mass*	43
JEO dry mass	**432**
JEO propellant mass	198
JEO wet mass	**630**
JRS platform mass	608
JRS science instruments mass*	20
JRS dry mass	**628**
JRS propellant mass	1604
JRS wet mass	**2232**
Total JME mass	**2862**
Adapter mass	40
Total launch mass	**2902**
Launcher capacity	3090
Margin w.r.t. launcher capacity	188 (=6%)

* = JEO and JRS payload of 36 and 17 kg respectively including 20% margin

This table shows that with the current assumptions a launcher margin (i.e. additional launch capability of the launcher after calculating the subsystem and system mass) of more than 6% is available.

In order to understand the potential of this additional mass on the dry mass of the two spacecraft, a sensitivity analysis was performed. Figure 47 shows the effect of the increase of dry mass on the launch mass, if it is fully used for payload. The graph shows the effect for different increases in payload power requirements, varying from 0W to 20W. These graphs include the required increase in solar panel size and ensuing mass increases on the spacecraft.

The table shows that, depending on the additional power required, an increase between 34 kg and 42 kg can be expected (with respect to the 40 kg and 20 kg currently included for JEO and JRS respectively). At this point a word of caution must be said: in view of the high radiation environment, it is well possible that part of this mass must be used for additional shielding or other unforeseen occurrences. The increase in payload mass presented here is therefore indicative.

Preliminary analysis has shown that electric propulsion has the potential to increase the mass in the Jovian System by approximately 100 kg [23]. However the increased cost and complexity must not be underestimated. This option should only be considered if the launcher performance is exceeded with the all chemical approach.

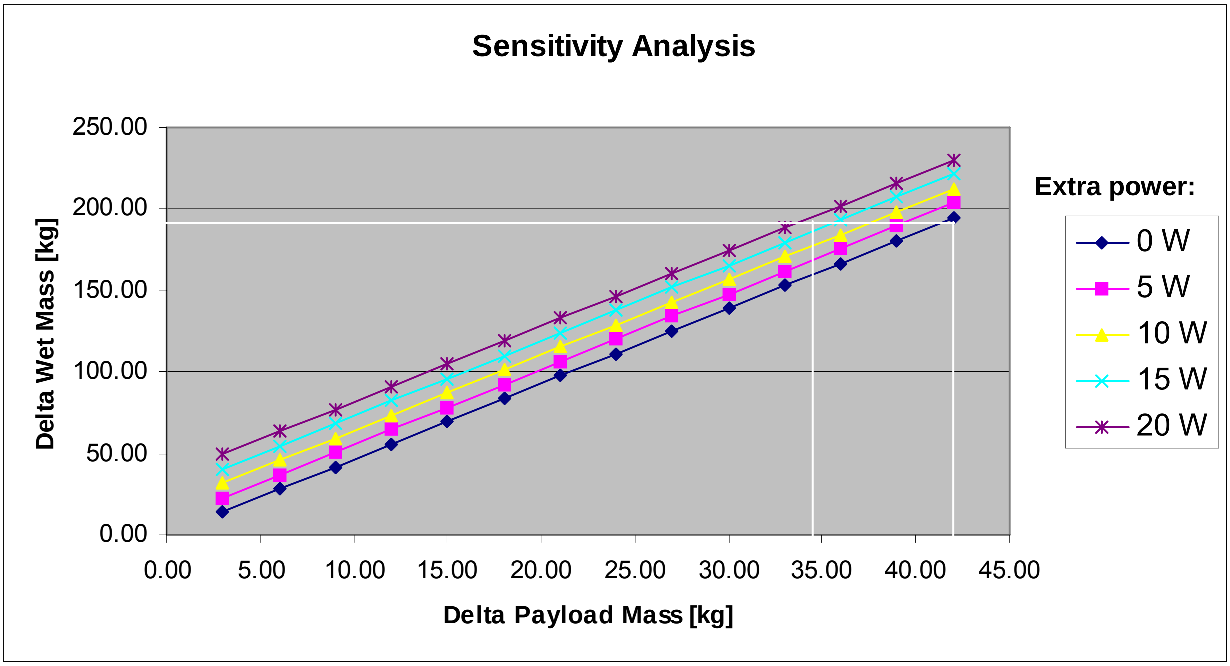


Figure 47: Sensitivity analysis of payload mass increase of the JEO and JRS on the mass [11]

JME-5.5 Power budgets

The following two tables provide the power budgets for JEO and JRS. Peak power of JRS is in excess of 400 W, however, the average values are well below 300 W. The resulting power design (solar arrays and batteries) are fully compatible with the modes shown in the tables.

		Thermal	AOCS	Comms	Propulsion	DHS	Mech	JRS Payload	Harness (excl. PSS)	TOTAL CONSUMPTION	
Cruise without Comms	Pon	103 W	18 W	11 W	106 W	59 W	8 W	6 W	6 W	317 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	75 %	100 %	9 %	86 %	0 %	13 %		61%	193.1 W
Tref 3615975 min	Total Wh	6207424 Wh	795515 Wh	638822 Wh	578556 Wh	3078888 Wh	60266 Wh	51226 Wh	228214 Wh	11638911 Wh	
Cruise with Comms	Pon	37 W	18 W	85 W	106 W	59 W	8 W	6 W	6 W	325 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	75 %	100 %	9 %	86 %	0 %	13 %		62%	201.5 W
Tref 328725 min	Total Wh	202714 Wh	72320 Wh	464598 Wh	52596 Wh	279899 Wh	5479 Wh	4657 Wh	21645 Wh	1103907 Wh	
Jupiter Orbit Insertion	Pon	0 W	18 W	85 W	106 W	59 W	0 W	6 W	5 W	279 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	0 %	0 %	100 %	100 %	100 %	0 %	0 %		91%	254.7 W
Tref 10 min	Total Wh	0 Wh	0 Wh	14 Wh	18 Wh	10 Wh	0 Wh	0 Wh	1 Wh	42 Wh	
			13	0	106	51					
Comms with Earth	Pon	72 W	18 W	167 W	106 W	59 W	0 W	6 W	9 W	436 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	100 %	75 %	40 %	9 %	86 %	0 %	13 %		50%	217.7 W
Tref 2144 min	Total Wh	2573 Wh	472 Wh	2384 Wh	343 Wh	1826 Wh	0 Wh	30 Wh	153 Wh	7780 Wh	
			5	11	10	40					
Science	Pon	72 W	18 W	11 W	106 W	59 W	8 W	6 W	6 W	285 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	75 %	100 %	9 %	86 %	0 %	100 %		58%	167.2 W
Tref 2144 min	Total Wh	2573 Wh	472 Wh	379 Wh	343 Wh	1826 Wh	36 Wh	230 Wh	117 Wh	5975 Wh	
Comms with JEO	Pon	72 W	18 W	85 W	106 W	59 W	0 W	6 W	7 W	353 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	100 %	75 %	100 %	9 %	86 %	0 %	13 %		67%	236.2 W
Tref 2144 min	Total Wh	2573 Wh	472 Wh	3030 Wh	343 Wh	1826 Wh	0 Wh	30 Wh	165 Wh	8439 Wh	

Figure 48: JRS power budget [12]

		Thermal	AOCS	Comms	Propulsion	DHS	Mech	JEO Payload	Harness (excl. PSS)	TOTAL CONSUMPTION	
Cruise without Comms	Pon	78 W	18 W	11 W	10 W	59 W	0 W	37 W	4 W	216 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	100 %	0 %	10 %	0 %	72 %	0 %	0 %		58%	124.4 W
Tref 3615975 min	Total Wh	4700768 Wh	0 Wh	63882 Wh	0 Wh	2585193 Wh	0 Wh	0 Wh	146997 Wh	7496839 Wh	
Cruise with Comms	Pon	34 W	18 W	40 W	10 W	59 W	8 W	37 W	4 W	209 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	0 %	10 %	0 %	86 %	0 %	0 %		43%	91.5 W
Tref 328725 min	Total Wh	184086 Wh	0 Wh	22134 Wh	0 Wh	279899 Wh	5479 Wh	0 Wh	9832 Wh	501430 Wh	
Transfer	Pon	36 W	18 W	32 W	10 W	59 W	8 W	37 W	4 W	203 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	75 %	100 %	0 %	100 %	0 %	13 %		73%	148.6 W
Tref 784800 min	Total Wh	470880 Wh	172656 Wh	413328 Wh	0 Wh	775382 Wh	13080 Wh	60430 Wh	38115 Wh	1943871 Wh	
			0	0	10	51					
Europa Orbit Insertion	Pon	0 W	18 W	11 W	10 W	59 W	0 W	37 W	3 W	137 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	0 %	0 %	100 %	100 %	86 %	0 %	0 %		53%	72.7 W
Tref 10 min	Total Wh	0 Wh	0 Wh	2 Wh	2 Wh	9 Wh	0 Wh	0 Wh	0 Wh	12 Wh	
			5	11	10	40					
Science 1 with Sun	Pon	18 W	18 W	11 W	10 W	59 W	8 W	37 W	3 W	163 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	75 %	0 %	0 %	86 %	0 %	75 %		69%	113.1 W
Tref 5114 min	Total Wh	1534 Wh	1125 Wh	0 Wh	0 Wh	4354 Wh	85 Wh	2354 Wh	189 Wh	9642 Wh	
Science 1 eclipse	Pon	42 W	18 W	11 W	10 W	59 W	0 W	37 W	4 W	179 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
Eclipse Mode	Duty Cycle	100 %	75 %	0 %	0 %	86 %	0 %	75 %		76%	136.6 W
Tref 1114 min	Total Wh	780 Wh	245 Wh	0 Wh	0 Wh	949 Wh	0 Wh	513 Wh	50 Wh	2536 Wh	
Science 2 with Sun	Pon	18 W	18 W	11 W	10 W	59 W	8 W	37 W	3 W	163 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	1 W	0 W	0 W	1 W	Paverage
	Duty Cycle	100 %	75 %	0 %	0 %	86 %	0 %	32 %		59%	97.1 W
Tref 5114 min	Total Wh	1534 Wh	1125 Wh	0 Wh	0 Wh	4354 Wh	85 Wh	1011 Wh	162 Wh	8272 Wh	
Science 2 eclipse	Pon	42 W	18 W	11 W	10 W	59 W	0 W	37 W	4 W	179 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	100 %	75 %	0 %	0 %	86 %	0 %	32 %		67%	120.5 W
Tref 1114 min	Total Wh	780 Wh	245 Wh	0 Wh	0 Wh	949 Wh	0 Wh	220 Wh	44 Wh	2238 Wh	
Comms with JRS with Sun	Pon	18 W	18 W	40 W	10 W	59 W	0 W	37 W	4 W	185 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	100 %	75 %	100 %	0 %	86 %	0 %	0 %		68%	125.1 W
Tref 336 min	Total Wh	101 Wh	74 Wh	226 Wh	0 Wh	286 Wh	0 Wh	0 Wh	14 Wh	701 Wh	
Comms with JRS eclipse	Pon	42 W	18 W	40 W	0 W	59 W	0 W	37 W	4 W	200 W	
	Pstdby	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	0 W	Paverage
	Duty Cycle	100 %	75 %	100 %	0 %	86 %	0 %	0 %		75%	149.6 W
Tref 184 min	Total Wh	129 Wh	40 Wh	124 Wh	0 Wh	157 Wh	0 Wh	0 Wh	9 Wh	459 Wh	

Figure 49: JEO power budget for the different operational modes [12]

JME-5.6 Data transfer assumptions

As mentioned before, the spacecraft memory is based on the expected data generation and the communication windows. This analysis was based on the following assumptions: The scientific mission of the JEO was assumed to be the maximum duration specified for this mission, i.e. 66 days. JEO generates approximately 50 Gbit of data every 10.7 days, which is the period of the JRS orbit. In view of these 50 Gbit, a 64 Gbit memory module has been selected. It is a possibility to double this memory in case of a problem during a transmission opportunity, to avoid the loss of gathered data. The excess data will have to be transferred during a number of subsequent JEO JRS communication opportunities. Once the JEO mission ends on Europa's surface, the JRS will survive up to two more years and will downlink both JEO and JRS data.

The total data volume generated during the 66 days will be 311 Gbit. This data will be stored on the JRS and subsequently relayed to Earth. The 320 Gbit memory allocation will allow for the storage of both JEO and JRS data. With the current power available for comms, the 1.5 m high gain antenna and the given distances and losses, a 30 kbps link will be available between JRS and Earth. Assuming a daily ground station allocation of 8 hours for the downlink of data, would allow 7 hours for science data relay and 1 hour for housekeeping data. With these conditions, 411 days would be required to send the JEO data to Earth; this means 345 days after the JEO mission has ended. As the JRS data will have to be downloaded as well, more time will obviously be required. The transfer time of the science data can only be decreased by increasing the ground station availability (e.g. two ground stations per day), however the cost of the additional station will increase considerably, which needs to be taken into account.

Assuming that the JRS can survive another year, the JRS could generate another 300 Gbit of data before its end of life. Both the JEO and JRS data volume can be increased accordingly, if the ground station availability is increased.

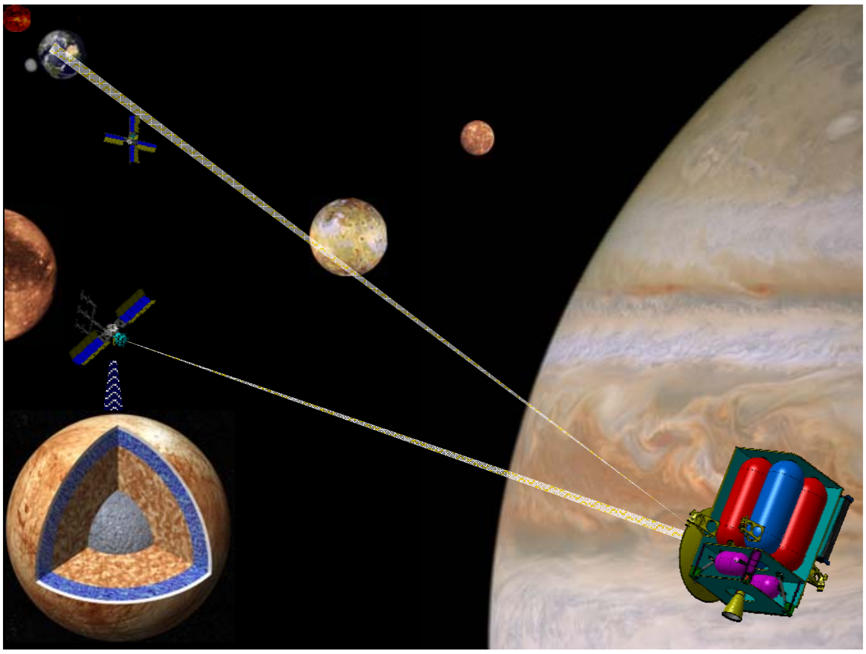


Figure 50: Schematic view of JEO-JRS-Earth communication sequence

JME-6 SUMMARY OF THE MAIN CHALLENGES

The JME Technology Reference Study is intended to identify the technologies required for a mission to the Jovian System or other outer planets. It is clear that a similar mission stretches the limits of our current technology and serious development efforts are required to enable a mission to the Jovian System. The development of these technologies and techniques will have to start within the next few years to make sure that such a mission can be achieved within the presented timeframe. Currently the following issues have been identified:

Radiation: The spacecraft electronics need protection against the radiation levels in excess of 6 Mrad after 4mm of Al. To avoid an excessive shielding mass, radiation hardened electronics will be required in the order of 1 Mrad. This will require extensive development in this field, making it the main challenge identified in this study. An effort is also required to identify the best suited radiation material, as well as the design and qualification of the shielding structure. The design of an integrated avionics box capable of operating up to a 1 Mrad dose at low mass will also be a must.

Power generation: The 5.2 AU distance from the Sun results in $1/25^{th}$ of the solar flux. Therefore solar power generators need to be compatible with low intensity, low temperature (GaAs LILT cells are foreseen) and very high fluence. This will require a new and costly development. In order to increase the efficiency of the solar arrays, solar concentrators will be required (see section JME-5.2). Presently a patent exists on this technology, deposited by Boeing, an issue that will need to be addressed in future. Problems with this particular concept need to be understood and solved, but alternative solutions shall be assessed as well. Should these developments prove to be unfeasible, RTG technology will have to be taken into consideration. The implications of an RTG system on the design have been assessed (see next chapter).

Material compatibility: Another issue that needs to be addressed is the behaviour of the selected spacecraft materials in this extreme radiation environment. The selected JEO materials shall also be compatible with the cleanliness requirements dictated by the COSPAR planetary protection rules.

Thermal: The spacecraft must be compatible with both the hot (Venus fly-by) and the cold case (Jovian System). The use of active heat transfer (fluid loop) could be required.

Low resource: Even if LILT cells and solar concentrators will be available, the available power will be quite limited. Therefore efficient low resource systems will be required for all subsystems, from communication systems to the payload. One promising way is the development of highly integrated payloads and avionics, which will limit both the mass and power requirements.

Payload: The payload design is based on the Highly Integrated Payload Suite (HIPS) concept, which seems the only approach that makes a small spacecraft approach feasible. This will require considerable developments, both from a technical point of view and from a programmatic point of view. Apart from the development of the required technology, a strong coordination will be required between the institutes responsible for the different payloads. The HIPS concept is currently at conceptual level and must be studied in detail. Activities have recently been started to better understand this concept. Another challenge lies in the development of a foldable Yagi ice penetrating antenna: the high level of folding required must be better understood and the performance of the radar as function of the ice contamination needs to be fully realised.

Communications: New developments will be required to operate in both X- and Ka-band: high data rate (3 Mbps) KA-X transponders for both TC and TM, high efficiency Ka-band SSPA (3.5 W RF, 30% efficient).

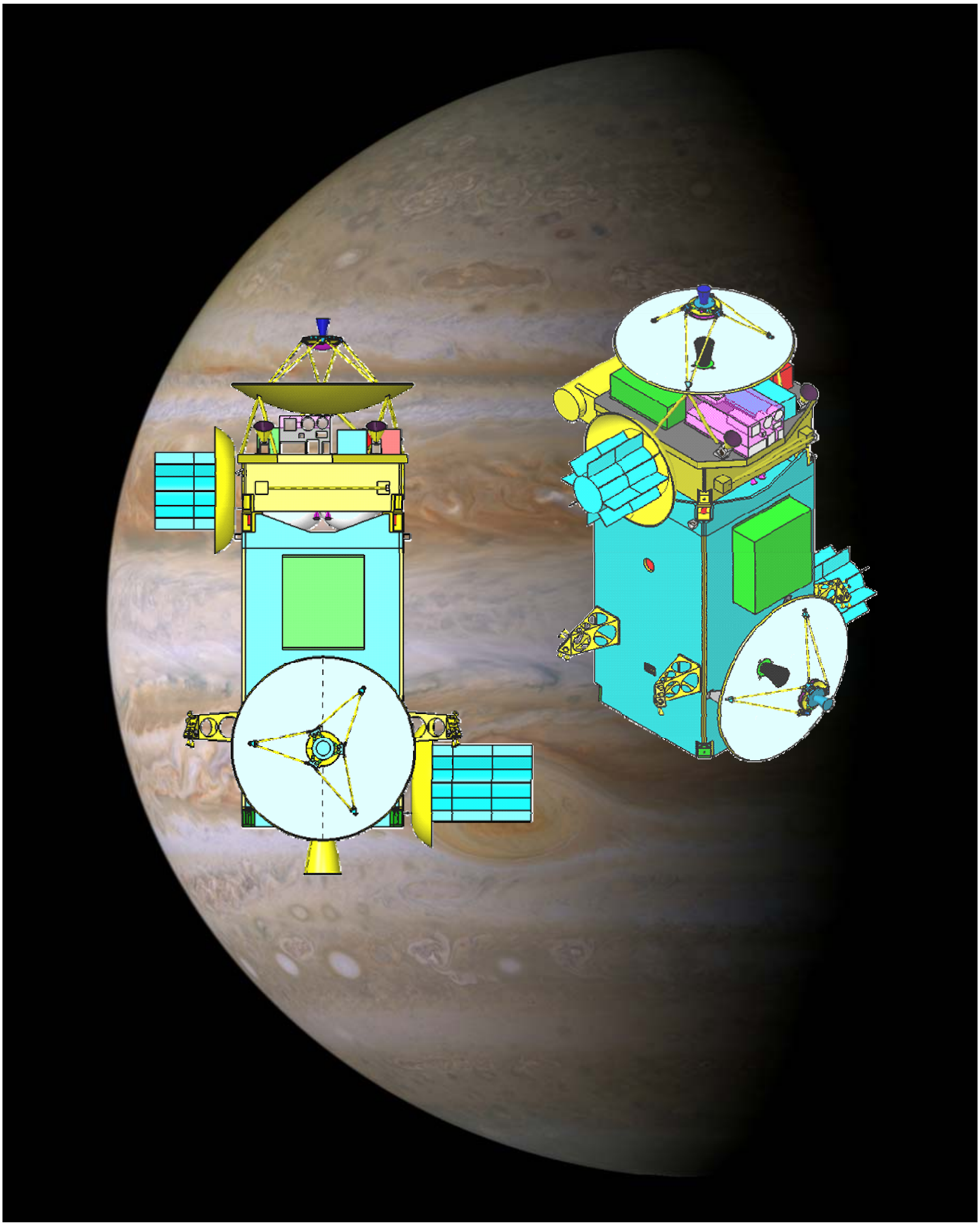
AOCS: The instruments (e.g. the radar) will require accurate nadir and attitude pointing control. Current systems are not compatible with the expected radiation environment, which will pose a considerable technology challenge. Furthermore, a new planet edge detection system will be required, as Europa does not have (sufficient) CO_2 in its exosphere, to allow for conventional limb detection systems. Further complications arise due to the large disk of Jupiter, masking the limb of Europa during the JEO orbit.

