REPORT (TECHNICAL, BUDGET, COST, MANPOWER, TRAVEL, AUDIT, ETC.)

EPIG CDF Study Summary Report

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1 REFERENCES

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RD[3] Scientific rationale for Uranus and Neptune in situ explorations, Mousis et al., Planetary and Space Science 155 (2018) 12–40

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2 ACRONYMS

Acronym	Definition
AOCS	Attitude and Orbit Control System
AOGNC	Attitude and Orbit Guidance, Navigation and Control
ASI	Atmospheric Structure Instrument
BC	Back Cover
BoL	Beginning of Life
BSSM	Back Shell Separation Mechanism
CDF	Concurrent Design Facility
DHS	Data Handling System
DM	Descent Module
DSN	Deep Space Network
DOF	Degree of freedom

Acronym	Definition
EDL	Entry, Descent and Landing
EDS	Entry Descent System
EOL	End of Life
EOM	End Of Mission
FDIR	Failure Detection Isolation and Recovery
FPA	Flight Path Angle
FS	Front Shield
FSSM	Front Shield Separation Mechanism
Gbits	Giga Bits
G/S	Ground Station
HGA	High Gain Antenna
HW	HardWare
IMU	Inertial Measurement Unit
JUICE	Jupiter Icy moons Explorer
LGA	Low Gain Antenna
LV	Launch Vehicle
MLI	Multi-Layer Insulation
MMRTG	Multi-Mission Radioisotope Thermoelectric Generator
MP	Main Parachute
MS	Mass Spectrometer
OBC	On-Board Computer
ODM	Orbit Deflection Manoeuvre
PAS	Parachute Assembly System
PC	Pilot Chute
PCDU	Power Conditioning And Distribution Unit
PEP	Planetary Entry Probe
RCS	Reaction Control System
RD	Reference Document
RF	Radio Frequency
RHU	Radio-isotope Heater Unit
RPC	Ring Plane Crossing

Acronym	Definition
RPS	Radioisotope Power System
RTG	Radioisotope Thermoelectric Generators
SADM	Solar Array Driving Mechanism
S/C	Spacecraft
SOFC	Solid Oxide Fuel Cell
SST	Study Science Team
TPS	Thermal Protection System
TRL	Technology Readiness Level
TT&C	Tracking Telemetry And Command
UOI	Uranus Orbit Insertion
USO	Ultra-Stable Oscillator

3 ACKNOWLEDGMENTS

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4 INTRODUCTION

4.1 Background

The EPIG (ESA Probe for Investigation of the Giants) study aims at exploring the feasibility of a European "stand-alone" mission to the Ice Giants (or Saturn as back-up), it corresponds to an ESA internal assessment of the capabilities by using European technology only to explore the outer planets. The study was carried out by an interdisciplinary team of experts from ESA in 8 sessions, starting with a kick-off on the 28th March 2019 and ending with an Internal Final Presentation on the 9th May 2019.

4.2 Objectives

The objective of the EPIG study was to:

- Assess a European mission to the Ice Giants (Uranus, eventually Neptune (and with Saturn as backup)) with launch in the period 2031-2045.
- Design a carrier spacecraft to deliver an atmospheric entry probe to one of the Outer Planets while using the carrier as a data relay.
- Reuse the Probe concepts already studied in the frame of the M* Ice Giants Mission and earlier PEP CDF studies (see RD[1], RD[4]).
- The design shall be based on the use of European technology and avoidance of RTGs and RHUs.
- For Uranus (baseline), the specific mission design shall reuse existing European technology and/or explore concepts that require technology developments if needed.
- For Saturn (Back-Up), the mission design shall be fully implemented with reuse of existing European technology.
- Highlight the technological areas for which mission enabling developments would be required (e.g. use of combustion/fuel cells for heat/electrical power generation on Uranus carrier).
- Assess the programmatic approach and the schedule constraints for the studied option(s).
- Assess the mission cost for the studied option(s).

5 SCIENCE OBJECTIVES & PAYLOAD

The Ice Giants, Uranus and Neptune, have been visited by the Voyager 2 spacecraft in 1986 and 1989, respectively. These two flybys raised many questions that need to be answered by dedicated missions. The Ice Giant system is a distinct class of planets, fundamentally different from the better explored gas giants, Jupiter and Saturn. Their study is critical and necessary to advance our understanding of the solar system origin and evolution (see RD[1] and RD[3]).

By extending the scientific objectives to Saturn, the use of a similar atmospheric probe as designed for the Ice Giants would lead to an improved understanding of the processes by which giant planets formed, including the composition and properties of the local solar nebula at the time and location of giant planet formation. This would allow to extend the legacy of the Galileo and Cassini missions by further addressing the creation, formation, and chemical, dynamical, and thermal evolution of the giant planets, the entire solar system including Earth and the other terrestrial planets, and formation of other planetary systems atmosphere using a single probe (see RD[5] and RD[6]).