Autonomy: The long mission duration and the hostile environment call for a highly autonomous mission capability. The costs of the mission operations for the long mission duration (7+ years for JEO and 9+ years for JRS) including extensive GAMs will be very high. Furthermore the exposure to the very high radiation will most likely cause a significant number of safe modes, upsets, etc. Having a robust autonomous system would preclude the necessity of expensive round the clock presence of spacecraft operators, with the exception of the critical manoeuvres (GAMs and orbit insertions). This will require developments in software, optical navigation, manoeuvre scheduling as well as sun sensors compatible with large solar flux variation. Additional autonomy is required for the instruments: Both the commissioning and the operation will have to be monitored on board as much as possible, in view of the short science mission. The commissioning phase must be very short, to allow for an acceptable operational phase. Furthermore, it will not be possible to receive all data in real time, as most data gathered by JEO will reach Earth after the 60-day operational phase. Intervention in case of problems from ground would be too late by then, as the JEO will have already crashed onto Europa.

Planetary protection: COSPAR planetary protection requirements for Europa are of the highest level. As JEO will impact on Europa, this will impose serious complications for the JEO: it will limit the materials that can be used and will require complex and costly integration procedures. A means must be found to optimise the operations requiring the highest cleanliness demands and allow for integration in environments with less stringent cleanliness, without compromising the achieved decontamination. In-flight decontamination by the severe radiation in the Jovian System must also be exploited as much as possible.

Microprobe impact: Impacting at high initial velocity (in the order of several km/s) on planetary bodies without an atmosphere is unfeasible. Any impactor concept will have to be slowed down to velocities below 500 m/s, requiring a deceleration stage with fully functional AOCS and propulsion subsystems requiring considerable mass. Even this will require materials and subsystems capable of withstanding very high impact shocks (~ 10 000 – 100 000 g, strongly depending of the impact strategy). Relatively low loads (> 10 000 g) are achievable if an attitude control system can be accommodated that will guarantee an impact direction along the longitudinal axis. In this case a specific axial design, tailored to a certain impact direction can be used. This concept will require a complicated AOCS system capable of maintaining the impactor axis perpendicular to the impact surface, which will require additional mass. This solution would be the most indicated approach, however the risk will remain high and such a system would require the majority of the payload resource complement available on the JEO. The alternative is a system with no preferential impact axis, requiring a design that is independent of the attitude. However, analysis has shown that the impact loads will exceed 100 000 g and is therefore deemed unfeasible, unless the impact velocity is significantly reduced below 500 m/s. Should the impactor survive impact, the access of instruments to the outside will be severely limited by the deformation of the impactor structure, making sampling very complex (this is valid for both impactor concepts). The other key problem is lifetime (non RHU design will survive in the order of 10 hours) and the communication with the orbiter. The very high risk of failure for an impactor must be taken into account when considering it against remote sensing payloads on board the spacecraft. See JME-3.3 for more details.

JME-RPS: THE RADIOISOTOPE POWER SOURCE OPTION



RPS-1 INTRODUCTION

After the original JME concept study, it was decided to assess the implications of Radioactive Power Sources (RPS) on the JME spacecraft configuration. The main reason for this was to see what changes were required with respect to the solar powered spacecraft option in case the solar cells would not be able to provide the required power. With this study it was possible to compare the two options and understand their respective benefits and drawbacks. This comparison was not only performed at system concept level, including technology development issues, but also considered management and AIV from the integration phase to the launch phase. The details of this study, performed by EADS Astrium, are based on the configuration in 2005 and therefore they do not reflect the latest status. This is the reason for deviations in the budgets and launch dates with respect to the JME results that are presented in the previous chapters. However the conclusions of the implications of RPS on the design are still fully applicable.

This chapter is based on the paper presented by EADS Astrium at the 2005 IAC conference [24]

RPS-2 THE USE OF RPS

The major advantage of using RPS is the access to an almost constant power source, irrespective of the distance to the Sun and eclipses. For mission concepts beyond Jupiter this is the most likely power source, if not the mission's key enabling technology. It must be noted that RTG systems are currently only developed in the USA and in lesser extent Russia (the Russian Angel RTG targets a much lower performance range of 0.7 We/kg).

One of the key parameters that demonstrate the efficiency of a given technology is the amount of power provided by one kg of equipment of the power system, also called specific power. In [25], it was found that, considering possible enhancement of solar arrays technology, a specific power of up to 3.9 We/kg could be reached (End-Of-Life for the reference mission), without considering the necessary addition of batteries for eclipse periods. With RPS, the efficiency objectives found in the literature show values up to 8We/kg (Beginning-Of-Life). Current RPS systems are however significantly less efficient: 5.2 We/kg (BOL) for the GPHS RTGs (Radioisotope Thermoelectric Generator). The last of these RTGs has been used in the Pluto New Horizons mission and therefore this system is not available anymore. The current development in this field is the MMRTG (Multi Mission RTG), a system that needs to be operable in both vacuum and atmospheric conditions. The resulting design is less efficient than the GPHS RTGs, namely ~3 We/kg (BOL) / ~2.5We/kg (EOL).

A significant drawback of RPS on the mission is linked to the Earth protection measures against radioactivity which requires building a thorough safety plan.

It must be acknowledged that it is difficult to predict the availability of an RPS system in the frame of the JME mission scenario: the first phase of the feasibility study indicated that new types of solar arrays could fulfil the needs in a limited timescale, and at an affordable cost. RPS concern a very sensitive technology, as it is associated with national strategies for defence. It is therefore difficult to be sure of the pertinence of what can be found in the open literature. Future cooperation policies might also include negotiations between many countries interested in nuclear-powered space systems, which might imply timescales not in line with the Jupiter exploration schedule. However this not only depends on technological, but also on political and programmatic aspects.

The advantages (+) and drawbacks (-) are summarized in Table 29.

Table 29: RPS advantages & drawbacks on system level

Technology	+	Specific power potentially twice as good as solar arrays
		Enabling technology valid for other missions beyond Jupiter
	-	Availability
		Potential socio-political opposition
Spacecraft conf.	+	No deployable/movable appendages
	-	Heavy stiff external mounting structure
Power system	+	JEO power independant of Sunlight conditions
		Very low power degradation
Thermal control	+	Use of RPS heat for the spacecraft
		No thermal problem on RPS close to Venus
	-	Specific cooling required under fairing
		Additional shielding to protect S/C against heat
		RPS cannot be switched off
Radiation	-	Additional shielding required
Instruments operation	+	No operation limitations (no power degradation)
		No constraint on orbit selection (eclipse duration)
	-	Potential micro-vibrations and magnetic pollution with dynamic energy converters
Safety	-	Safety planand complex AIV issues
		Higher propellant budget imposed by Earth fly-bys at higher altitudes for planetary protection
Cost	-	RPS are (far) more expensive

RPS-3 RPS TECHNOLOGY

RPS-3.1 Current state

The current generation of RPS are Radio-Thermal Generators (RTG). These RTGs flew on Galileo, Ulysses, Cassini and Pluto New Horizons as shown in the following figures. As mentioned before, this kind of RTG is not available anymore.

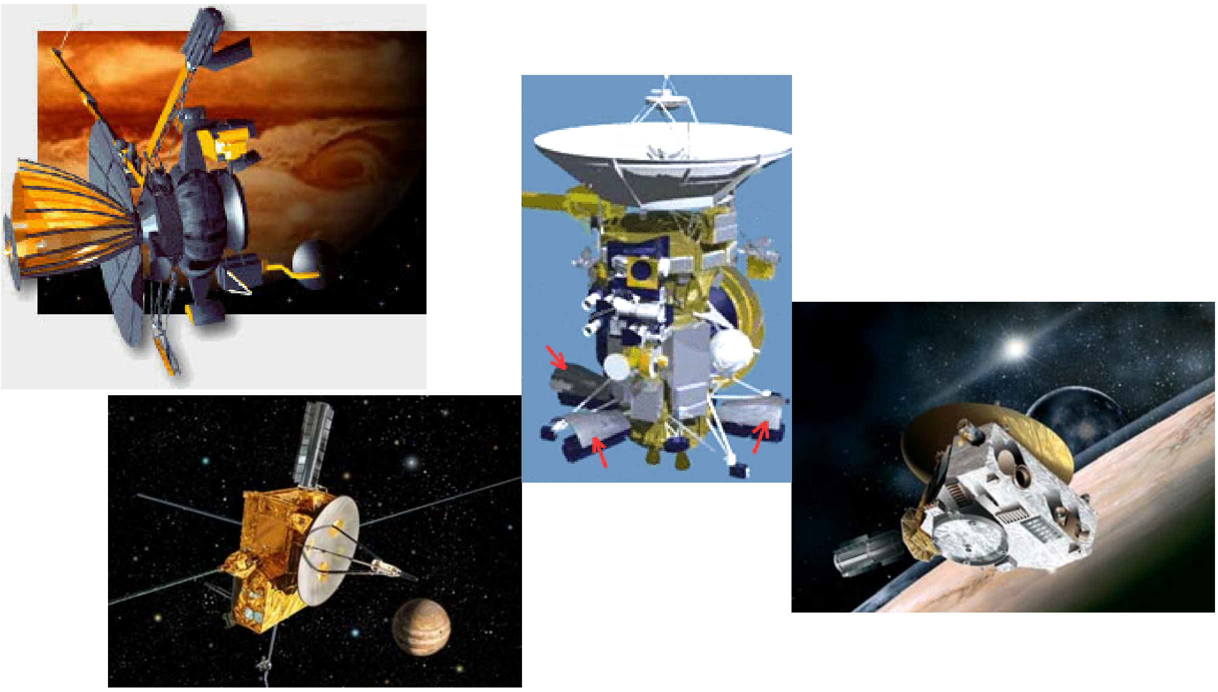


Figure 51: GPHS RTG powered missions

RPS-3.2 New generation under development

New Multi-Mission RTG (MMRTG) and Stirling Radioisotope Generator (SRG) are being developed in the USA and are scheduled for completion by 2009. The assigned objectives for a second generation of these RPS have already been identified [26,27], as shown in the following figure.

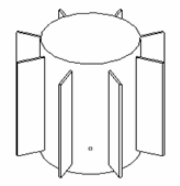
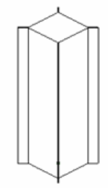
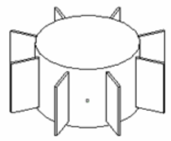
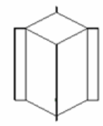
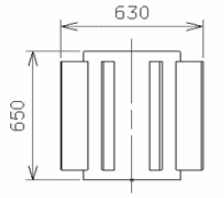
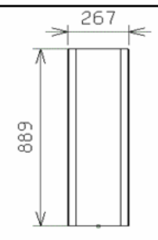
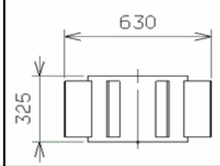
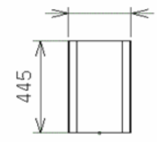
	Under development 2009		Supposed > 2015	
	MMRTG	SRG	MMRTG 2nd generation	SRG 2nd generation
Isometric view				
Envelope			Assumed dimensions 	Assumed dimensions 
Mass	34 kg	27 kg	17 kg	14 kg
We after 9 years	108 W	100 W	80 W	83 W
H/W cost	20 M$	15 M$	Assumed at 20 M$	Assumed at 15 M$
Nr of GPHS modules	8	2	4	1

Figure 52: US R&D on MMRTG and SRG

RPS-3.3 Expectations for future RPS

Beyond those developments, there exist other technologies such as Brayton Radioisotope Generators (BRG), Radio-isotope Thermo-Photo-Voltaic (RTPV) and Alkali Metal Thermal-to-Electric Converters (AMTEC) [28]. Their potential performances are summarized in Table 2.

Table 30: US investigations on future RPS technologies

Technology	Target power conversion efficiency	Target specific power
mini-Turbo Brayton converter	25-36%	9-13 W/kg
Free-Piston Stirling	>30%	>8 W/kg
Segmented Thermoelectric	10%	>5 W/kg
Thermophotovoltaic	15-23%	>8 W/kg

RPS-3.4 European expertise: Ulysses

The Ulysses spacecraft was launched October 6, 1990 with a Space Shuttle on top of an Inertial Upper Stage (IUS), which provided the energy required for the escape orbit to Jupiter.

The installed RTG provided a power of 267W (BOL). Originally the BOL performance of the RTG would have been 292W, but the Ulysses launch was delayed by 4 years due to the Challenger accident, leading to a decay of the performance.

Ulysses carried 10.75 kg of plutonium dioxide, corresponding to a total activity level of 1.325×10^5 Curie and 4.4kW of heat. The loss in thermal power was approximately 0.8% per year. The power regulation was performed by dumping excess power to dedicated heaters.

As the RTG system cannot be switched off, the spacecraft units were powered on all the time, and by that a near thermal equilibrium was established. The only changing temperature source was the varying distance to the Sun from 1.2AU to 5.2AU.

The temperature of the RTG itself was quite high, namely 270°C on the fins when fully operational. This was a major problem for the integration into the Shuttle cargo bay. At the time of integration the RTG system was cooled and pressurized by a special fluid loop system. Inside the Shuttle cargo bay, a reservoir was installed for cooling as it can be seen in the next figure. After separation from the IUS, the RTG fluid system was separated after which the RTG was depressurised. This allowed the particle emission to reach its maximum value.

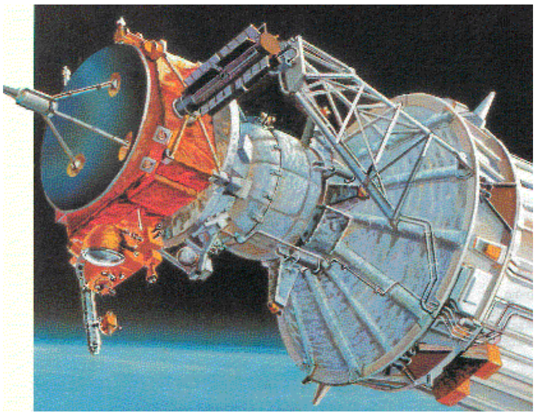


Figure 53: Ulysses with RTG cooling connections on the IUS after separation from the Space shuttle

The subsystems and instruments had to be protected against the radiation effects (mostly neutrons) from the RTG. Therefore all instruments were mounted on the side opposite to the RTG. Special protection was obtained by a tungsten shield for very sensitive instrument electronics. The electronics were selected to be radiation hard to 100krad.

The safety for personnel was a major issue and special medical investigations were performed from before the RTG AIV until 1 year after the mounting took place. All personnel that installed the RTG were wearing special body dosimeters and special finger ring dosimeters.
Special training sessions were held to train the mounting procedure in detail, limiting the exposure to the real RTG later on, as shown in the next figure. The mounting team consisted of 2 mechanical

and 2 electrical engineers. The actual time spent for the real integration was below 2 min per person and in the end the dosimeters did not indicate any measurable exposure to radiation.

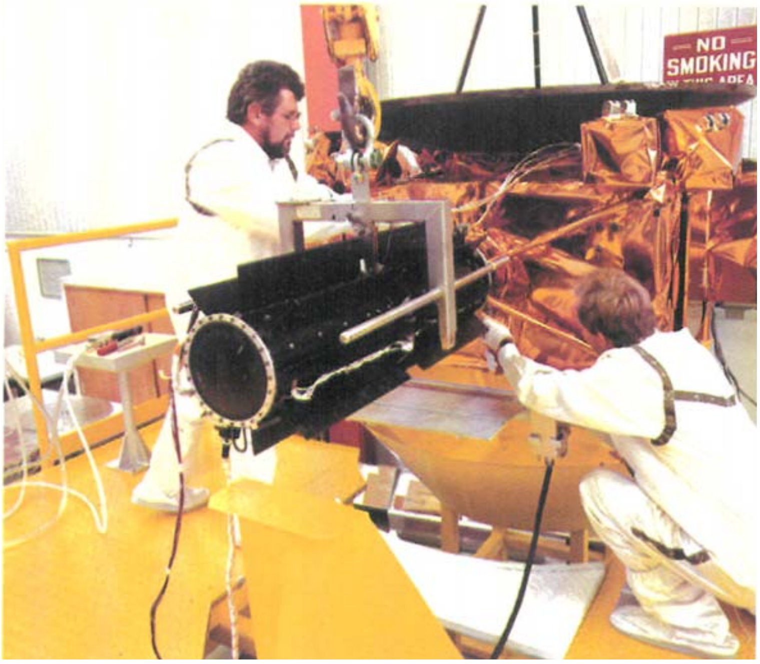


Figure 54: RTG dummy installation training with heated RTG at 70°C. Mounting tools are air driven tools

RPS-4 IMPACT ON TRAJECTORY

RPS-4.1 Drivers

The implication of RPS is that the risk of any accidental impact of the spacecraft on Earth must be minimised. There are certain phases of the baseline mission that require operations close to Earth. These occur during the main transfer, where gravity assists take place at Earth. Also, in the early mission stages, the spacecraft is injected into a GTO like orbit and apogee raising manoeuvres performed.

RPS-4.2 Risks at Earth fly-bys

Two examples are considered involving launch in 2012 (best case) and 2016 (nominal case). Figure 55 shows the effect on transfer Delta-V. This includes the Earth escape Delta-V from GTO and a Delta-V at Jupiter pericentre to insert to the 0,9Mkm pericentre by 20 Mkm apocentre insertion orbit. An additional Delta-V equal to 150 m/s was baselined for the study for a fly-by perigee close to 3000 km.

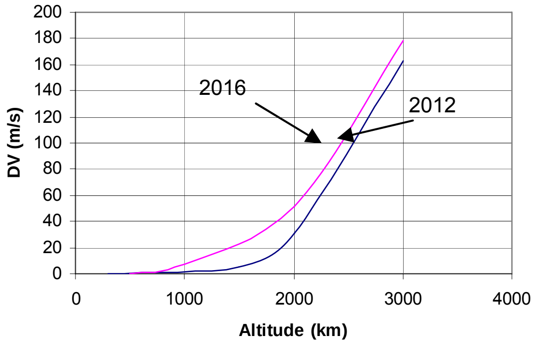


Figure 55: Delta-V penalty for increased altitude Earth fly-bys

The typical 3 sigma dispersion ellipse at a fly-by would be less than 50km. Depending on the type of tracking values below 10 km are achievable. The small size of this dispersion ellipse is achieved via a series of small manoeuvres, taking place over a period of two to four weeks typically, that progressively refine the target, as shown in Figure 56, with tracking periods in-between.

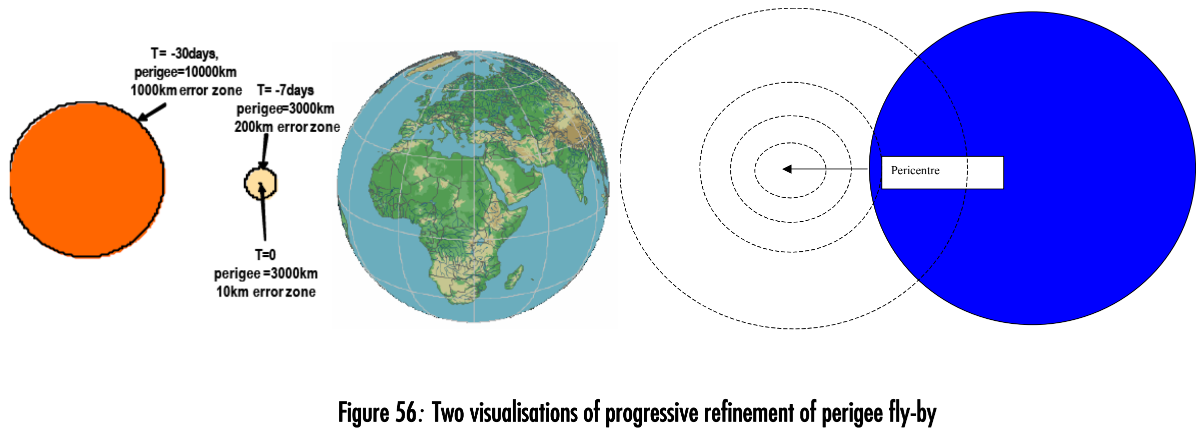


Figure 56: Two visualisations of progressive refinement of perigee fly-by

Therefore the following is applicable:

- A point in the mission exists when the 3 sigma dispersion ellipse is reduced to the extent where it no longer implies Earth impact
- Increasing fly-by altitude means that this point in the mission occurs earlier
- This leads to the extension of the region of the approach timeline over which a spacecraft failure (i.e. preventing dispersion correction manoeuvre taking place) may safely occur
- Alternatively, if a failure occurred before this threshold distance that leaves the possibility of Earth impact open, a longer period exists to recovery of the error. A delayed manoeuvre is then possible to guarantee Earth avoidance (with a small Delta-V penalty)

RPS-4.3 Risk mitigation by apogee raising

Soyuz-Fregat would normally inject into a GTO-like orbit with approximately 200 km perigee and apogee in the range of 42 000 to 60 000 km. Therefore risk mitigation strategies have to be identified to avoid Earth impact in case of a total failure occurring when in GTO or in an intermediate apogee orbit.

Significant perturbations are:

- Lunar solar gravity: perturbs the perigee altitude. The perturbation is strongly dependent on the relationship between the line of apses and the node of the Lunar orbit;
- Atmospheric drag: progressively reduces the apogee altitude. The perigee altitude has a strong influence on this rate. Typical lifetimes are:
  - Perigee altitude 200km: 15 years
  - Perigee altitude 300km: 160 years

In a GTO case, 200km would be too low to achieve an orbit life time long enough not to endanger Earth. Raising the perigee to 500km would ensure that the orbit would have a lifetime of hundreds of years. The cost of raising the perigee would be 30m/s. Apogee raising from this higher perigee would also cost a further 35m/s. In the higher apogee case (66000 km), the perigee would need to be raised to 700km at a cost of 35m/s. Apogee raising from this higher perigee would also cost a further 55m/s.

However, although it is possible to raise perigee to make the orbit stable for hundreds of years in the event of spacecraft failure, the preceding Delta-V analysis assumes that the spacecraft is functional to be able to do this. Alternatively, the launcher could be used to perform a higher perigee injection. However, the injection mass penalty can be high for raised perigee. A Delta-V equal to 80m/s was baselined for the study.