Through the scientific return provided by an atmospheric probe, the highest priority is to determine the planet's bulk composition, including abundances and isotopes of heavy elements, while a second priority is the determination of the compositional, thermal and dynamical structure of the atmosphere.

5.1 Payload Components

Proposed instruments (model payload for the purpose of technical study) to address the planet's bulk composition are the Mass Spectrometer (measuring the atmospheric composition) and the Atmospheric Structure Instrument, providing supporting information on altitude profile (e.g. by pressure) and on the thermal condition, allowing for derive mixing ratio profile and detect possible condensation. The structure of the atmosphere could be addressed by the Atmospheric Structure Instrument, Camera/Radiometer, Photometer, and the USO/Doppler wind experiment. It must be noted that these instruments are a representation of a possible future payload (model payload) to provide reasonable requirements for the technical study. RD[1] provides additional information with respect to the assumed probe payload composition and potential future evolution.

Instrument	Mass [kg]	Power [W]	Data rate [kb/s]	Volume envelope [mm]	Notes/Heritage
Atmospheric Structure Instrument (ASI)	1.50	6.00	0.16	TEM, PPI: 205×30 Ø ACC: 79×58×68	Three core sensor packages: - three-axial accelerometer (ASI-ACC) - pressure profile instrument (ASI-PPI) - temperature sensors (ASI-TEM)
Mass Spectrometer	6.00	9.60	0.13	200×200×100	Ion Trap Mass Spectrometer, Rosetta/Ptolemy heritage
Ultra Stable Oscillator (USO)/Doppler wind experiment	1.80	12.00	n/a	150×150×118	USO for Doppler Wind Experiment; Huygens heritage
Camera/radiometer channels	1.44	9.60	1.747	100×100×200	For atmospheric cloud features; 17.5° FoV, 4 filters; Venus Express heritage
Photometer	0.36	1.20	0.00026	30×30×80	

Table 1: Baseline Model Payload for Atmospheric Probe

6 MISSION ARCHITECTURE OPTIONS

As discussed in section 4, a key objective of the Study was to explore two mission options, one to Uranus (and possibly Neptune) and as backup one to Saturn with similar science objectives, to gather scientific data from the descent of an atmospheric probe on the target planet. Also transfers to Neptune have been initially investigated but with the given constraints (mainly due to the avoidance of nuclear power) no feasible profile for A64could be found. Therefore a mission to Neptune is not further described in his report. Basic common elements of the mission architecture for the baseline and the backup are the following:

- Launcher: Ariane 64
- Carrier Spacecraft delivering the Atmospheric Probe at the target planet
- Atmospheric Probe

However, given the sizeable differences in heliocentric distances (Uranus ca. 18.3 - 20.1 AU, compared with Saturn ca. 9 - 10.1 AU), significant differences in energy generation were to be expected; while the Saturn carrier concept is fully based on solar power generation, the Uranus carrier concept was developed by exploring alternative energy sources than purely solar power and/or nuclear power. The final choice for solid oxide fuel cell is described in section 7.3.4. As such, the two mission options considered are summarized in the following figure:

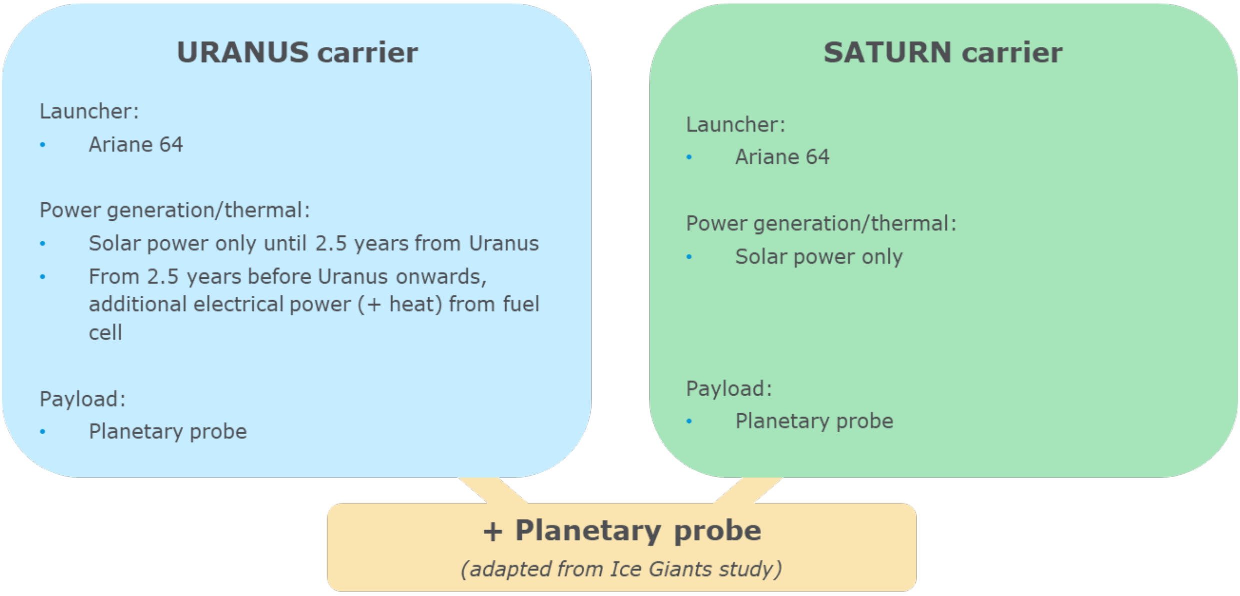


Figure 1: Mission Options

The detailed design of the carrier spacecraft is presented in section 7 for Uranus, section 8 for Saturn while section 9 provides a summary of the probe design that is common for both planets.

7 BASELINE ARCHITECTURE URANUS MISSION

7.1 Interplanetary transfer to Uranus

For the carrier, a number of interplanetary transfers were considered; an overview of these is presented in Appendix A. The key parameters to be considered for a launch opportunity are the following:

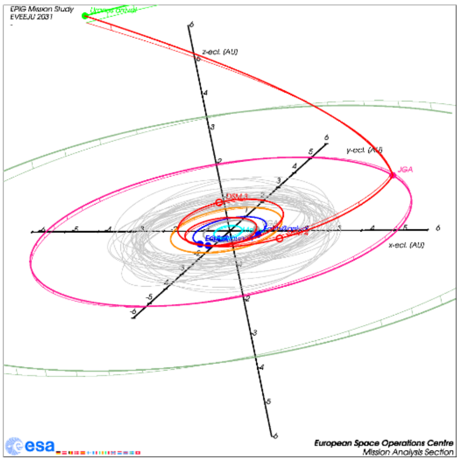
- Balance between the launcher performance (required escape velocity) and required Δv. These two factors determine the mass available for the carrier & probe composite.
- Arrival conditions of the probe (relative velocity at arrival) which determine the compatibility with the probe designed in RD[1] and RD[4].

As such, the transfer EVEEJU31 was selected as baseline, due to its high available wet mass, low Δv and intermediate probe arrival conditions (relative arrival velocity less than 23 km/s). It was initially desirable to choose a transfer without a Venus fly-by, however based on the preliminary system mass assessments, such options did not seem feasible and were discarded. As such, it is noted that the Venus fly-by likely drives the warm case for the system, which is particularly driving for the needed cryogenic fuel storage that will be further discussed in the following sections (see 7.3.6.1).

Note that the selected baseline does not have an adequate mission backup within the 2030's. An option could be to set the baseline as 2030 and to add an additional Earth flyby; or consider launches in the 2040's timeframe.

Mission architecture		
Launcher	Ariane 6.4	
Launch Date	Baseline	23/5/2031
	Alternative*	2030 or 2040's
Trajectory	Baseline	EVEEJU31
	Alternative*	EEVEEJU30 or TBD

***An earlier launch opportunity could be found in 2030 by adding an addition Earth fly-by, otherwise launches are only available in the 2040's



Case	EVEEJU31
Launch	2031/5/23+LP (3.45 km/s, -16.8 deg)
VSB	2031/9/18 (7686 km)
ESB 1	2032/6/11 (3656 km)
DSM 1	2033/8/13 (444 m/s)
ESB 2	2034/5/25 (300 km)
DSM 2	2034/9/18 (251 m/s)
JSB	2036/2/7 (795000 km)
Uranus arrival	2041/9/8 (11.8 km/s)

Figure 2: EVEEJU31 Interplanetary Transfer

Figure 2 shows the interplanetary transfer and a timeline with salient mission data, including Earth escape velocity and declination, swing-by altitudes, DSM sizes and Uranus arrival velocity. Figure 3 shows the probe entry conditions.