RPS-5 IMPACT ON SPACECRAFT DESIGN

RPS-5.1 Configuration

By switching to RPS instead of solar arrays for on-board power generation, the following equipments have been removed:

- solar array, solar array drive mechanisms & hold-down points on both JRS & JEO
- one battery on JEO

The following equipments have been added:

- 3 RPS stacked together on JRS
- 2 RPS stacked together on JEO
- thermal & radiation shielding on JRS & JEO
- tripods for mounting RPS on JRS & JEO

The main design drivers are:

- need for dissipating RPS heat towards space
- need for RPS rapid mounting/dismounting in the last sequences before launch
- Soyuz fairing constraints
- RPS position permitting cooling under fairing before launch
- RPS position limiting thermal impact on Fregat (the Fregat needs to be maintained in the range 10°C-12°C)

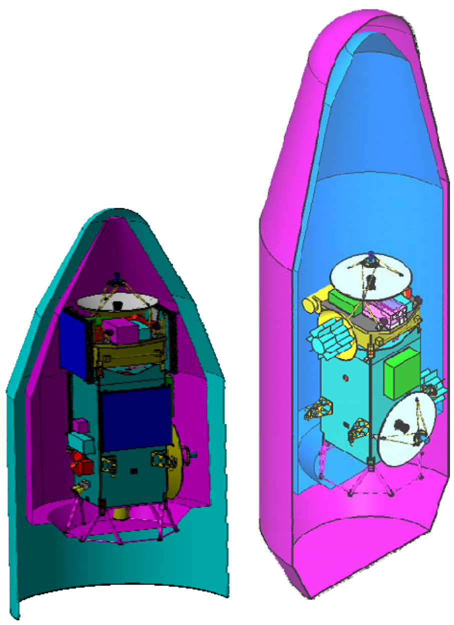


Figure 57: JME with Solar arrays and with RPS respectively under Soyuz S and ST fairing

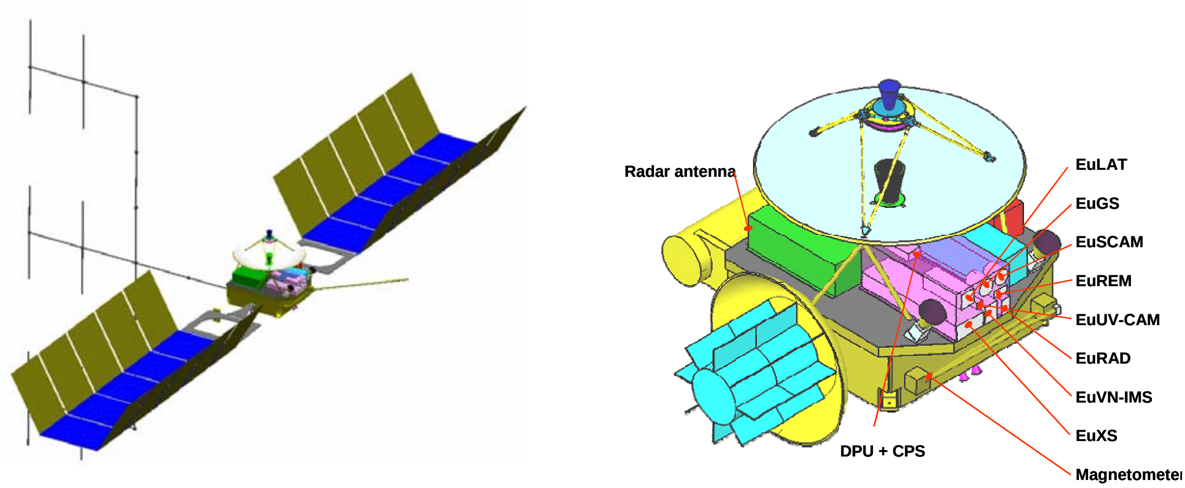


Figure 58: JEO configuration with solar arrays and the alternative with RPS (with folded radar)

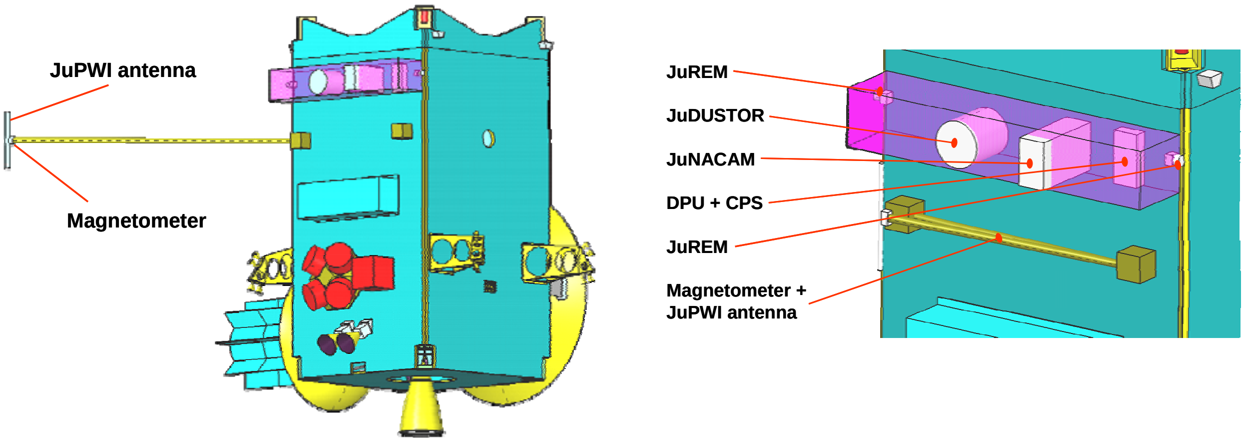


Figure 59: JRS alternative configuration with RPS

RPS-5.2 Thermal

In order to save internal power and minimize the number of required RPS (3 for JRS & 2 for JEO), a promising solution would be to use the RPS thermal dissipation to heat units with the help of Loop heat-pipes (LHP). The Astrium Mini-LHPs presented in Figure 60 already flew on FOTON M2 in 2005. Such LHP allows transferring up to 50W for a mass of 100g. It can be easily mounted on a conductive plate in contact with the RPS. Heat transfer can be controlled (on/off) by using a dedicated heater (max 1W) on LHP reservoir.

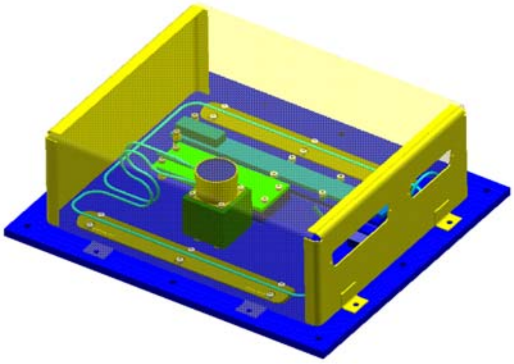


Figure 60: Mini-LHP flown on FOTON M2 (2005)

Two main thermal issues have been identified as critical with respect to RPS use for JME mission:

- Launcher environment
- spacecraft hot flight environment (Venus Flybys)

Before launch, STARSEM, as baseline, supplies the V-SOTR ventilation system which is able to process the JME RPS thermal dissipation. It is available in the clean room and under the fairing with a flow rate of 7000 m^3/hour. However, due to launcher operation constraints, there are two service interruptions of V-SOTR identified for which an alternative cooling system is mandatory:

- Transfer & launcher erection, after integration of spacecraft on FREGAT and fairing mounting: V-SOTR interruption of 8 hours
- Last launch sequence: 2 hours until take off in nominal conditions, up to 4 hours in case of aborted launch

Such service can be provided by a specific adaptation of the existing G-SOTR system:

- the hot source can be located at RPS bottom attachment;
- during transfer & launcher erection, the cold source can be connected to an external portable tank;
- on the launch pad, the cold source can be connected to a ground tank, with a dedicated routing all through the launcher to keep the connection during countdown sequence until take-off (the control or RPS temperature needs to be verified up to the end)

Close to Venus the temperature will increase due to the closer Sun. This point was already analysed in the previous study for the spacecraft itself. Regarding RPS integrity & operation however, no criticality is identified so far (to be confirmed once the actual RPS design specifications are available).

RPS-5.3 JME comparative budgets

The budgets of the JME with RPS are now compared to the ones of the study with solar arrays. The main changes correspond to the removal of solar array drive mechanism, heating power provided by recovery of the RPS dissipation and the JEO battery charging removal.
Table 31 presents worst case conditions, occurring during communications. Payload power needs during full science lead to lower requirements.

Table 31: JME comparative power budget with 20% system margin

	JRS		JEO	
	RPS	SA	RPS	SA
Power	10 W	10 W	9 W	69 W
AOCS	23 W	23 W	23 W	23 W
CMDS	29 W	29 W	29 W	29 W
Comm's	123 W	123 W	46 W	11 W
Thermal	0 W	54 W	0 W	28 W
Harness losses	6 W	8 W	3 W	5 W
Payload	6 W	6 W	0 W	28 W
Total with 20% system margin	236 W	304 W	133 W	231 W

The comparison of masses between RPS and solar arrays considers a hypothetic 80We RPS with a specific power between 6W/kg and 8W/kg. This was introduced as a requirement for a future RPS, and clearly needs considerable development effort. If current technology would have been assumed, the RPS system would have been much more demanding (more RPS sources required). Assuming the innovative electrical power capacity, 3 RPS are required for JRS and 2 for JEO. Propellant masses are calculated with ΔVs of Table 32 obtained for a launch in 2018. In the RPS option, 230m/s (without margin) are added on the interplanetary ΔV to account for GTO perigee and fly-by altitude increase.

Table 32: Comparative Delta-V budget (incl. 10% system margin)

	RPS	SA
Interplanetary DeltaV	2785 m/s	2485 m/s
DeltaV performed by JEO alone	1402 m/s	1402 m/s
DeltaV performed by JRS alone	305 m/s	305 m/s

Table 33 presents JRS and JEO masses including RPS of 10kg each (3 for JRS and 2 for JEO), while Table 34 presents the overall system mass budget, for different RPS specific power performances.

Table 33: Comparative JRS & JEO mass budget (RPS at 8 W/kg)

	JRS		JEO	
	RPS	SA	RPS	SA
Power	68 kg	110 kg	48 kg	106 kg
AOCS	8 kg	8 kg	8 kg	8 kg
Propulsion	135 kg	135 kg	40 kg	40 kg
CMDS	26 kg	26 kg	26 kg	26 kg
Comm's	41 kg	42 kg	25 kg	26 kg
Structure & Mechanisms	143 kg	145 kg	71 kg	73 kg
Thermal	10 kg	10 kg	7 kg	6 kg
Radiation shielding	8 kg	8 kg	27 kg	27 kg
Total	440 kg	484 kg	251 kg	311 kg

Table 34: JME system mass comparison (20 % system margin added)

	6 We/kg RPS	7 We/kg RPS	8 We/kg RPS	SA
JRS platform	543 kg	535 kg	528 kg	580 kg
incl. power system	*97 kg*	*89 kg*	*82 kg*	*132 kg*
JRS science instruments	18 kg			
JRS propellant	1788 kg	1760 kg	1740 kg	1694 kg
JRS wet mass	2349 kg	2313 kg	2285 kg	2292 kg
JEO platform mass	311 kg	305 kg	301 kg	373 kg
incl. power system	*67 kg*	*62 kg*	*58 kg*	*127 kg*
JEO science instruments	35 kg			
JEO propellant	220 kg	216 kg	214 kg	257 kg
JEO wet mass	565 kg	556 kg	549 kg	664 kg
Total launch mass (without adapter)	2914 kg	2869 kg	2835 kg	2956 kg
Launcher adapter	70 kg			
Launcher capacity	3000 kg			
Margin	0.5%	2.0%	3.2%	-0.9%

There is a 120kg mass gain on the total launch mass of the RPS option at 8W/kg with respect to the solar arrays option. The negative mass margin for the solar array option is due to the selected launch date. This shows that to obtain a significant benefit over solar power, the specific power must be in the order of 6-8 We/kg. Current RPS technology would result in mass budgets similar to the solar power option.

Table 35 shows the different possibilities with respect to the available routes using the Venus-Earth-Earth fly-by strategy (optimum in terms of Delta-V). Using RPS removes the 2016 launch option because of the time required for development. But it permits to launch in 2018 (optimistic assumption on the development time) and 2023, whereas it was not possible with the solar arrays option. However, no other launch window (2020, 2024 and 2029) can be opened irrespective of the RPS technology used.

Table 35: JME compared possible launch windows

Launch dates	2016	2018	2020	2023	2024	2026	2029	2031	2032
option with SA	OK	NOK	NOK	OK	NOK	NOK	NOK	OK	OK
option with 2nd gen MMRTG	N/A	OK	NOK	OK	NOK	OK	NOK	OK	OK
option with 8 W/kg performance	N/A	OK	NOK	OK	NOK	OK	NOK	OK	OK

RPS-6 SAFETY IMPACT

A Safety assurance programme shall cover all aspects of personnel, environment, flight hardware and facilities safety. In any case, emphasis must be placed on hazards control.

A number of regulations have also to be taken into account related to:
- liabilities and responsibilities of the countries and of the suppliers;
- transportation;
- safety management, including pre-launch and launch specific safety requirements;
- Apogee raising and Earth fly-by sequences.

A safety engineering activity specific to the use of RPS has therefore to be carried out starting from the first stage of the feasibility evaluation tasks to the final test and integration activities on the launcher site. This activity is likely to represent an important part of the engineering work during the program.

It is therefore proposed to perform several analyses covering:
- safety;
- transportation and storage;
- emergency plans;
- environmental impact

All those activities are to be defined in a dedicated document called “safety plan” to be approved by the customer organization during the preliminary development phases. It is to be highlighted that all safety related tasks need to be conducted in parallel with all development tasks and supported by a dedicated organization integrated in the prime project team (safety coordinator and related supports).

US past and present experiences of RTG on-board satellites permit to consider their methodology as very mature, and should therefore be a good basis for the definition of European (or worldwide) rules regarding the use of nuclear materials in space. Description of US methodology regarding safety can be found in [29].

RPS-7 IMPACT ON THE DEVELOPMENT PLAN

RPS-7.1 Drivers

Due to the presence of RPS, the development of the JME space segment is additionally driven by the following requirements and considerations:

- A specific nuclear safety approach must be implemented, which is new in Europe;
- The use of RPS is tied to political events. A good dialog with European citizens is therefore recommended to convince that all actions are undertaken to keep risks under an acceptable level;
- RPS technology development schedule is very uncertain: first the best technology must be identified. Then it must be understood if the cost will require international cooperation and finally if Europe will build its own fuel production capacity

RPS-7.2 RPS technology development

Assuming that Europe is willing to develop its own radio-isotope generator, the following schedule can be drawn [28] [30] [31]. It must be noted that the timeline is very optimistic and assumes that the development has started in 2006. In view of the programmatic and technical complexity it is very likely that this schedule needs to be extended by several years.

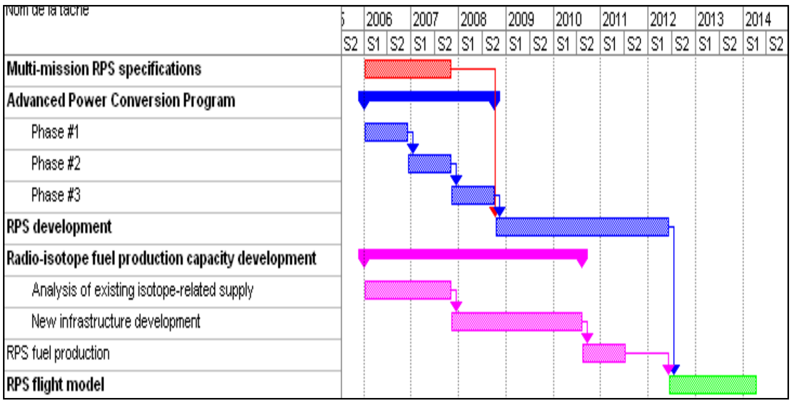


Figure 61: RPS technology development activity

RPS-7.3 Overall program development

The development plan remains the same as in the previous study expect for the technology development activity duration. The same comments as for the previous figure are valid here.

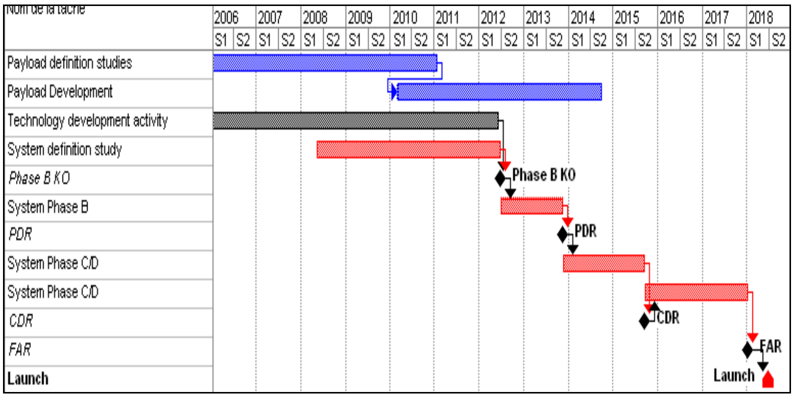


Figure 62: JME development plan

RPS-8 CONCLUSION

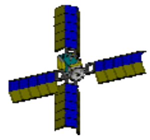
RPS-8.1 RPS baseline and solar power system baseline comparison

The RPS main assets are the following:

- A mass gain can be achieved, provided that the RPS shows a performance greater than 6-7 W/kg to provide a significant mass gain as compared to solar arrays
- With a theoretical RPS specific power of 8 W/kg, a target value that often appears in strategic documents, a mass gain of 200kg can be expected on the launch mass
- no thermal problem is foreseen when getting closer to the Sun for the Venus fly-by
- RPS provides the same power in both lit and eclipse conditions. Therefore the Europa orbit orientation with respect to Sun will only depend on the instruments
- RPS is insensitive to radiation contrarily to solar cells. Therefore, the power system design is simplified, and the science mission doesn't need to be downgraded during its processing
- The RPS technology may also be used by other missions beyond the Jovian System. This is not the case of solar panels whose efficiency will be too low beyond the Jovian System

However, the following must be born in mind:

- Development, procurement, cost and, to a lesser degree, AIV risk are major issues
- A conservative approach was used for this study: 4.7 W/kg EOL (goal value for 2nd generation MMRTG). In this case, the mass gain obtained by suing the RPS, as compared to the solar arrays, is almost cancelled out by the additional DV imposed by the Earth fly-by altitude increase
- Moreover, the active thermal control required under Soyuz fairing is likely to add mass, and thus to reduce the total launch mass capacity



The solar array main assets are the following:

- Use of solar cells is politically/socially accepted
- Solar power systems are based on a harmless technology that doesn't require a specific and complex safety plan
- No damaging effects on the spacecraft due to radiation or excess heat
- Limited development time required, experience is available in Europe: can be ready on time
- Once the correct cell is developed, no significant procurement problems are expected
- Cost is considerably lower than radioactive power sources
- Straightforward AIV activities, much experience available
- No exotic limitations regarding safety, including potential Earth gravity assists
- Solar power can also be used in association with SEP, which could offer the means to save propellant mass, and permits to have yearly launch windows

Therefore, Solar Power is the preferred energy source. However, if the required solar cells cannot be developed, RPS alternatives must most probably be used.

RPS-8.2 RPS enabling technology in Europe

RPS is a mandatory technology for future scientific missions beyond Jupiter.
The most efficient approach for developing RPS is to design a multi-mission RPS, in order to share the costs between several missions, to avoid non-recurrent costs, and to optimize the development times.

The best future RPS technology is still under investigation:

- Candidates are RTG, SRG, RTPV, BRG and AMTEC;
- The objective of the best specific power (W/kg) is the mass gain;
- The objective of the best energy conversion efficiency is for safety and cost. Depending on the technology, the JME Pu-238 requirement varies between 3kg and 12kg, knowing that the cost will not be less than 1.5 M€/kg.
- There are other technical requirements to be considered such as lifetime (~100 000 hours), reliability, electro-magnetic and micro-vibration cleanliness, radiation hard equipments (in the case of the Jovian System)

It appears today that only the USA show real funding for a research on RPS. However, European skill exists in relevant areas such as Stirling energy conversion and in the nuclear industry.

RPS-8.3 The safety issue

The safety needs to be formally approached. It is not a challenge and can be managed. But it is mandatory due to the high toxicity of Plutonium. It is recommended to have a unique European applicable document on nuclear safety. Debates on safety issues can be difficult and time demanding. It is therefore very important to decouple this debate of the mission schedule.

Rules integrated in the safety plan need to be approved by everyone concerned from the beginning of the programme to avoid any impact on the development schedule, which is critical due to the launch window.

The safety plan realisation will also be time demanding due to the international implications, the number of regulations and the numbers of documents that are required.

RPS-8.4 RPS for European scientific space missions

It is clear that the use of RPS has a lot of programmatic implications. Contrary to technical issues, such impact is hard to quantify in terms of time. In particular, there are a number of open questions that Europe must address regarding RPS technology development. These questions are not easy to answer, but must be answered to define a clear strategy. These questions are:

- Does Europe want full autonomy on this technology, or should Europe to aim for international cooperation with USA or Russia, or worldwide?
- What is the real cost of RPS development and is Europe ready for this financial effort?
- Is there any potential conflict with national defence policies regarding nuclear fuel?
- What is the acceptability level by European public?

It can be concluded that the development of the RPS technology is mandatory for scientific missions beyond Jupiter, and, if for some reason the solar power option would not be feasible for Jupiter, also for missions to the Jovian System itself.

If it is deemed necessary, European RPS development will be a long-term development and if required for JME, it should be started as soon as possible. Even if this is started now, it is unlikely to be available for missions to be launched around 2020.

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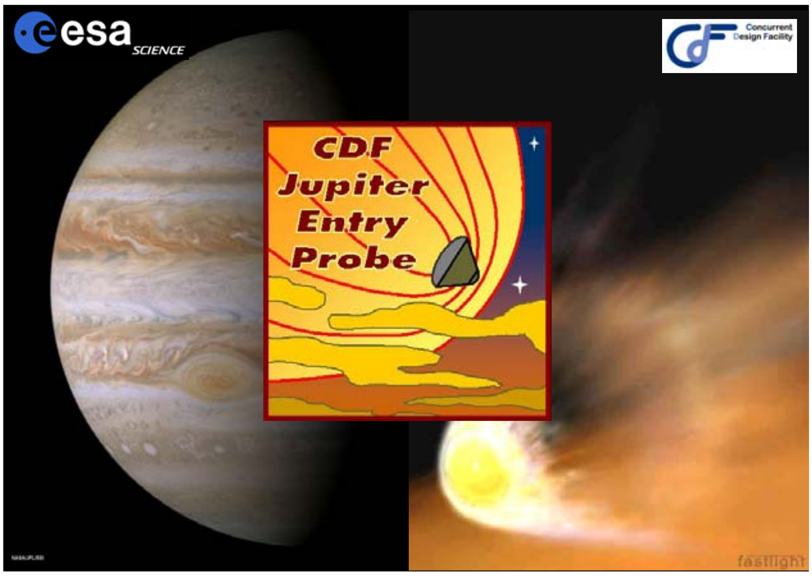
THE JOVIAN ENTRY PROBE – PROBING JUPITER'S ATMOSPHERE



JEP-1 INTRODUCTION

This section focuses on the Jovian Entry Probe (JEP) study, which was performed by ESA's Concurrent Design Facility (CDF) and is part of the Jovian Technology Reference Studies. This particular study aimed to understand the requirements for a minimum resource probe capable of entering the Jovian atmosphere up to a pressure level of 100 bar.

It is important to note that this study was not performed as an add-on to the JME study: it was intended as a study looking into the feasibility and implications of the probe. The reference trajectories and spacecraft masses do not reflect the JME numbers in the previous chapters, as the study was performed prior to the latest JME update.

The following requirements and constraints apply to the study:

- Carry the probe to Jupiter and release it at the correct time
- Perform entry and descent into the Jovian atmosphere at near equatorial latitude (with an option of non-equatorial descent up to -30deg/+30 deg, if possible)
- Measure atmospheric properties in-situ down to a depth corresponding to 100 bar atmospheric pressure using a given strawman payload
- Transmit the data in real time to the accompanying orbiter
- Achieve a final orbit for magnetospheric measurements with the orbiter
- Achieve multi-probe mission, if mass allows
- Use of highly integrated payload: 12 kg; 30 W; 5 l; 353 bps

- Launch vehicle: Soyuz Fregat 2-1b from Kourou
- Preferred launch dates: 2016 or 2023
- Avoidance of Jovian ring when defining probe approach, while not exceeding maximum allowable distance during communications
- Design shall be compliant with Beagle 2 Enquiry Board recommendations and Huygens Lessons Learned
- Maximum heat flux during entry: 500 MW/m^2 (assumed as maximum capability for present Thermal Protection System (TPS) technology)

JEP-2 MISSION DESIGN DRIVERS

This mission concept is driven by four main challenges: The Jovian atmosphere, the high entry velocity, launch mass restrictions and communications.

The main issue with the atmosphere is related to the uncertainties regarding the aerothermal phenomena. These uncertainties strongly complicate the design of the heat shield, since they impose significant margins to be added to the design, to compensate for these uncertainties. As a consequence, this leads to a likely over-dimensioned thermal protection system: depending on the entry latitude the resulting TPS mass fraction is in the order of 50% to 70%.

The entry velocity cannot be reduced below ~47 km/s, as explained in section JEP-3 . At these velocities and due to the previously mentioned limitations, the aerodynamic phenomena in the atmosphere cannot be properly computed, leading to uncertainties in the calculation of the heat fluxes. Further, the very high thermodynamic fluxes are at the limit of present TPS technology capabilities. The very high deceleration loads (in excess of 1700 m/s^2) additionally require dedicated qualification of the probe's components.

The relatively modest launcher provides the upper limit for the launch mass, while the fulfilment of the mission requirements provides the lower limit. This in turn poses a limit to the maximum TPS mass and therefore to the maximum entry velocity.

The high temperature and pressures in the atmosphere at lower altitudes further complicate the design of the entry probe, since the design needs to offer adequate protection against these conditions.

The strong attenuation of radio signals by the atmosphere below the 20 bar level impose stringent design requirements for the communication systems on both the probe and the orbiter. Furthermore, the trajectory of the carrier spacecraft will have to allow for a continuous communication with the probe during the deployment and relay phase.

JEP-3 MISSION ARCHITECTURE

The mission composite (Orbiter + Entry Probe) shall be launched by Soyuz-Fregat 2-1b into a highly elliptic orbit (HEO). The spacecraft is then inserted into a hyperbolic Jupiter transfer orbit by its own propulsion system with a two-burn sequence. The launcher performance into the optimal HEO is 2 346 kg including adapter.

The Jupiter transfer trajectory is of the VEEGA type; including a Venus swing-by and two Earth swing-by's aiming at Jupiter impact for release of the entry probe. No mid-course manoeuvre is required except for navigation corrections.

Two cases have been considered: single probe or two probes onboard of the same orbiter. During cruise the probe is attached to the orbiter spacecraft and uses the orbiter's power supply to perform periodic instrument checkout and possibly software updates.

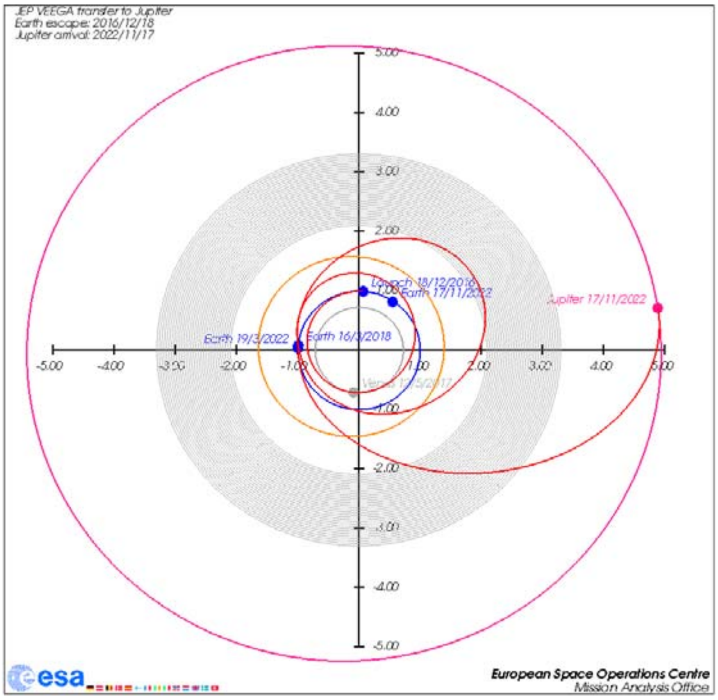


Figure 63: The VEEGA Transfer Trajectory

Sufficiently before Jupiter arrival, the Orbiter deploys the entry probe (in short sequence, should two probes be considered) and performs an Orbit Deflection Manoeuvre (ODM) to get into a safe non-entry trajectory.
The time of probe release compared to the entry time sizes the delta-V cost of the ODM and the error on the Flight Path Angle (FPA) at entry which is constrained by probe TPS design. The selected release time is 90 days before entry with a delta-V cost of 89 m/s and a FPA error at entry of less than 1 deg.

During the coast phase, the probe is uncontrolled and unguided. It uses its own power system to perform communications with the Orbiter, and timer switches are used to activate automatic sequences.

While the probe coasts to its entry point, the Orbiter performs a Ganymede swing-by to reduce its incoming velocity and therefore reduce the delta-V cost of the Jupiter Orbit Insertion (JOI). The insertion orbit is the orbit from which the relay with the probe(s) is performed during their entry and descent. This is a 4x200 Jovian Radii (R_J) equatorial orbit around Jupiter (for near equatorial entry and descent). The perijove radius is a compromise between distance for probe relay (the closer, the lower the required power) and radiation protection (the closer, the higher the dose). The apogee corresponds to the required final orbit of 15x200 R_J. The JOI manoeuvre takes place 1 hour before perijove arrival, requires 570 m/s and its duration is about 0.5 hours.

The start of the probe entry phase is defined as the point where the probe reaches 450 km altitude above 1 bar (the 1 bar level is used as a reference zero level for altitude measurement). During this phase the probe relays flight instrumentation data (used for trajectory reconstruction) to the Orbiter with the exception of the period of blackout caused by the plasma sheath around the probe.
Due to Jupiter's massive gravity field, the spacecraft will accelerate considerably as it approaches perijove. The consequence for the entry probe is that the inertial velocity at entry will amount to around 60 km/s with only a weak dependency on the hyperbolic entry velocity. As Jupiter's rotation period is less than 10 hours, the equatorial atmospheric rotation speed is almost 12.6 km/s. Therefore, the actual atmospheric entry velocity depends strongly on the entry location. For a prograde, near-equatorial entry, the relative entry velocity is thus 47 km/s.

The science data relay phase occurs after the front heat shield and back cover have been released and the main parachute has been deployed. At this point all instruments will take measurements from the Jovian atmosphere and send them back to the Orbiter. The relay phase ends after the one hour communications window when the probe has reached 100 bar depth in the Jovian atmosphere. At this point the probe's mission is complete.

During the relay, the Orbiter needs a relatively slow, constant-rate slew manoeuvre (rate ca. 17 deg per hour), to keep its high-gain antenna trained upon the current probe location.
It is noted here that Direct-To-Earth communication from the probe is only possible during the early phases of entry. This is due to the low Earth elevation with respect to Jupiter's local horizon in the analysed 2022 arrival case; the very high rotation speed of Jupiter only allows for a short visibility time of the probe.

After the relay phase, the Orbiter will reach its final orbit for magnetospheric measurements, which has a line of apsides aligned with the sun direction. This is achieved by the combination of a propulsive manoeuvre of 500 m/s at apojove to increase the altitude of perijove and a Jovian satellite tour (a sequence of five Ganymede swing-by's) to rotate the line of apsides as needed.

Table 36: Delta-V budget for baseline and option 1

Manoeuvre	1 probe Mission Delta-V (m/s)	2 probe Mission Delta-V (m/s)
Satellite tour/Apojove raise	30	30
Perijove raise (PRM)	500	500
Jupiter orbit insertion (JOI) 1 hr before perijove	570	570
Orbit deflection man. 70d before entry	N/A	120
Hyperbolic probe release	N/A	0
ODM 90d before entry	89	42
Probe release from hyperbolic	0	0
Mid-course manoeuvre	0	0
VEEGA	30	30
Escape from HEO	626	626
Inclination change	82	82
GTO to HEO	692	692
Total incl. gravity loss	2668	2740
Final total incl. margin	2801	2878

The mission Delta-V budget is shown in Table 36 for a single probe and a two-probe mission. The table is based on the 2016 launch window, which is the worst case between the selected target launch dates.

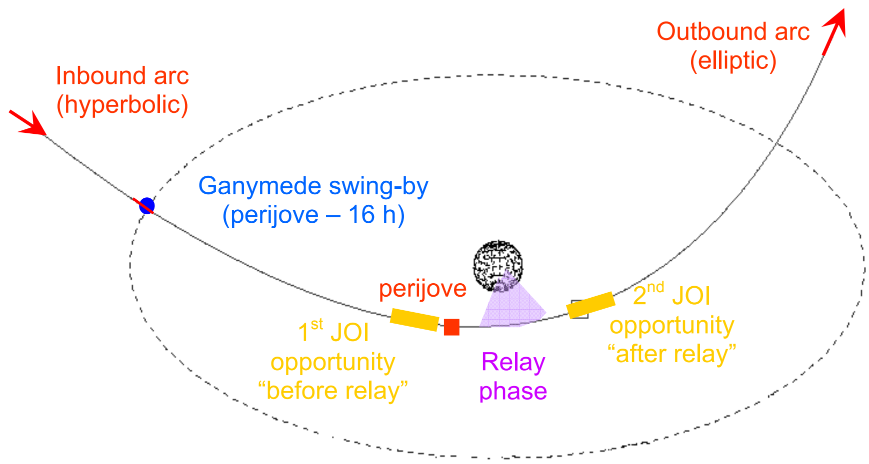


Figure 64: Swing-by augmented JOI

JEP-4 ALTERNATIVE OPTIONS

This study has assessed multiple scenarios, including the option of carrying two probes on one spacecraft, as well as off-equatorial entry and release from orbit, to see implications of these scenarios. The main results are shortly summarised in this chapter.

JEP-4.1 Two probes

A two probe mission would be preferable to enable different entry points and more data collection, and to allow for redundancy. Nonetheless, due to launcher mass constraints, the case of two probes is only considered feasible if the maximum pressure depth is reduced to 40 bar for both probes. In this case the two probes are deployed into approach orbits of different inclinations, leading to a difference in the entry and descent locations, the first probe aiming at a latitude of 3.6 deg N, the second at a latitude of 6.8 deg S. The relative entry velocity for both probes is slightly higher than 47 km/s.

JEP-4.2 Off-equatorial entry

The preferred probe entry latitude is non-equatorial between 30 deg N and 30 deg S. Entry at high latitude would require an approach from an inclined trajectory and an increase in delta-V to retarget the Orbiter for an equatorial insertion.
For the high-latitude probe, the inclination can be fairly low, which will limit the rise in relative entry velocity but would require a steeper entry. Or, the inclination can be larger, in which case a larger increase in relative velocity is incurred, but the entry angle could be kept relatively shallow.
This effect is due to the fact that the perijoves of the arrival hyperbolae are all close to equator and that shallower angles can only be achieved close to the perijove.

As an example, for a descent latitude of 15 deg South, the lowest inclination possible is 25 deg, leading to an entry FPA of -16 deg and an entry velocity around 49 km/s, while the larger inclination leads to –10 deg FPA and an entry velocity of about 50 km/s. In any case, heat fluxes occur in excess of 500 MW/m^2, exceeding available TPS capabilities. In addition, the high-latitude probe would require the communication relay to be conducted at an oblique angle with respect to the equatorial Orbiter. For these reasons, non-equatorial entry has been considered as non-affordable.

JEP-4.3 Release from orbit

Release from capture orbit requires a higher Delta-V and gives an allowable probe mass of approximately 230 kg while it doesn't reduce the probe's entry speed enough to enable a lighter probe (46-47 km/s for capture vs. 47.4 km/s for hyperbolic). Furthermore, the communications between the probe and the orbiter become much harder, as there cannot be a continuous one hour communication window, due to the eclipses caused by Jupiter. As there were no other clear advantages for the release from capture orbit, this option was discarded.

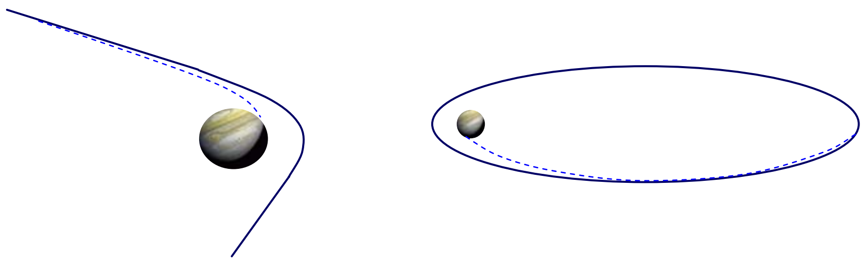


Figure 65: Hyperbolic release vs. from orbit

JEP-5 AEROTHERMODYNAMICS

The assumed probe shape is similar to the Galileo probe design, containing a front shield with a half cone angle of 45°, as shown in Figure 66:

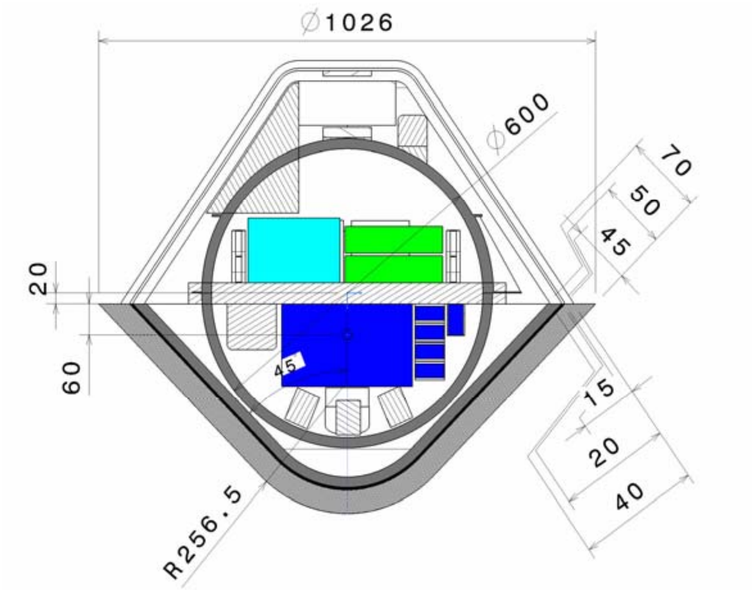


Figure 66: Probe geometry

The equatorial entry parameters are the following:

- Entry Altitude: 450 km
- Entry Velocity: 47.4 km/s
- Entry Angle: -7.5°

JEP-5.1 Equatorial entry

Two entry mass cases, 310 kg and 280 kg, have been studied depending on the final altitude respectively pressure level (100 bar and 40 bar). Figure 67 presents the entry and descent trajectories.

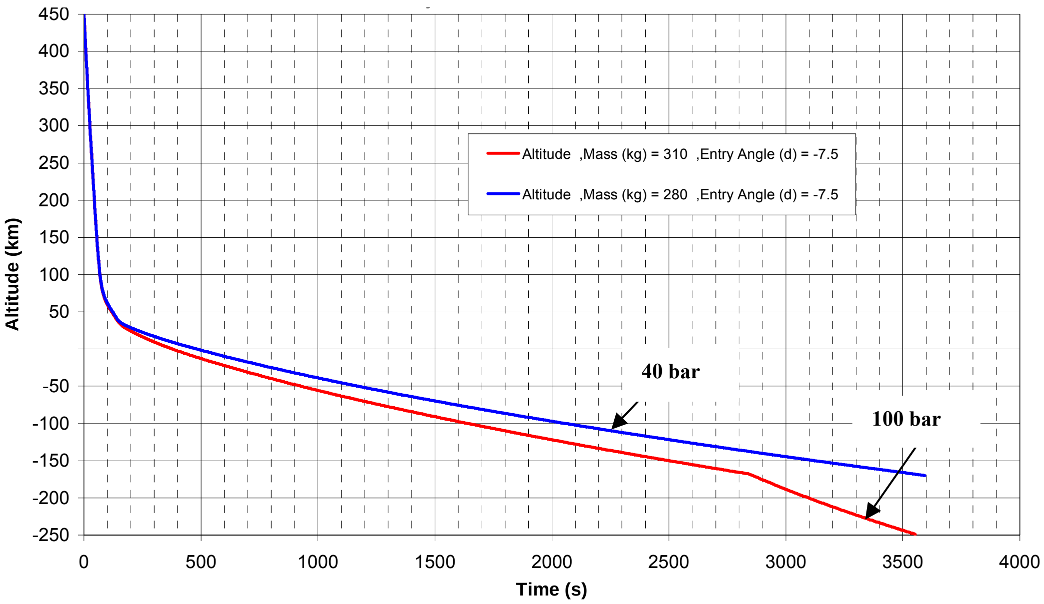


Figure 67: Altitude vs. Time for equatorial entry

Relatively similar acceleration profiles are obtained for both cases with a peak around 1 700 m/s^2 at 69 s after the entry point (Figure 68). The smaller peaks correspond to the pilot chute deployment, release of the pilot / back cover / deployment of the main chute and the release of the front heat-shield.

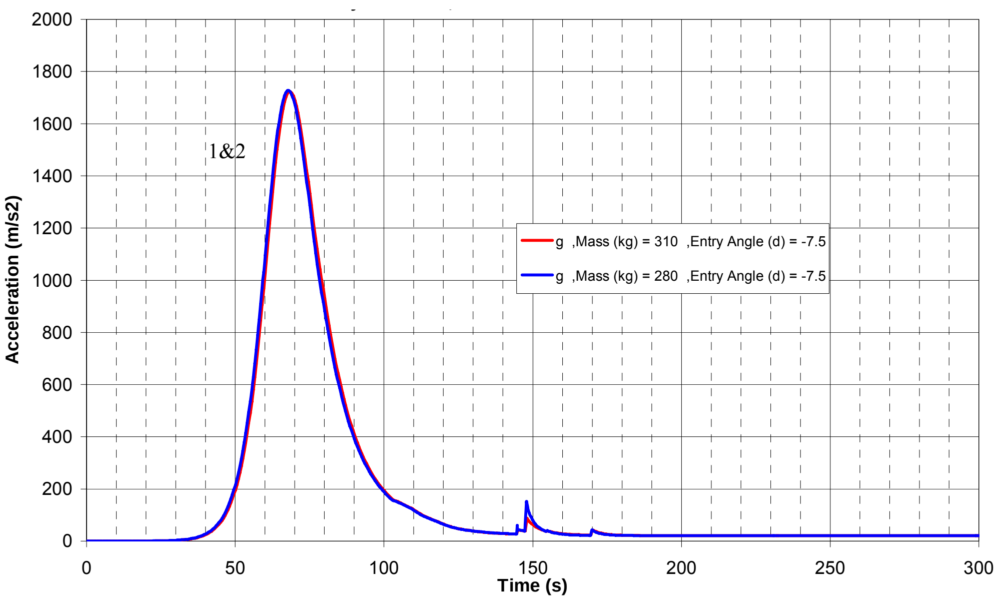


Figure 68: Acceleration vs. Time for equatorial entry

The radiative heat fluxes at the stagnation point in both options are presented in Figure 69.

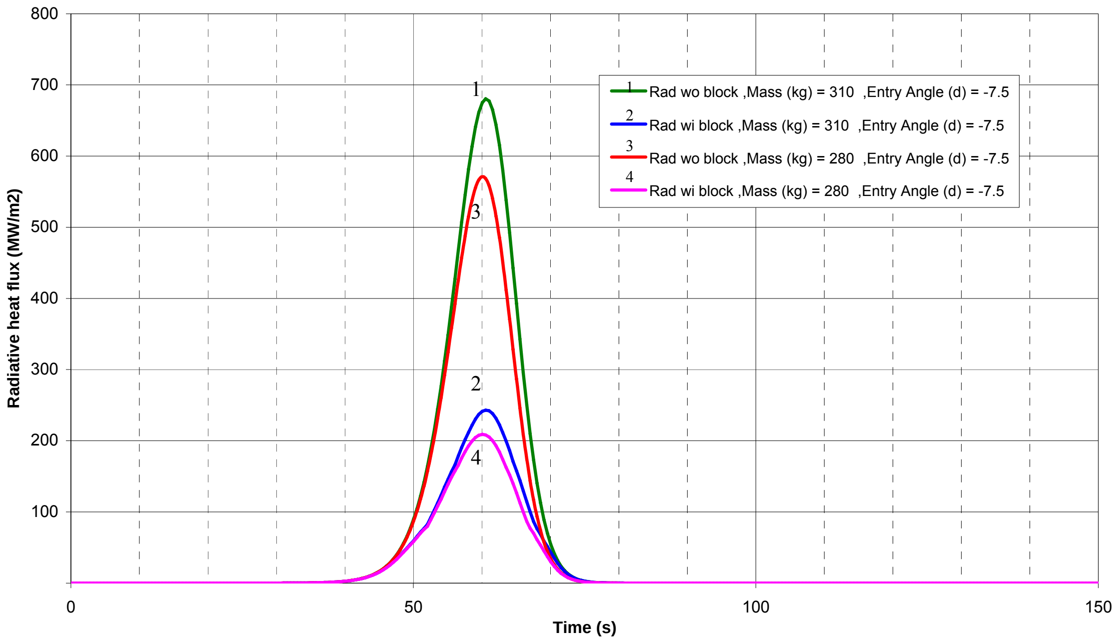


Figure 69: Heat Fluxes vs. Time for an equatorial entry

The heat flux distribution along the front shield, at the stagnation point, mid-cone, edge and base point is presented in Figure 70 for the 100 bar option.

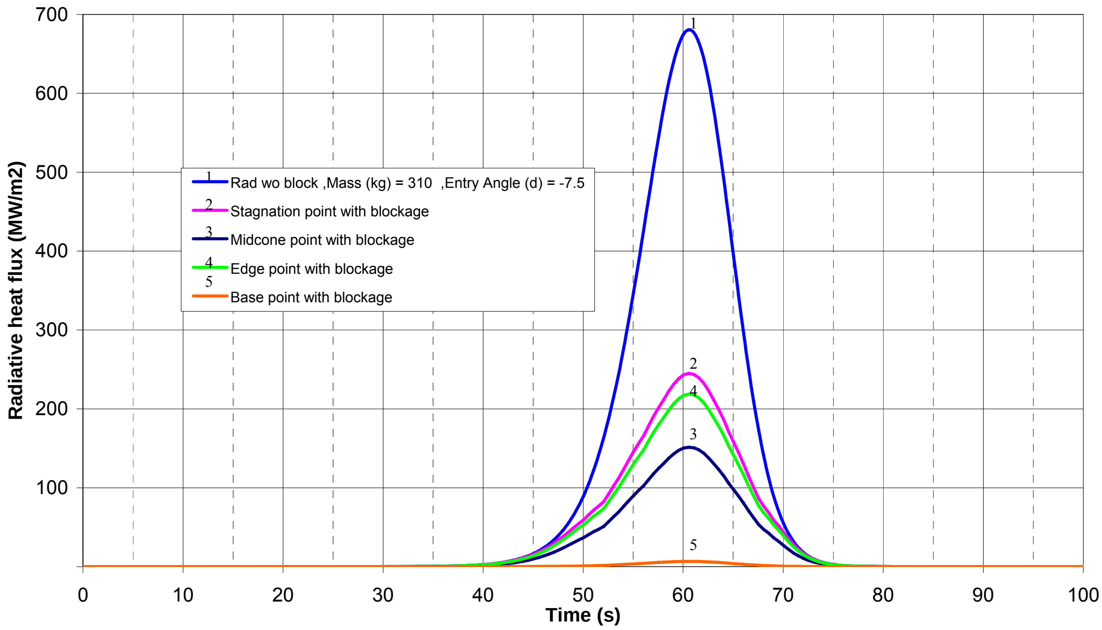


Figure 70: Heat Fluxes vs. Time over the front shield surface for a Final pressure of 100 bar

Only 3-dof analyses were performed, therefore the probe stability during entry and descent could not be confirmed. The distance between the CoG location and the back cover/front shield interface is -38.7 mm which is about 3.8% of the base diameter and therefore lower than 4.5%, which was the requirement for the Galileo probe. This point would need to be addressed in further detail.

JEP-5.2 Non-equatorial entry

For a non-equatorial entry, the entry mass was assumed to be 500 kg in all cases. The used dimensions of the probe are 1.30m for the base diameter with a nose radius of 0.65 m. The radiative heat fluxes are presented in Figure 71. Due to the very high level of the radiative heat fluxes (>1 GW/m^2 even with blockage), the non-equatorial entry mission is beyond present technology capabilities.

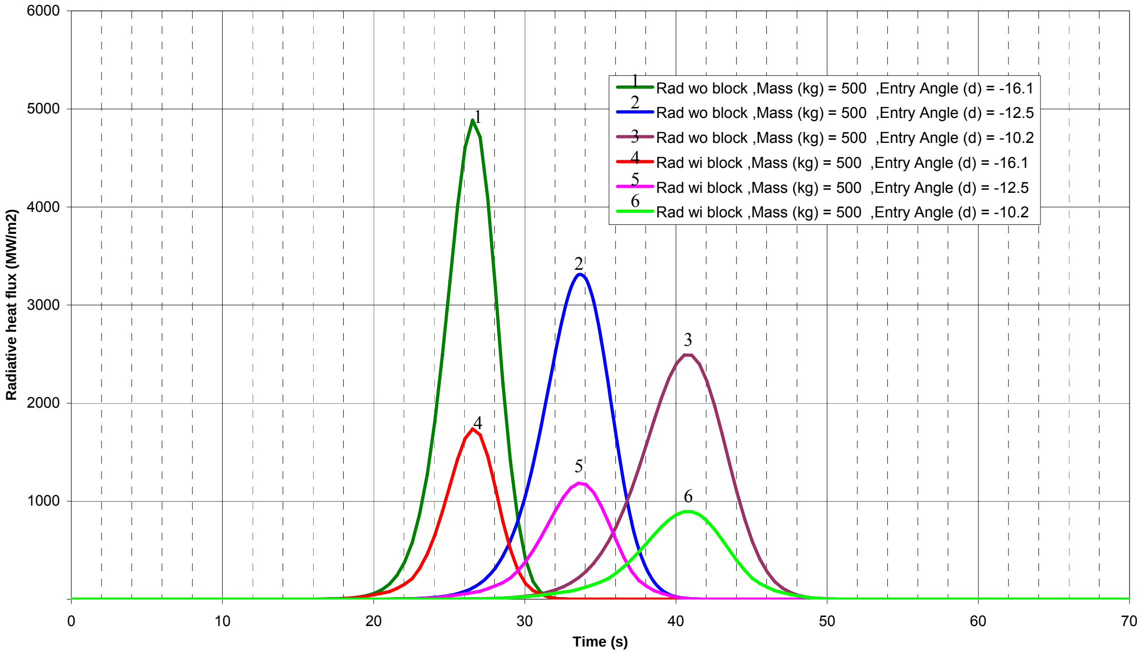


Figure 71:Heat Fluxes vs. Time for non equatorial entry

JEP-6 THERMAL PROTECTION SYSTEM

One of the major feasibility drivers of the overall mission is the design of the thermal protection system and the availability of a suitable material capable to withstand the very high radiative and convective heat fluxes. A significant effort of the study was dedicated to the screening of potential heat shield concepts. As a result, a Galileo-like shield based on Carbon-Phenolic ablator still appears as the most promising solution.

The material considered as reference in this study is part of a family whose characteristics are close to the one used for the Galileo mission. The present availability of the material could not be confirmed. In any case, a dedicated development would be required for Europe.

Analysis has shown that if the ablator is applied using the SEPCORE concept (Figure 72), a mass reduction of about 25% can be achieved compared to a conventional ablator with a cold structure concept. This is mainly due to the fact that the ablator is mounted on a hot structure, which is insulated against the inner compartment using lightweight insulation, possibly fibres. Considerable mass savings are obtained due to reduced ablator thickness and the use of a more efficient insulation.

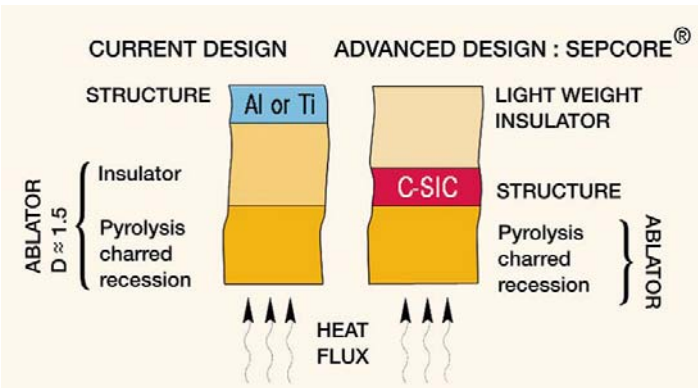


Figure 72: Classical vs. SEPCORE TPS

Alternative options to be considered in later project phases are heat shields based on either Carbon-Carbon or Carbon-SiC ablators.
The TPS design is shown in the following figure:

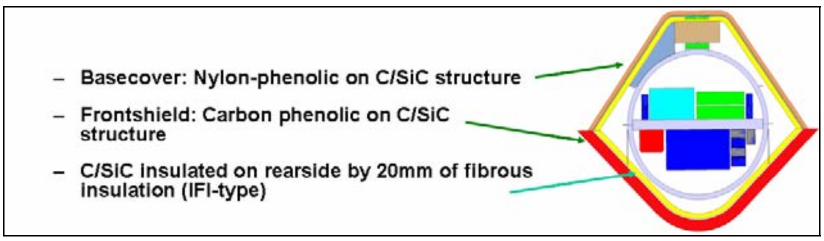


Figure 73: TPS schematic

Due to the uncertainties on the aerothermodynamic fluxes and loads as well as the TPS material characteristics in such entry environment, a robust margin philosophy has been applied (> 40% overall).

JEP-7 DESCENT SYSTEM

In the nominal case, the end of mission will occur when the probe reaches a depth corresponding to 100 bar. Due to communications constraints this will have to be achieved in less than one hour, otherwise measurements performed at low altitudes cannot be relayed back to the Orbiter. Therefore the requirement for the parachute system is to provide a flight time of around one hour to the final altitude. In addition, the parachute shall safely separate the probe from the heat shield by increasing the area of the separated elements, obtaining a ballistic coefficient that is sufficiently different (factor 2).
A minimum parachute designed to provide the above separation leads to a flight time in excess of 1 hour to achieve 100 bar altitude. Therefore, the parachute system needs to include a release mechanism so that the probe can accelerate in the last part of the descent (see Figure 74)

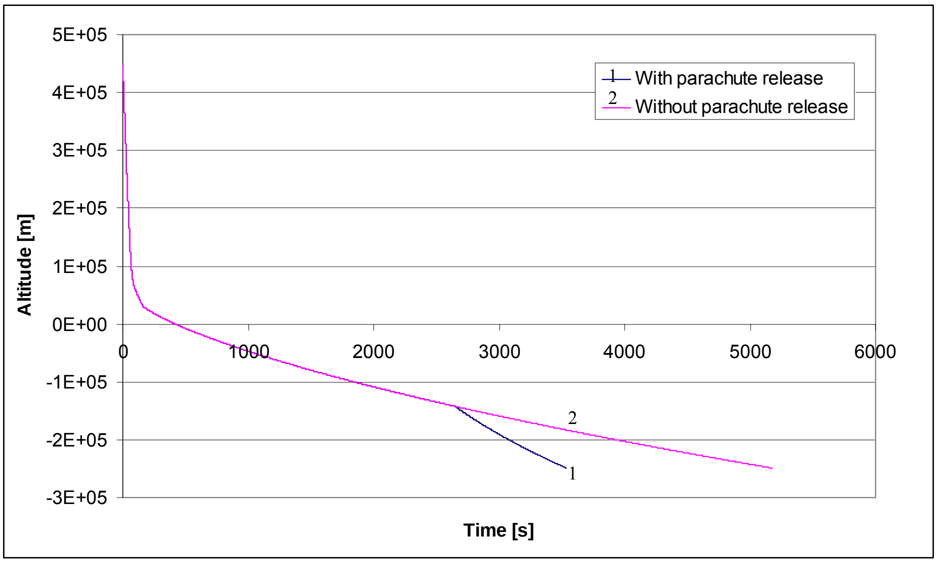


Figure 74: The descent trajectory

The descent system consists of a pilot parachute attached to the back cover with a diameter of 1.47m and a C_d of 0.52. The pilot is deployed at Mach 1.1 by a mortar, triggered by an accelerometer, g switches and a timer as backup. A main chute with a diameter of 2.28 m is deployed by the back cover once it is separated.

The descent module together with the front shield will continue the descent under the main chute for another 20 sec, allowing for stabilisation before a timer triggers the front shield release. The descent module will then continue its descent under the main parachute with the scientific payload operational. The main parachute is finally released after 47 min.

Conical ribbon technology was selected for the parachute due to its superior performances at high dynamic pressures (opening of the parachute occurs at q=12 kPa) and structural integrity.

Furthermore, this type of parachute fulfils the stability requirement, although a small penalty has to be paid in terms of drag coefficient.
Dacron is proposed as the material for the canopy construction and Kevlar for the lines. At the time of release, the atmosphere temperature will still be sufficiently below the material performance limits.

JEP-8 COMMUNICATIONS

The following communication architecture is foreseen:

- *Coast phase:* data transmission to the orbiter occurs via a back cover antenna. Carrier recovery and Doppler tracking can be achieved by VLBI (Very Large Base Interferometry). To reduce the power consumption, a 3 hours total transmission time is assumed during the whole cruise phase at a data rate of 8 Kbps
- *Entry phase:* during this phase (~3 minutes) no transmission will be possible (black-out) due to attenuation by the plasma cloud
- *Parachute deployment and descent till 0.2h from entry:* after the back cover separation and parachute deployment, the Descent Module (DM) helix antenna will start to transmit. The telemetry signal will be received by the orbiter and VLBI until 0.2h after the entry. After that the Earth will be below the 'Jovian horizon' and VLBI can not detect the probe's carrier signal anymore
- *Descent, from 0.2h after the entry till 100 bar pressure altitude:* data transmission between probe and orbiter will take place via the DM patch array antenna. A minimum net data rate of 353 bps is required

A variable power system is foreseen to cope with the very strong atmospheric attenuation (up to ~24 dB at 100 bar). The maximum power consumption of one link is 225 W. Link redundancy (as in Galileo and Huygens), would imply unacceptable power consumption. Therefore, only cold redundancy has been assumed.

The frequency band selection is a trade-off between the conflicting factors of atmospheric attenuation and synchrotron radiation of Jupiter. The high atmospheric absorption is due to ammonia, water, sulphide and phosphine in the Jovian atmosphere (polar molecules). The synchrotron radiation originates in the Jovian magnetic field and depends on the geometry of the orbiter antenna orientation.
Frequencies below S-band (2GHz) need to be considered to limit attenuation. Therefore, the reuse of the Huygens frequency band (S-band) is not possible. On the other side, below 1.3GHz, the synchrotron radiation is expected to increase, overcoming the positive effect on signal attenuation. As a result of this trade-off, a 1.3GHz system (L-Band) has been selected for this mission.

All considered mission cases give a positive margin for a minimum data rate of 370 bps (353 bps + 5% margin), a maximum power of 100W and an antenna size on the Orbiter of 4m for the 100 bar cases. This antenna size will cause considerable accommodation problems for the carrier and needs to be properly understood if this concept is selected for further study. Because of the high attenuation, deeper altitudes into Jovian atmosphere would imply a significant increase of resources to maintain the link budget margins and is therefore considered unfeasible with the selected configuration.

JEP-9 DESCENT MODULE CONFIGURATION

A trade-off has been performed between several different structural concepts for the Descent Module (DM). In the end a spherical titanium sealed vessel was selected, with an internal pressure of 1 bar. The option of a fully internally pressurised DM (100 bar) has been rejected due to expected leakage during long cruise to Jupiter, the structural loads and the handling risks.

Under uniform external pressure, a thin-walled sphere buckles at a fraction of the pressure that would cause the same vessel to fail under uniform internal pressure. Therefore, the vessel has been stiffened with circular ring frames. These frames are also used to support the equipment shelf and serve as an attachment for the interface brackets of the DM to the front shield.
The DM features a single internal equipment shelf that hosts the entire internal equipment. In particular, the vessel contains the science payload, the CMDU, PCDU, Comms transponders and amplifiers, batteries as well as the L-Band patch and helix antenna on the outside. A volume reduction exercise of this equipment has been performed to decrease the required dimensions of the DM.

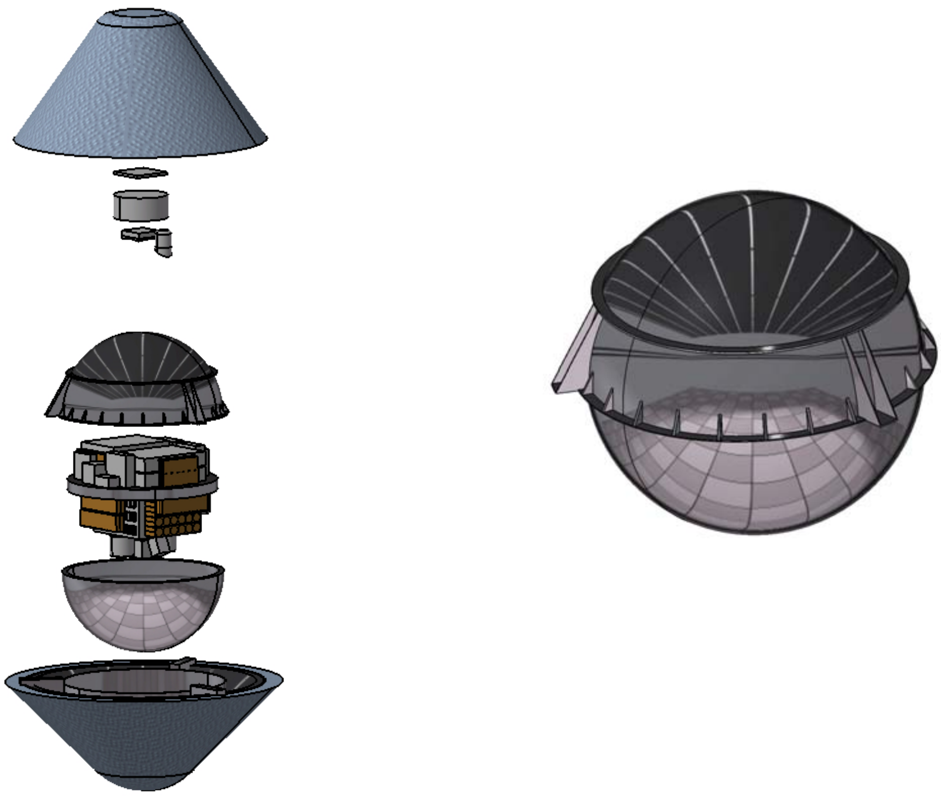


Figure 75: Exploded view of the probe+the DM

For stability during the descent of the DM into the Jovian atmosphere, vanes are added on the DM. Inlets and windows are added for the strawman payload, as it is needed for its operation.
As the overall dimensions and mass of the probe result from the dimensions of the DM (which is sized by the equipment volume), the probe mass cannot be reduced below a certain threshold unless high electronics integration is pursued, something that should be kept in mind for further studies.

JEP-10 BUDGETS

This chapter provides an overview of the main system budgets for the entry probe.

JEP-10.1 100 bar probe mass budget

The minimum configuration mass budget is shown in the following table:

Table 37: 100 bar probe mass budget (mass in kg)

Subsystem	Mass (kg)
Structure	30.7
Thermal control	145.7
Mechanisms	9.3
Comms	6.8
Data handling	11.5
GNC	1.6
Power	18.1
Harness	10.1
Instruments	10.4
DLS	6.3
Total dry mass	250.5
20% system margin	50.1
Total mass with margin	**300.6**

JEP-10.2 Mission options comparison

This section shows a comparison of several options that have been investigated during this study.

The "larger P/L" option concerns a single probe with increased payload mass as a result of:

- 100% increase of the P/L mass
- 100% increase of the P/L volume
- 50% increase of the P/L power
- 50% increase of the P/L data rate

This option was analysed to assess the sensitivity of the probe design to changes in the payload.

Note, that all dry masses include 20% system margin.
The following table shows the available launch mass margin for the analysed mission options:

Table 38: Mission architecture comparison with launch margin

	Baseline (100 bar)	**Two 40-bar probes**	**Larger P/L**	**Two S/C**	**From orbit**
Total Delta-V with margin [m/s]	2801	2878	2801	2641	3002
Orbiter 1 dry [kg]	700	700	700	700	700
Orbiter 2 dry [kg]	--	--	--	700	--
Tot prop mass [kg]	1236	1409	1268	1903	1644
Probe 1 dry [kg]	300	268	350	--	300
Probe 2 dry [kg]	--	268	--	--	--
Total launch mass [kg]	2236	2646	2318	3304	2944
Launch margin SF 2-1b	24%	10%	21%	-12%	2%

JEP-10.3 Power budget

The following table shows the power requirements of the baseline probe (100 bar) as a function of the different phases: passive coast (90 days prior to entry), check-out during coast (3 hours total) and the entry and descent phase (1.25 hours, including check-out).

The relevant power architecture is based on two different types of primary battery, $LiSOCl_2$ for the timers during coasting and $LiSO_2$ for the PCDU. This approach gives the minimum system mass.

Table 39: Power budget per phase

Coast Phase	*90 days*		
DHS		0.3 W	
GNC		0 W	
Comms		0 W	
		648 Wh	
Check out during coast	*3h total*		
DHS		22 W	
Payload		5 W	
Comms		50 W	
		231 Wh	
Entry + descent	*1h 15*	**Average**	**Max**
Comms		104 W	225 W
DHS		22 W	22 W
Payload		28 W	29 W
GNC		22 W	22 W
		220 Wh	298 W
Total Wh (30% margin)		1429 Wh	
Total Pmax (30% margin)			388 W

This overview shows that a significant gain can be obtained if the power consumption during cruise can be reduced. This consumption is mainly due to the timers (triple redundant). By integrating this timer function on a chip and by drawing the power from the main batteries, while maintaining the required redundancy (this is currently being assessed and deemed feasible), the energy required during the coast phase can be reduced by a factor of 10.

JEP-11 CONCLUSION

The study has shown that, for the given payload, a Jupiter entry probe of about 300 kg can be designed reaching a depth corresponding to a pressure of 100 bars.

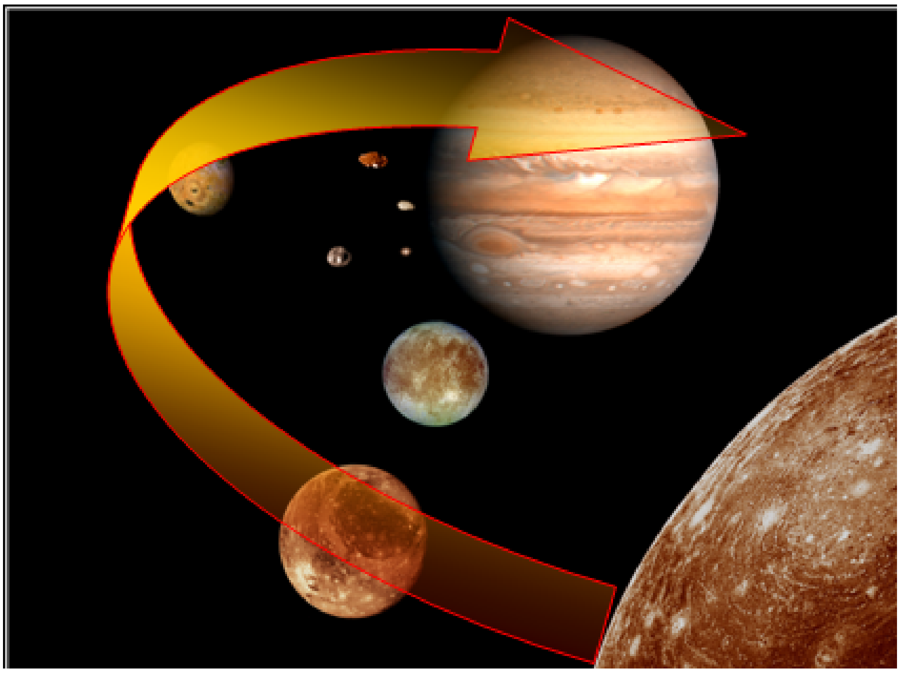
A smaller probe (about 10 % lighter) could be achieved if the requirement is relaxed to an altitude corresponding to a pressure of 20-40 bar. In this case, the mission mass capability would allow for two identical probes on one carrier (~700 kg dry mass) launched by a Soyuz-Fregat. Therefore, the requirement of atmospheric deep sampling needs to be traded against sampling in two different shallower atmosphere locations. On the other hand, lower altitudes corresponding to pressures in excess of 100 bar quickly become unfeasible because of the very high atmospheric attenuation and the associated low link margin or high communication power. Due to the very low resources and the atmospheric attenuation, communications are a major problem, driving the probe and spacecraft design. Furthermore, the distance between probe and relay spacecraft is limited to the ~4 R_J range to limit the required power to reach the carrier spacecraft.

Entry from a hyperbolic approach trajectory takes place 90 days after release. Only near equatorial latitudes can be targeted, as the study has shown that for higher latitudes the entry heat fluxes exceed the present capabilities of ablative thermal protection systems.

Jupiter entry probes face extreme aerothermodynamic challenges: the identification of adequate TPS is very challenging. Therefore the probe design includes a generous margin for TPS design. This is mainly due to the large uncertainty in the calculation of heat fluxes and performance of TPS in this thermal load range. Such uncertainties come from the fact that design and qualification will have to rely only on partial representation of the physical phenomena and on a reduced environment (testing in a representative environment is considered unfeasible, leading to large uncertainties in theoretical models).

High complexity and extreme test conditions are major cost drivers. The TPS design and qualification is the most critical issue of the mission. Therefore, a careful margin philosophy is required and the option of flying two identical probes may help reducing the mission risk. Next to this, highly integrated electronics will be required to minimise the required resource allocation.

THE JOVIAN SYSTEM EXPLORER



JSE-1 INTRODUCTION

This chapter provides an overview of the on-going Jovian System Explorer (JSE), a TRS currently performed with EADS Astrium (F, UK and D). This study addresses magnetospheric payload requirements in combination with the release of an atmospheric probe (the JEP). The mission scenario is selected to thoroughly investigate the *implications* of magnetospheric payloads and the accommodation and release of a probe.

As the study is still ongoing, the information does not reflect the latest developments. An update will be provided later this year once this study is completed.

JSE-1.1 Outstanding questions related to the Jovian magnetosphere

As explained before, the Jovian System is very much like a miniature solar system, as a result of the many similarities with our solar system. By gaining a good understanding of the Jovian System's evolution, answers could very well be found regarding the formation and evolution of solar systems. By following the history of the universe, we can understand what happens not only to stars themselves, but also to the raw material for life, which they continuously alter through their evolution [32].

Three main themes have been identified by the scientific community concerning the exploration of Jupiter. These themes were addressed during the ESLAB 39 symposium (Cosmic Vision 2015-2025 – Call for themes) [33]:

- Study the composition of Jupiter to understand the formation of giant gas planets
- Study of Europa to understand its physical state and its potential for supporting life
- Study of the Jovian magnetosphere (including the interactions with the Jovian moons)

The challenge of studying Europa has been addressed in more detail during the Jovian Minisat Explorer Study. Therefore the Scientific Payloads and Advanced Concepts Office has decided to investigate the technological challenges that are associated with the study of the other two points. This is achieved by the study of two technology reference cases: the in situ measurement of the Jovian atmosphere by means of entry probes (see previous chapter for details) and the study of a Jovian magnetospheric explorer (The Jovian System Explorer). Even though the JEP study has been addressed in the previous section, the accommodation of the probe has not been addressed in detail. Therefore the JSE study also focuses on this aspect.

Despite the wealth of information provided by previous missions, especially the Galileo orbiter, the understanding of the Jovian magnetosphere is far from complete. The following issues shall be the focus for this reference study and are a summary of topics discussed in [34]:

The most critical and fundamental open question is the role of the solar wind and the interplanetary magnetic field in shaping the topology of the magnetosphere and driving its dynamics. The size of the magnetosphere is strongly influenced by the solar wind pressure. However it is unclear whether these influences are limited to the outer magnetosphere or if it also affects the mid and inner magnetosphere and if so, to what extent. Does magnetic reconnection of the interplanetary and magnetospheric field lines play a role in the overall energy and mass budget of the Jovian System? Can the coupling of the solar wind energy and mass influence or even drive magnetospheric convection?

Major breakthroughs have been achieved regarding the understanding of the Earth's magnetosphere by a continuous and global study of the auroral emissions simultaneously with the solar wind parameters. These observations enabled the development of advanced magnetic field models, which helped to establish the links between auroral features and magnetospheric key regions. In the Jovian System, the link between the magnetospheric plasma and the auroral features is still to be conclusively established, and significant additional knowledge is required to fully understand the mechanism of this interaction.

A last point mentioned here is the study of the magnetosphere as a particle accelerator. It still is unclear how the neutral Io plasma (Io expels approximately 1 ton/s of mainly SO_2 through volcanic activity) is accelerated from a few eV to energies in the 100 keV to MeV range. The mechanism that releases the SO_2 particles in the interplanetary space, even into Earth's magnetosphere, also needs to be understood. The mid to deep tail regions are still largely unexplored, and as they have a fundamental effect on the overall dynamics of the magnetosphere, they are of key interest for the understanding of the Jovian magnetosphere.

JSE-1.2 Considered scenarios

In view of the previous considerations, the following areas are considered as prime targets for this study:

- the magnetotail
- the dayside magnetopause
- the polar region (auroral studies)

Depending on the selected configuration, one or more of these areas can be investigated. A one spacecraft mission will not be able to cover all these goals in a satisfactory manner, but will result in a less expensive and more versatile concept, while a multiple spacecraft configuration allows for a selection of different orbits, to optimise the different observation scenarios, even if it will be more complex and expensive.

The following orbits have been assessed in the context of the Jovian System Explorer study:

- Equatorial: 15x100 R_J or 15x 200 R_J to study the inner and outer magnetotail respectively (55 day and 140 day orbital periods)
- Equatorial: 70x15 R_J to study the magnetopause (35 day orbital period)
- Polar: 70x15 R_J, to study the magnetopause and the polar regions (35 day period)

Issues to take into consideration:

- The perijove altitude of 15 R_J has been baselined to avoid the main radiation belts of Jupiter, which are especially strong in the area starting in the vicinity of Europa and going inwards to Io
- Orbits have their apocentres at the region of interest (70 R_J between Jupiter and the Sun, 100-200 R_J in the tail)
- An orbit going through both the magnetotail and the magnetopause would most likely be too distant from the poles (in case of polar orbit) and in any case the orbital period would be too long to perform multiple measurements (194 day period).
- The feasibility of accommodating one (or more) atmospheric probes is also assessed. This implies that, if feasible, the selected orbit must be compatible with the probe's mission profile, which includes the communication between spacecraft and probe(s).

Initial orbit configuration specification

The following provides a summary of the orbit scenarios that have been envisaged for this study. The reference launcher is the Soyuz Fregat 2-1b.

A) Single spacecraft:

- This option should give the maximum mass available for an entry probe. The minimum solution still yielding interesting data would be a single spacecraft in the equatorial plane. The two orbits of interest would be a 15x70 R_J and 15x200 R_J

- Hyperbolic release for a probe is included

Figure 76: Single 15 x 70 RJ equatorial orbit

B) Two spacecraft in equatorial orbit:

- One 15x70 R_J orbit (magnetopause orbit), one 15x200 R_J orbit (magnetotail orbit)
- Alternatively, the 15x200 R_J could also be a 15x100 R_J orbit, to observe the inner part of the tail. Both orbits could be assessed by the same spacecraft by changing its orbit during the mission

- Hyperbolic release for a probe is included

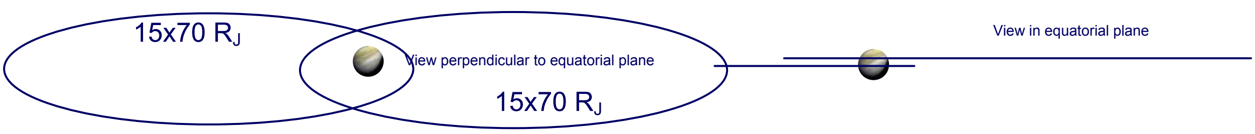


Figure 77: Dual 15 x 70 RJ equatorial orbit

C) Two spacecraft in polar and equatorial orbit:

- Same orbits as in B, but in this case the 15x70 R_J concerns a polar orbit, allowing for auroral measurements
- The natural orbit evolution of the polar orbit is also of interest to see what kind of coverage would be available if the orbit is allowed to be perturbed by Jupiter's J2 effect
- Hyperbolic release for a probe is included

Figure 78: 15 x 70 RJ equatorial and polar orbit

JSE-1.3 Science requirements

The following requirements were provided for this study: Generally speaking, to investigate space plasmas, all plasma properties at all timescales and length scales are of interest. Time and length scales are to a large extent determined by the number of spacecraft in orbit and shall not be treated here. Plasma properties include:

- Magnetic field (DC and AC in all three dimensions)
- Plasma particles
  - Electrons (energy/velocity and 3-D direction)
    - Low energy (>10 eV)
    - Medium energy (~10 keV)
    - High energy (~MeV)
  - Ions (energy/velocity and composition in 3-D)
    - Thermal ions
    - Suprathermal
    - Cosmic rays
  - Interaction of particles
    - With neutral particles (both high and low-energy)
    - With photons (x-rays, gammas etc.)
    - With dust
- Electric field (DC and AC in all three dimensions)

JSE-1.4 Reference payload

The tables below show the instrument suite that was provided as strawman payload for this study based on a spin stabilised spacecraft. As for the JME, this is not intended as a definitive payload, but the instruments are representative of magnetospheric payloads and help clarify their implications on the spacecraft.

Table 40: JSE strawman payload

Instrument	Acronym	Mass (kg)	Peak power (W)	Size (cm3)	Data rate (kb/s)	Comment
3-D E-field analyser	EFA-3D	7	5	2 packages of ~25x20x25 (axial boom) 4 packages of 30x15x10	0-200	4x40m wire booms, frequency up to ~3 MHz for AKR/AHR
DC magnetometer	DMAG	1.5	0.6	5x5x3	0.28 - 8.2	assuming 2 m boom. E-m cleanliness requirements might drive to longer boom and thus higher instrument mass (TBC)
AC magnetometer	AMAG	1.5	0.1	10x10x10	0.19 - 200	assuming 1m boom.
Electron/Ion Electrostatic analyser	ELA	2.5	3	2 packages of (10x10x15)	0.5 - 15	
Ion composition analyser	ICA	5	8	45x25x20	0-60	
Active Spacecraft Potential Control	ASPOC	2	2	20x15x20		5,000 hr operation, optionally 2 instruments to increase lifetime (TBC)
Neutral particle detector	NPD	2.5	3	15x15x15		
Energetic particles detector	EPD	1.8	2	2 packages of ~5x5x5	0 - 13	
Dust analyser	DustA	2	3	3 packages of D=10, L=20	0.02	
Camera (VIS)	JoCa	1.65	1	300x180x105	2.1	Only operating when in vicinity of Jupiter (miniumum mode of 0.1 kbps for non dynamic mode if data rate is exceeded)
UV Camera	UVCam	1	1	40x40x100	0.2	Only operating when in vicinity of Jupiter, not in combination with magnetospheric
Power Supply Unit	PSU	1	4.3	120x110x60		
Data Processing Unit and Central Processing Unit	DPU and CPU	2	1.9	120x75x20		
Margin (20%)		6.29	7.0			
Total		**37.7**	**41.9**	**-**		**-**

The following table provides an overview of typical accommodation requirements for the strawman payload defined for the JSE study.

Table 41: Accommodation and FOV specs for strawman payload

Instrument	Acronym	Number of packages	Position requirements / FOV
3-D E-field analyser	EFA-3D	4 (wire)	4x40m wire booms: In spin plane separated by 90 deg
DC magnetometer	DMAG	1	2 m boom: in spin plane
AC magnetometer	AMAG	1	1m boom: in spin plane
Electron/Ion Electrostatic analyser	ELA	2	5 x180 deg in spin plane (⊥ x // to spin axis)
Ion composition analyser	ICA	1	12 x 160 deg in spin plane (⊥ x // to spin axis)
Active Spacecraft Potential Control	ASPOC	1	Opening ±15 deg, along spin axis
Neutral particle detector	NPD	1	9 x 180 deg, in spin plane (⊥ x // to spin axis)
Energetic particles detector	EPD	2	12 x 260 deg, in spin plane (⊥ x // to spin axis)
Dust analyser	DustA	3	100 deg, one ⊥ to spin axis, two along spin axis: one in sun and one in anti sun direction
Camera (VIS)	JoCa	1	90 deg along spin axis in anti-sun direction. The location should be as close to the spin axis as possible, provided that the FOV requirements are met. Attention shall be paid w.r.t to the main engine and possible axial antenna's
UV Camera	UVCam	1	90 deg along spin axis in anti-sun direction

The following information is a summary of the on-going JSE study and is taken from [19] and [23].

JSE-2 STUDY OBJECTIVES

The objective of the Jovian System Explorer Technology Reference Study is to establish a cost efficient and technologically feasible mission architecture for a multi-mission spacecraft exploration of the Jovian magnetosphere and atmosphere, while providing a preliminary assessment of the programmatics, of the enabling technology required developments, and a rough order of magnitude cost assessment. Furthermore the study shall demonstrate that the proposed mission configuration fulfils the scientific requirements.

The scientific context selected for this study is the investigation of the Jovian magnetosphere with emphasis on possible fleet definition, complementary to the expected science return of the NASA JUNO mission, and the in-situ investigation of the Jovian atmosphere, with emphasis on the feasibility to accommodate an entry probe as defined in the JEP CDF study. This is much different from the former Jupiter Minisat Explorer TRS, which was dedicated to the exploration of the Jovian moons and especially Europa. The mission architecture baseline investigated in that frame consisted of a 3-axis stabilised relay orbiter in orbit around Jupiter and a small Europa orbiter.

The output of the current study is the definition of a feasible mission architecture, encompassing number of spacecraft and their engineering design, launch options, launch and transfer scenarios, capture scenario and orbital combinations in the Jovian System in support of a programmatic analysis aimed at:

- providing the technology drivers
- identifying solutions for reducing the overall system cost and solving technology issues
- preparing a technology development and demonstration plan for mission enabling and mission enhancing technologies

The following table shows the options that have been considered for this study. In the end scenario 2.1 was selected, as it was the best reference to understand the implications of systems that were not yet addressed in the JME study.

Table 42: Considered scenarios for the JSE study

	S-F (direct or GTO)	AR5 ECA direct injection	Type of transfer	Type of carrier spacecraft	Type of passenger spacecraft	Operational orbits	Comments
Scenario 1	direct		VEE	1 spinner	1 atmospheric probe	Equatorial 15Rjx70Rj	Basic magnetospheric / atmospheric mission with one probe
Scenario 2.1	GTO		VEE	2 identical spinners (separated after launch)	1 atmospheric probe	2 equatorial 15Rjx70Rj (with opposite apses)	Complete magnetospheric mission with 2 orbiters and one probe
Scenario 2.2	GTO		VEE	2 identical spinners (separated after launch)	None	1 equatorial 15Rjx70Rj, 1 polar 15Rjx70Rj	Basic magnetospheric mission with 2 orbiters, one in polar orbit
Scenario 2.3	GTO		VEE	1 spinner	1 TBD passenger (separated after PRM)	Carrier spinner in equatorial 15Rjx70Rj	Basic magnetospheric mission with 1 orbiter and one TBD passenger
Scenario 3.1	GTO		VEE	1 3-axis stabilised JRS-like	1 atmospheric probe	Galileo-like Jovian tour	Magnetospheric / atmospheric / moons remote sensing mission with one probe
Scenario 3.2	GTO		VEE	1 3-axis stabilised JRS-like	1 impactor (separated before arrival)	Galileo-like Jovian tour	Magnetospheric / atmospheric / moons remote sensing mission + Europa impactor
Scenario 3.3	GTO		VEE	1 3-axis stabilised JRS-like	1 TBD passenger separated after PRM	Galileo-like Jovian tour + relay orbit	Magnetospheric / atmospheric / moons remote sensing mission + TBD orbiter (e.g. Europa orbiter)
Scenario 4	GTO		SEP	1 3-axis stabilised JRS-like S/C with SEP	1 TBD passenger separated after PRM	Galileo-like Jovian tour + relay orbit	Magnetospheric / atmospheric / moons remote sensing mission + TBD orbiter (e.g. Europa orbiter)
Scenario 5	GTO		MEE	2 identical spinners (separated after launch)	None	2 equatorial 15Rjx70Rj (with opposite apses)	Basic magnetospheric mission with 2 orbiters, no Venus fly-by
Scenario 6		X	VEE	2 identical spinners (separated after launch)	2 identical atmospheric probes	1 equatorial 15Rjx70Rj, 1 polar 15Rjx70Rj	Complete magnetospheric / atmospheric mission with 2 orbiters (one in polar orbit) and 2 probes
Scenario 7.1		X	VEE	1 3-axis stabilised JRS-like	2 identical atmospheric probes	Galileo-like Jovian tour	Magnetospheric / atmospheric / moons remote sensing mission with 2 probes
Scenario 7.2		X	VEE	1 3-axis stabilised JRS-like	1 impactor (separated before arrival)	Galileo-like Jovian tour	Magnetospheric / atmospheric / moons remote sensing mission + Europa impactor
Scenario 7.3		X	VEE	1 3-axis stabilised JRS-like	1 TBD passenger separated after PRM	Galileo-like Jovian tour	Magnetospheric / atmospheric / moons remote sensing mission + TBD orbiter (e.g. Europa orbiter)
Scenario 8		X	MEE	1 3-axis stabilised JRS-like	2 identical atmospheric probes	Galileo-like Jovian tour	Magnetospheric / atmospheric / moons remote sensing mission with 2 probes, no Venus fly-by
Scenario 9		X	VEE	1 3-axis stabilised JRS-like and 1 spinner	1 atmospheric probe and 1 TBD passenger	1 equatorial 15Rjx70Rj, and 1 Galileo-like Jovian tour / relay orbit	The full mission !

JSE-3 Baseline Mission Scenario

The baseline mission scenario consists of a Soyuz-Fregat launch into a GTO orbit of a stack comprising two identical spinners and one atmospheric probe. The two spinners are separated just after launch and follow the same trajectory up to Jupiter capture. The transfer trajectory is a Venus–Earth-Earth Gravity Assist trajectory and the capture scenario includes a Io gravity assist followed by an insertion manoeuvre with a pericentre at a Jupiter distance about 3.5 R_J and an apojove of 420 R_J.

The probe release occurs about 90 days before arrival, followed by an Orbit Deflection Manoeuvre. The probe relay phase is expected to occur shortly after the pericentre, which means that the insertion manoeuvre must be delayed or advanced by a few hours. Optionally the gravity assist can be performed at Ganymede, with a small Delta-V penalty, in order to provide more time between the gravity assist and the insertion burn and probe data relay critical events. The applicable transfer scenario is shown in Figure 79. The propellant budget is derived from a worst case Delta-V envelope of the applicable transfer cases.

It must be noticed that this preliminary mass budget exhibits an atmospheric probe mass allocation which is just at the limit of the ESA CDF JEP study conclusions on the minimum feasible mass for a 40bar probe. Therefore it was agreed that if the baseline scenario detailed design phase demonstrates that the probe mass allocation is really not enough, then one spinner would be discarded and the baseline mission scenario would then consist of only one spinner and one atmospheric probe (both potentially more efficient). This option would still be very valuable in terms of study outcomes, since the lessons learned would be mostly the same (except for the spinner separation strategy and the communications with two spacecraft), even if the scientific merit is reduced.

Launch Dates	Jan 2017	Jan 2017	Mar 2019	Mar 2020	May 2021	Jun 2023	Aug 2024
Transfer duration (years)	5.75	6.75	7.1	6	7.2	6.6	7
Departure declination	16°	18°	-3°	4°	-14°	-20°	-20°
Departure Vinfinity	3.05 km/s	3.32 km/s	3.114 km/s	3.449 km/s	3.569 km/s	3.3 km/s	3.481 km/s
LEOP correction	~35 m/s						
Apogee raising ΔV	692 m/s (400000km apogee)						
Gravity Losses (~2%)	~15 m/s						
Inclination change	0 m/s (all departure declinations within +/-20°)						
Escape ΔV	495 m/s	570 m/s	512 m/s	608 m/s	645 m/s	564 m/s	618 m/s
Gravity Losses (~5%)	~25 m/s	~29 m/s	~26 m/s	~30 m/s	~32 m/s	~28 m/s	~31 m/s
Deep Space ΔV	262 m/s	88 m/s	20 m/s	0 m/s	0 m/s	0 m/s	0 m/s
GA Navigation ΔV (15m/s per GA)	45 m/s (VEE)	45 m/s (VEE)	60 m/s (EVEE)	45 m/s (VEE)	45 m/s (VEE)	45 m/s (VEE)	45 m/s (VEE)
Jupiter approach navigation ΔV	20 m/s						
Approach Vinf	6.09 km/s	5.97 km/s	5.57 km/s	5.58 km/s	5.48 km/s	5.6 km/s	5.85 km/s
Capture ΔV via Io GA. Perijove = 3.5Rj, Apojove = 420Rj	~530 m/s	~525 m/s	~445 m/s	~445 m/s	~440 m/s	~450 m/s	~500 m/s
Gravity Losses (~3%)	~16 m/s	~16 m/s	~13 m/s	~13 m/s	~13 m/s	~13 m/s	~15 m/s
Perijove Raising Manoeuvre	~375 m/s						
TOTAL	~2510 m/s	~2410 m/s	~2213 m/s	~2278 m/s	~2312 m/s	~2237 m/s	~2346 m/s

Figure 79: Applicable Transfer Scenarios

JSE-4 Spacecraft design

As explained at the beginning of the chapter, the study is currently ongoing, therefore the level of detail is limited. However the composite spacecraft overall configuration at launch is shown in Figure 80.

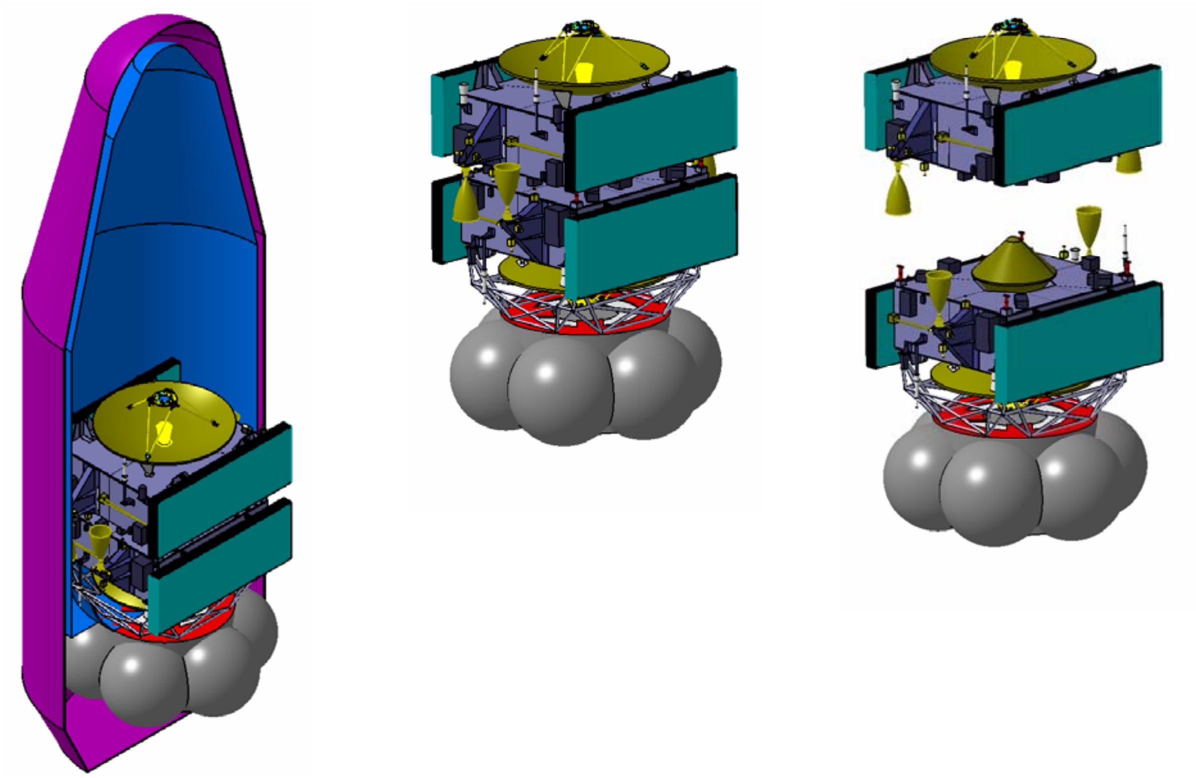


Figure 80 Composite Spacecraft Launch and release configuration

The deployed and stowed configurations of the spinner are shown in Figure 81, together with an internal view. Both spinners are identical except for the atmospheric probe presence (if applicable) and the propulsion tanks configuration. Each spinner accommodates two fixed solar arrays of five panels totalling about 27m^2, which provide up to 300W EOL. A battery module of 40.5 Ah / 875 Wh is proposed in view of the BOL capacity requirement of 835 Wh.

A two 500 N, 325 s I_{sp} main-engine configuration is the recommended baseline, with the main engines mounted on external engine pods at diametrically-opposed positions around the centre of mass. This external mounting is necessary to allow accommodation of other mission elements (in particular the large HGA and the probe), as well as to allow vertical stacking of the two spinners in the launcher fairing.

A set of 8x10N thrusters, arranged in two clusters of four on the main engine pods, provide single failure tolerant attitude control, able to satisfy the worst case slew rate requirements experienced prior to probe-relay phase. This system provides the necessary force and torque capabilities for spin rate and spin axis pointing control, as well as longitudinal and transverse manoeuvres.

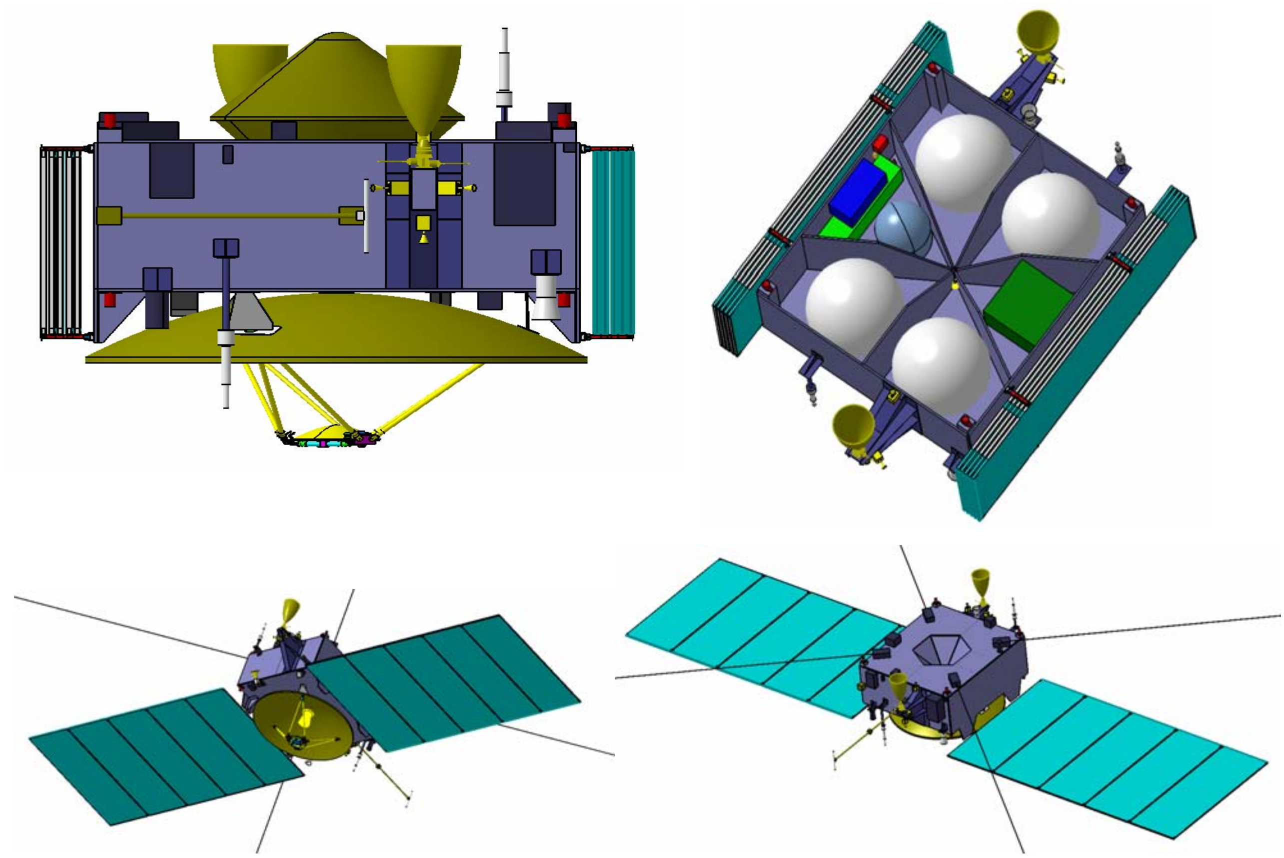


Figure 81 Stowed and deployed spinner Configuration

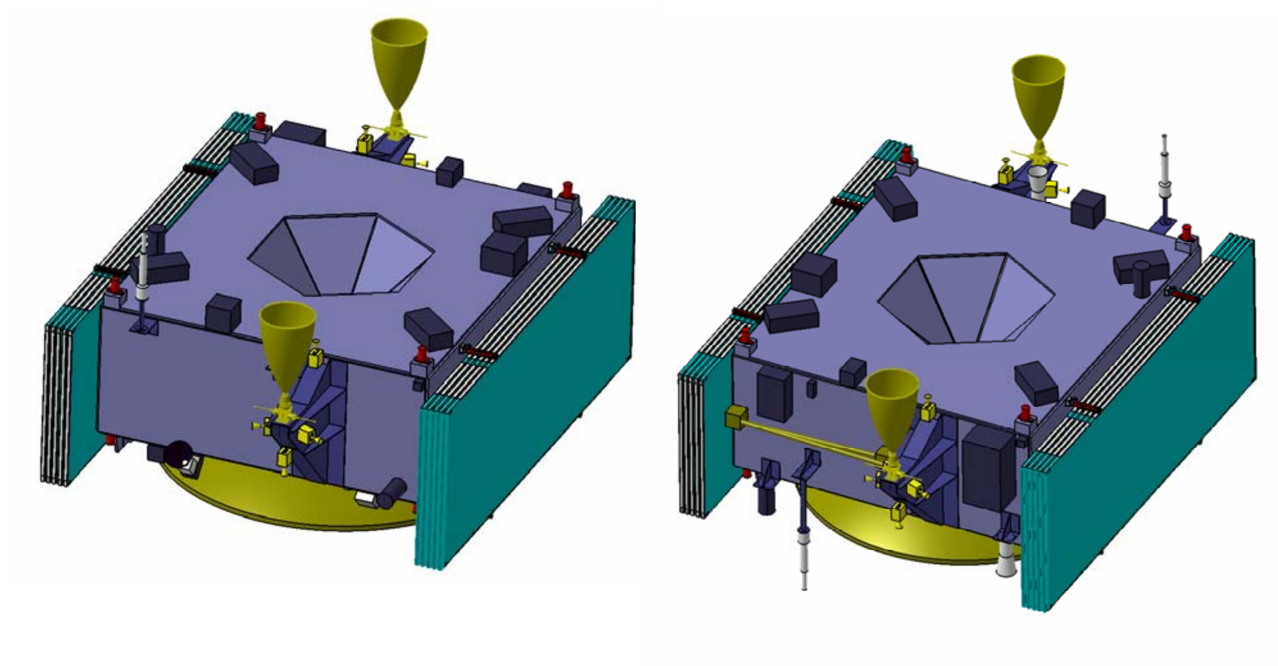


Table 43: Accommodation of strawman payload (stowed configuration) [23]

The spacecraft candidate functional architecture, based on the highly integrated avionics suite is depicted in the next figure.

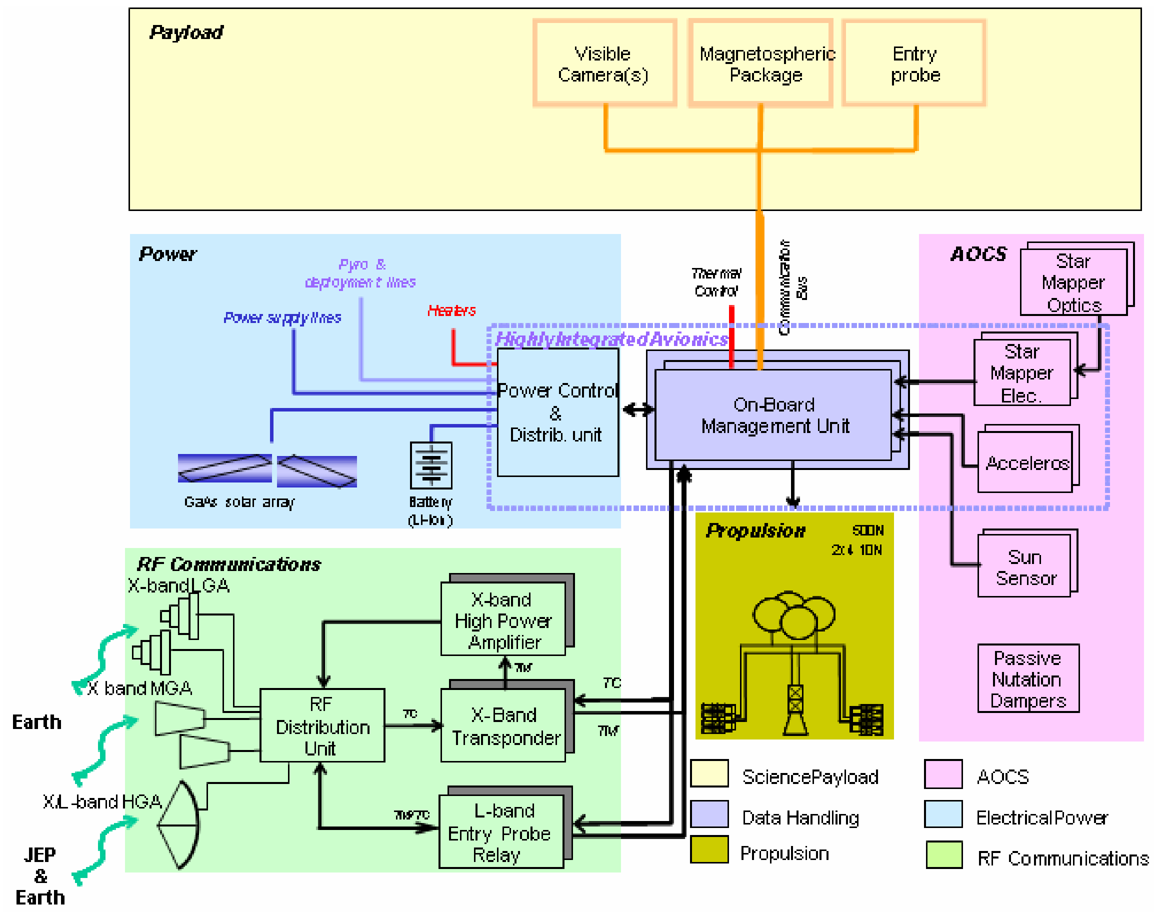


Figure 82 Functional Architecture

Since the spinner is nominally always Earth-pointed and assuming a downlink period of 8 hour per day, a solid state mass memory of a few Gbit is sufficient to recover a few missed station visibilities.

JSE-5 Concluding remarks

As the JSE study has not been concluded yet (the final report is pending), only a quick overview is provided of the JSE study. Detailed mass budget and spacecraft summaries will therefore have to be provided at a later date. However, a preliminary result and a list of lessons learned is presented in this document.

The following System Drivers have been identified in this study and are a direct consequence of the selected baseline [23]:

- Payload and probe relay performance:
  - Magnetospheric payload requires a spinning spacecraft
  - Payload accommodation constraints (40m wire booms, magnetometer boom, EMC)
  - Probe has imposed a large HGA diameter (>2m), with major configuration impacts
  - Probe ruled out using Ka-band, because of spinner pointing performance limitations (even with an articulated antenna)

- solar array sizing :
  - power budget estimated at 300W EOL
  - required iteration on capture trajectory to reduce 1MeV e- equivalent fluence
  - different solar cell cover glass thicknesses considered, 500µm finally not optimal due to the losses caused by the cover glass (this is valid for the JSE scenario, since the radiation dose is much lower than for JME)

- configuration drivers :
  - spin stabilisation constraints: inertia ratios and alignment constraints (antenna boresight, atmospheric probe longitudinal axis and main engine thrust axis must all be aligned with the spin axis)
  - solar cells and HGA are on the same side as the sun and Earth are in the same direction w.r.t the spacecraft
  - main engines on the opposite side, to avoid long battery-powered manoeuvres (due to trajectory geometry during apogee raising and Earth escape burns, JOI and PRM burns)
  - atmospheric probe preferably on the opposite side from HGA and solar cells, to maintain Earth / Sun pointing attitude during probe release

- critical launch mass margin: structure in CFRP, high performance engine (500N EAM with I_{sp}=325 s. However, this requires a long nozzle, complicating the engine accommodation)

- trajectory constraints: radiation doses, pointing strategy, thermal impacts (LILT temperature at Venus fly-bys)

- As a consequence of all these points and the required propellant, the study has shown that a scenario with two magnetospheric spinners and a 300 kg probe is unfeasible with a Soyuz Fregat 2-1b launch. Alternative scenarios are a two spinner scenario without probe or a one probe and one spinner configuration.

As a result, a list of enabling technologies has been identified and summarised in Table 44:

Table 44: List of enabling technologies for JSE

System	Technology	Technology validation approach
Electrical	Rad-hard components	Specify, design & qualify new components. Delta-qualify existing components
	Shielding material	Specify, design & qualify an enclosure box adapted to JSE objectives
Avionics	Software for high autonomy	Architectural design & validation
	Highly integrated Avionics	Consolidate HICDS technology program
	Star mapper and X-beam Sun sensor optical heads in harsh environment	Delta-qualify existing designs, radiation effects assessment and mitigation
	Autonomous spin axis manoeuvre scheduling	System simulator to validate operational scenarios
Power	LILT GaAs cells for harsh radiation environment	Specify & delta-qualify GaAs triple junction cells
	Customised Solar Arrays (high mechanical load)	Modify and delta-qualify panels / hinges

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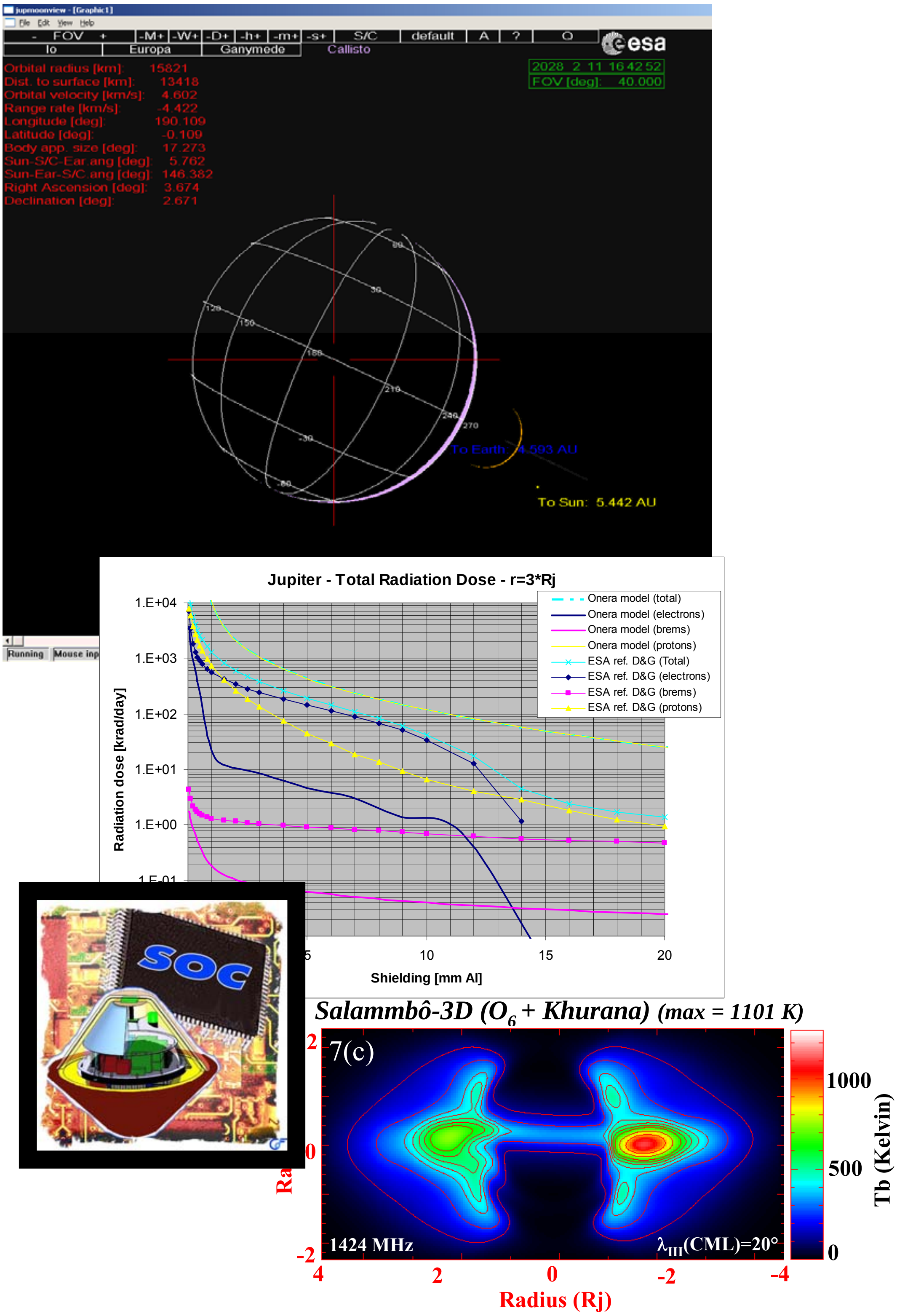
ONGOING AND PLANNED JOVIAN ACTIVITIES



ONGOING ACTIVITIES

The previous chapters provided an overview of the Jovian TRS studies that have been performed. In view of the identified problem areas additional activities have been started. This section provides a very concise overview of these activities. The level of detail is limited as the studies are ongoing.

Testing of Low Intensity Low Temperature Solar cells in representative environment

The journey to Jupiter for Jovian spacecraft will include a flyby to Venus - a hot and high solar intensity environment (the solar intensity at Venus is 2620 W/m2 and the solar cells will be exposed to temperatures of 450 K) - and will end up in the Jupiter system - a cold and low solar intensity environment.

The solar intensity at Venus is 2620 W/m^2 and the solar cells will be exposed to temperatures of 450 K, while at Jupiter the solar constant is 1/25 of the solar constant at Earth (i.e. 55 W/m^2) and temperatures can be as low as 120 K. Solar cells will degrade quickly in the severe radiation environment of the Jupiter system, effectively experiencing within a few weeks a radiation fluence equivalent to 15 years in the sort of geostationary Earth orbit typical for telecommunication satellites. This makes it essential to test them under representative conditions in order to allow accurate performance prediction. However, the simulation of a high radiation, low intensity environment is not trivial (typically, equivalent testing for geostationary missions requires much lower radiation fluences and can be performed around room temperature).

In an ESA-funded project, state-of-the-art 'triple-junction' GaAs based solar cells are being tested in-situ under environmental conditions based on the JME study results, in order to verify whether their 'End-Of-Life' performance is sufficient to allow adequate generation of power without requiring an unacceptably large (and correspondingly heavy) solar array. Triple junction GaAs solar cells, as well as the individual pn junction elements which make the device, are being tested under varying conditions: the cells are irradiated by 1 MeV electrons with total fluences up to 10^{16} e-/cm^2 and temperatures of 120 K and 300 K in order to study the degradation profile and understand how it differs from degradation in Earth orbit. These experiments are still ongoing and results are being analysed, although the first preliminary results are promising they also underline the need to test under conditions which are as representative as possible of the real mission.

As explained before, the individual pn junction elements of the cell are studied next to the full cell to determine which degradation mostly influences the cell performance at LILT conditions. The next figure shows the general trend of the degradation of the solar cells as a function of the total electron fluence, which can be seen as a measure for mission lifetime. The degradation is relatively low up to a fluence of ~10^{14} and then the performance drops dramatically. The precise reason for this behaviour is still being studied and more tests will be performed to increase the quality of the statistics of the experimental results.

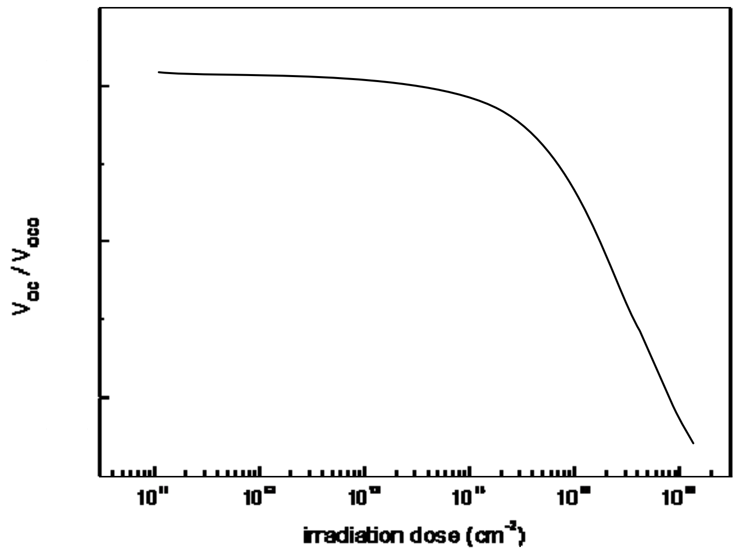


Figure 83: Solar cell degradation as function of fluence

Further testing will be performed in the first quarter of 2007. Together with a careful evaluation of the potential to tailor the solar cell design to Jupiter conditions, this activity will help to define the roadmap of activities necessary to facilitate a mission to the Jupiter system.

JME Tour Simulations

The trajectories of the JME spacecraft are quite complex and the numerous gravity assists offer many opportunities to study the bodies used for these GAMs.

A combination of existing and new tools is required to better understand these opportunities and to visualise the orbits. Currently ESOC has developed the JOVI tool that provides information on the fly-by parameters of the Jovian moons. In addition another tool is being developed providing scientific information of the entire Jovian tour. The latter tool is not yet available, however some results of the JOVI tool are summarised in the next section.

Additional activities are ongoing to provide visualisations of the entire JME trajectory. Currently several options are being assessed, varying from in house ESA tools to commercial tools such as Celestia and STK. This work is currently ongoing.

Science opportunities with JEO and JRS

As shown in the JME section, there will be several opportunities to perform scientific observations during the gravity assist manoeuvres (GAM) at the Galilean moons during the JEO and JRS tours.

Presently, there are 16 GAMs foreseen for the JEO: 1 at Io (performed while in composite configuration with the JRS), 6 at Ganymede, 2 at Callisto, 2 more at Ganymede, and 5 at Europa. The closest proximities to the surfaces of the moons range between 301 km at Io and 3068 km at Ganymede. The following table shoes details of the JEO GAMs. It is important to note that many of the GAMs will occur when the sub satellite point is on the dark sides of the moons. In these cases instruments that take data in the visual wavelength ranges will not be operated. Other instruments operate in different wavelength ranges and meaningful scientific observations are possible.

Table 45: Overview of the JEO GAMs

Moon	Event date [yyyy-mm-dd]	Event time [hh:mm:ss]	Illumination condition	Distance to surface [km]	Body apparent size [deg]	Sun-S/C-Earth angle [deg]
Io 1	2026-07-28	15:55:00	light	301	81	0.150
Ganymede 1	2027-07-14	08:18:40	dark	318	83	6.350
Ganymede 2	2027-09-23	21:44:21	dark	1452	66	3.281
Ganymede 3	2027-11-05	20:11:46	dark	3068	50	8.319
Ganymede 4	2027-12-04	11:10:02	dark	1209	69	10.213
Ganymede 5	2027-12-25	22:23:40	dark	352	83	10.381
Ganymede 6	2028-01-09	05:52:48	dark	1697	63	9.747
Callisto 1	2028-01-22	03:01:18	light	2143	56	9.604
Callisto 2	2028-02-11	17:37:49	dark	1739	60	5.756
Ganymede 7	2028-03-17	09:44:56	light	971	72	1.003
Ganymede 8	2028-04-07	20:58:34	light	1360	67	5.132
Europa 1	2028-05-03	02:31:12	dark	1715	51	8.792
Europa 2	2028-05-10	00:38:27	dark	441	76	9.468
Europa 3	2028-06-03	18:01:26	dark	509	74	10.714
Europa 4	2028-06-14	04:29:17	on terminator	304	80	10.771
Europa 5	2028-07-01	14:18:23	light	2421	43	10.144

The following figure shows a simulation of the view of the JEO 10 minutes before closest proximity to Io at the first GAM. Io is shown in yellow and the nadir direction is marked as a red dot. In the background Jupiter and its rings are shown in orange. In the top left corner several orbital parameters are listed, such as distance to the surface, which is 5512 km at the time of this simulation. Ten minutes later, at closest proximity to the surface, this distance will shrink to 301 km and the field of view will be completely filled with Io’s surface.

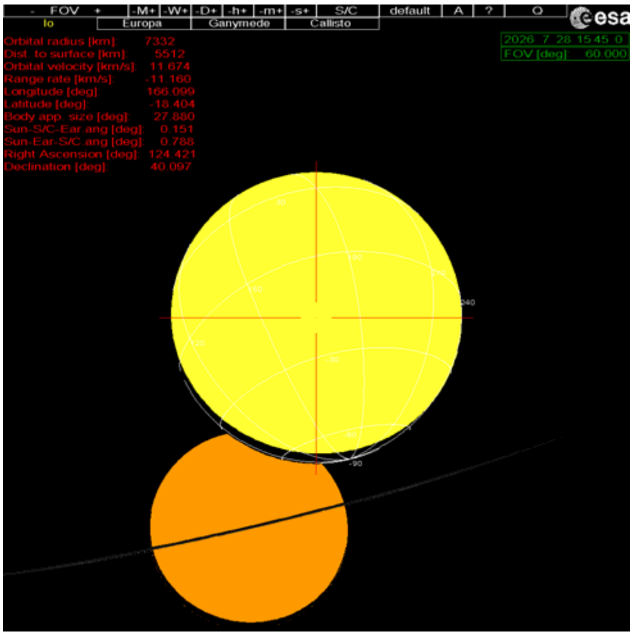


Figure 84: Simulation of the viewing angle and illumination conditions 10 minutes before closest approximation during the gravity assist manoeuvre at Io

For the JRS, there are 10 GAMs planned: 1 at Io, 6 at Ganymede, 2 at Callisto, and 1 more at Ganymede. The closest proximity to the surfaces of the moons range between 318 km and 6 295 km. Details of the JRS GAMs are shown in Table 46. Here, as with the JEO, most GAMS will occur when the sub satellite point on the dark sides of the moons.

Table 46: All GAMs planned for JRS

Moon	Event date [yyyy-mm-dd]	Event time [hh:mm:ss]	Illumination condition	Distance to surface [km]	Body apparent size [deg]	Sun-S/C-Earth angle [deg]
Io 1	2026-07-28	15:54:26	light	339	80	0.151
Ganymede 1	2027-07-14	08:18:40	dark	318	83	6.350
Ganymede 2	2027-09-23	21:44:21	dark	1452	66	3.218
Ganymede 3	2027-11-05	20:11:46	dark	3068	50	8.319
Ganymede 4	2027-12-04	11:10:02	dark	1209	69	10.213
Ganymede 5	2027-12-25	22:23:40	dark	352	83	10.381
Ganymede 6	2028-01-09	05:52:48	dark	1697	63	9.474
Callisto 1	2028-01-22	03:01:44	light	2145	56	8.604
Callisto 2	2028-02-11	17:42:52	dark	1975	58	5.756
Ganymede 7	2028-03-17	10:13:18	light	6529	32	1.067

HIPS Studies

Activities have been initiated to study the critical items of a Highly Integrated Payload Suite. A preliminary study of the HIPS principle was performed and parallel to this effort a breadboarding activity has been started of a front-end with accompanying SpaceWire interface and Leon-III based processor to test the working of a common DPU. A follow-on study regarding the detailed design of a potential payload suite which consists of a number of highly integrated instruments has been started in the first quarter of 2007. Among the critical items which will be addressed in the study are:

- Thermo-mechanical, optical & electrical design of the HIPS
- Heat flow in the integrated system to minimise the impact on the front-ends of the instruments
- Telecommanding and data rate of multiple front-ends handled by a common DPU including a possible software architecture.
- Cross-calibration issues including co-alignment among the different front-ends to maximise science return
- Assembly, integration and verification of a highly integrated payload suite

The results of the detailed design study can be used in a potential next phase to build an engineering model of a highly integrated payload suite in order to perform validation tests of the HIPS principle. Parallel to this study the breadboarding of two front-ends in combination with a common DPU will be performed to test critical issues of a HIPS system in an early phase.

Study of thermodynamic properties of the Jovian atmosphere at hypersonic entry

Currently an activity is ongoing to study the thermodynamic properties of He/H2 high-temperature components, which are encountered during Jovian entry. For these conditions it is necessary to revisit the chemistry models that are currently used for analysis of hypersonic entry conditions.

The properties are needed mainly in thermochemical-nonequilibrium hypersonic-flow solvers that are used to predict heat-flux distributions on spacecraft's critical surfaces during atmospheric entry. To do this, the properties of the relevant species (He, He^{+}, He^{++}, H, H^{+}, H^{-}, H_2, H_2^{+}, H_3^{+}, e^{-}) present in the high-temperature environment must be computed (projected temperature range is anticipated between 50 and 100,000 K).

This study will provide the data in these temperature ranges, which are currently not available. The result of this study can then be used for thermochemical-nonequilibrium calculations required for thermodynamic computations necessary to design the TPS system for a Jovian entry probe.

System On a Chip (SoC) study for JEP and JEO

The overall objective of the System on a Chip (SoC) CDF Study is to assess the benefits or otherwise of implementing SoC technologies into specific space mission designs.

The study activity design cases used studies previously done in CDF and projected them into the future to assess the system level differences and implications (mass, power, volume, operability, complexity, risk, etc.) of the System on a Chip approach. The study also assessed which functions are interesting to integrate in a single chip and what functions should be left independent. At first the application of technologies that are existing as well as under development was evaluated. The final objective was to elaborate the requirements for the system on a chip for subsequent technology investigations where appropriate.

Figure 85: SoC study logo

The JEP and JEO designs were used as references for this study. The final report is currently being compiled, however the following conclusions can be drawn [35]:

- A SoC solution for the JEP mission was derived enabling mass, volume and power reductions. A reduction in the Probe size was found to be possible with internal reconfiguration;

- Some power system reduction is possible for the JEP mission. The new design enables the use of fewer batteries because of a 10 fold reduction in timer power using SoC technologies. The 2.5 kg allocation for timer batteries reduces to 0.42 kg;

- MEMS technologies were applied to the JEP GNC: the IMU was replaced with SoC technology. This resulted in an ITAR free, low cost, space qualified option with a slight but acceptable performance degradation:
    - Small gain in mass and volume; 0.2 kg mass reduction
    - 8 W power reduction
    - 100 krad radiation capability

- DHS resource reduction is not the main factor, as the PCDU and SSPA are the dominant units and no significant changes have been identified using SoC;

- SoC solutions for the JEO mission were derived, leading to a small mass reduction and simplified payload and avionics development in both hardware and software;

- Revision of the shielding design for both 300 krad and 1 Mrad solutions lead to reduction in shielding mass when compared to the baseline JME design;

- All SoC technologies need to be available by 2016 for a 2019 launch.

PLANNED ACTIVITIES

Next to the previously mentioned ongoing activities, a set of planned activities are being prepared. The following list provides a short overview.

- A CDF study on radiation effects of Jovian environment on components and materials. This study aims to provide a credible development programme towards Jupiter qualified components by studying all aspects relevant to radiation (e.g. list critical components with current status, sector analysis and reconfiguration of components in spacecraft, specification of possible processes and design rules for radiation hardened electronics, ESD effects);

- A study on radiation mitigation methods. This study, which is closely related to the afore mentioned CDF study, is intended to understand the best way to shield components;

- Mechanical deployment of a scaled model of the EuGPR. This activity is intended to assess the feasibility of the in-space deployment of a folded antenna with many hinges. During the JME study this was identified as a major concern, which needed further analysis;

- A study of the solar concentrator concept proposed in the JME study. This analysis is proposed to identify the problem areas in the current concentrator concept and to study possible alternatives.

SUMMARY AND CONCLUSIONS

This document has presented an overview of the Jupiter related studies performed by ESA and industry for the identification of technologies required to enable resource limited cost-driven missions to the Jovian System. Top-level scientific objectives taken from inputs by the scientific community and relevant scientific reference papers have been taken as a guideline to determine the consequences of the payload on the spacecraft. Three mission scenarios have been accordingly considered and assessed, in conjunction with preliminary assumptions for a highly integrated, limited-resources strawman payload suite which allows for a meaningful payload for a low resource mission approach:

- The exploration of Europa: The Jovian Minisat Explorer
- In situ exploration of the Jovian atmosphere up to 100 bar: The Jupiter Entry Probe
- Study of the Jovian magnetosphere and the Jovian System: The Jovian System Explorer (study update is still in process).

Mission analysis, system architecture, critical technologies and development aspects have been identified for each of the above scenarios. Furthermore, risk and cost drivers as well as some specific management aspects related to an optional RTG based design have been discussed.

Other scenarios such as Europa impactor/penetrator systems have also been studied. For the impactors it was concluded that they are very complex, high risk and expensive in terms of mission, spacecraft and especially payload resources. Although the scientific interest is fully understood, this approach is not deemed compatible with the low resource concept presented in this document.

Next to the previously mentioned mission concepts, an overview has been provided of the ongoing and planned studies related to the Jovian System. These activities are mainly targeted at critical technology issues related to radiation effects and mitigation, power generation and the higly integrated payloads.

In conclusion, baseline mission design and technology concepts have been presented for JME, JEP and JSE scenarios, based on orbiters and entry probe concepts. The critical technologies that are required for these concepts have been presented, with the message that unless their development is initiated soon, the presented scenarios cannot be realised, even for a launch in 2019.

ACKNOWLEDGEMENTS

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ACRONYMS

3D-MCM	Three Dimensional MultiChip Module
AC	Alternate Current
AIV	Assembly Integration and Verification
AMAG	AC MAGnetometer
AMTEC	Alkali Metal Thermal-to-Electric Converter
AOCS	Attitude and Orbit Control System
ASPOC	Active Spacecraft Potential Control
AU	Astronomical Unit
BOL	Begin Of Life
BRG	Brayton Radioisotope Generators
C_D	Coefficient of Drag
CDF	Concurrent Design Facility
CDMU	Central Data Management Unit
CFRP	Carbon Fibre Reinforced Plastic
CME	Common Module
CoG	Centre of Gravity
COSPAR	Committee On SPAce Research
CPS	Central Power System
CPU	Central Processing Unit
CRETE	Configuration REvisit Targeting Europa
D&G	Divine and Garrett
DC	Direct Current
DM	Descent Module
DMAG	DC MAGnetometer
DPU	Data Processing Unit
DSM	Deep Space Manoeuvre
DustA	Dust Analyser
	3-D E-field analyser
EFA-3D	
ELA	Electron/Ion Electrostatic Analyser
ELRR	Europa Low Resource Radar
EMP	Europa Micro Probe
EMPIE	Europa Microprobe In Situ
EOI	Europa Orbit Insertion
EOL	End Of Life
EPD	Energetic Particles Detector
ESA	European Space Agency
ESD	ElectroStatic Discharge
ESOC	European Space Operations Centre
ESTEC	European Space Research and Technology Centre
EuGPR	Europa Ground Penetrating Radar
EuGS	Europa Gamma-ray Spectrometer
EuLat	Europa Altimeter
EuMAG	Europa Magnetometer
EuRad	Europa Radiometer
EuREM	Europa Radiation monitor
EuSCam	Europa Stereo micro-camera

EuUVcam	Europa UV camera
EuVN-IMS	Europa Visible and Near Infrared Mapping Spectrometer
EVEEGA	Earth-Venus-Earth-Earth Gravity Assist
FE	Finite Element
FoV	Field of View
FPA	Flight Path Angle
GaAs	Gallium Arsenide
GAM	Gravity Assist Manoeuvre
GIRE	Galileo Interim Radiation Environment
GPHS	General Purpose Heat Source
GPR	Ground Penetrating Radar
GTO	Geostationary Transfer Orbit
HEO	Highly Elliptical Orbit
HGA	High Gain Antenna
HIPS	Highly Integrated Payload Suite
ICA	Ion Composition Analyser
IMU	Inertial Measuring Unit
I_{SP}	Specific Impulse
ITAR	International Traffic in Arms Regulation
IUS	Inertial Upper Stage
JEO	Jovian Europa Orbiter
JEOMA	JEO Microprobe Analysis
JEP	Jovian Entry Probe
JME	Jovian Minisat Explorer
JoCa	Jovian Camera
JOI	Jupiter Orbit Insertion
JRS	Jovian Relay Spacecraft
JSE	Jovian System Explorer
JuDustor	Jupiter Dust Detector
JuMag	Jupiter Magnetometer
JuNaCam	Jupiter Narrow angle camera
JuPWI	Jupiter Plasma Wave Instrument
JuREM	Jupiter Radiation monitor
LEOP	Launch and Early Operation
LILT	Low Intensity Low Temperature
LHP	Loop Heat Pipe
MEMS	Micro Electrical Mechanical System
MGA	Medium Gain Antenna
MMH	MonoMethyl Hydrazine
MMRTG	Multi Mission Radioisotope Thermal Generator
NASA	National Aeronautics and Space Administration
NPD	Neutral Particle Detector
OBMU	On Board Memory unit
ODM	Orbit Deflection Manoeuvre
P/L	Payload

PCDU	Power Control and Distribution Unit
PRM	Perijove Raising Manoeuvre
PSM	Perijove Stabilisation Manoeuvre
RAM	Random Access Memory
RHU	Radioisotope Heating Unit
R_J	Jupiter radius
RPS	Radioactive Power System
RSE	Radio Science Experiment
RTG	Radioisotope Thermal Generator
S/C	Spacecraft
SCI-AP	Planetary Section of the Payloads and Advance Concepts Office
SiC	Silicon Carbide
SEU	Single Event Upset
SLU	Static Latch-Up
SoC	System on a Chip
SRG	Stirling Radioisotope Generator
SSPA	Solid State Power Amplifier
TBC	To Be Confirmed
TM	Telemetry
TM/TC	Telemetry and Telecommand
TPS	Thermal Protection System
TRS	Technology Reference Study
UV	Ultra Violet
UVCam	UV Camera
VEEGA	Venus-Earth-Earth Gravity Assist
VIS-NIR	Visible and Near Infrared
VLBI	Very Large Baseline Interferometry

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