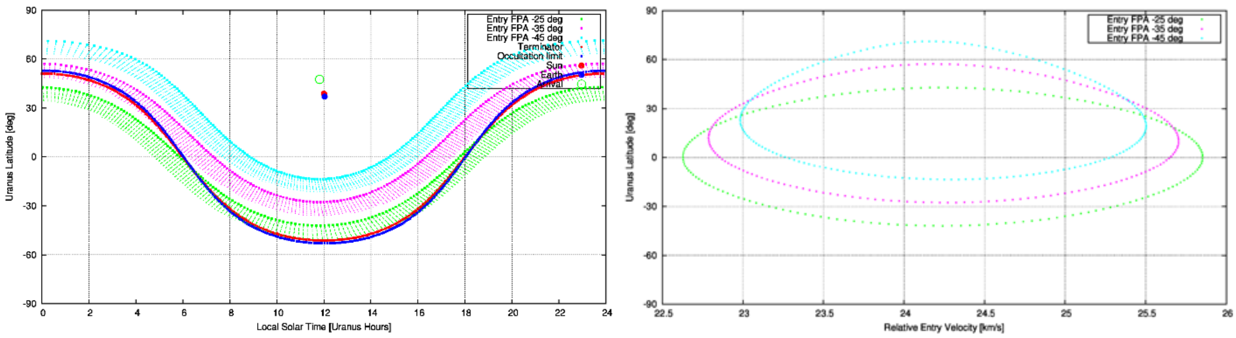


Figure 3: EVEEJU31 Entry Conditions and Velocity

7.2 Mission phases

The mission phases are presented in Table 2. The interplanetary transfer (10.3 years) is further subdivided in 2 sub-phases, given that at a certain point in the trajectory the solar power provided by the solar arrays is no longer sufficient to meet the spacecraft power requirements. This happens 7.8 years into the transfer and the remaining 2.5 years the alternative energy generation source that is used until arrival to Uranus, this alternative system is detailed in 7.3.4 and 7.3.5.2.

The probe coasting phase, after release from the carrier, is 5 days followed by the entry (6 minutes) and descent (60-90 minutes) phases respectively. The reason for 5 days coasting phase duration are further detailed in section 9.1.1. For the carrier to relay the probe data back to Earth 2 days have been allocated. The actual probe data should be possible to download in 3 hours, so there is sufficient margin for several attempts.

Mission phases	
LEOP	TBD
Interplanetary transfer (total)	10.3 years
- Interplanetary transfer (SA power >**208 W**)	7.8 years
- Interplanetary transfer with alternative energy generation (SA power <**208 W**)	2.5 years
Probe coast	5 days
Probe entry + descent	6+60-90 min
Data relay before EOM	2 days (TBC)

Table 2: Uranus mission phases

7.2.1 System modes

The system modes for the Uranus carrier are described here below:

Launch and early operations mode [LEP]	• Commissioning of the equipment • Preparation and injection into interplanetary transfer
Cruise Mode [CRM]	• 3-axis stabilized interplanetary transfer • Periodic communications to Earth • Energy provided by solar arrays (and secondary batteries)
Hibernation mode [HIM]	• Spin-stabilized interplanetary transfer • All units in minimal power hibernation mode • Receiver always on • Alternative energy generation via fuel cells
Probe Internal Check-out [PICM]	• Checkout of probe instruments during transfer, while the probe is still attached to the carrier
Communication mode [COM]	• Communication back to Earth
Receiving Mode [RCM]	• 3-axis stabilized, Earth-pointing mode for awaiting commands from ground (during pre-planetary navigation campaign)
Safe mode [SAM]	• Spacecraft kept in safe state • Earth-pointing, optimized to Sun
Manoeuvre mode [MAM]	• Performing main orbit manoeuvres using the thruster(s).
Probe Communications mode [PCOM]	• 3-axis stabilized mode pointing at probe to receive probe data during descent

Table 3: Uranus carrier system modes definition

The main equipment that are on, off or duty-cycled in each mode are presented in Table 4, with a mapping between mission phases and system modes. The mission feasibility heavily relies in the energy balance during the last 2.5 years where the power provided by the solar array is not sufficient to complete the mission. For that purpose, a Hibernation Mode similar to that performed by the Rosetta mission has been developed in order to reduce the overall energy required during the last part of the cruise; therefore, minimum power consumption while still maintaining the basic functionalities needed to preserve the safety of the spacecraft.

Mission phase	LEOP	Inter-planetary (0- 8yrs)	Inter-planetary transfer (last 2.5 years)					Probe release + descent	
System modes	**LEP**	**CRM**	**HIM**	**PICM**	**COM**	**RCM**	**SAM**	**MAM**	**PCOM**
Transmitter									
Receiver									
OBC									
AOCS (STR)									
SADM									
Thrusters									
PCDU									
Alternative energy generation									
Probe	Heat	Heat	Heat		Heat	Heat	Heat	Heat	

Green – High duty cycle; Orange – Low duty cycle or in stand-by; Red - OFF

Table 4Uranus carrier duty cycles (design cases), the allocation to the Probe is based on heater power in the absence of RHUs

7.2.2 Mission timeline and operational concept

The mission timeline and operational concept for Uranus is presented in Figure 4 (first 7.8 years) and Figure 5 (last 2.5 years).

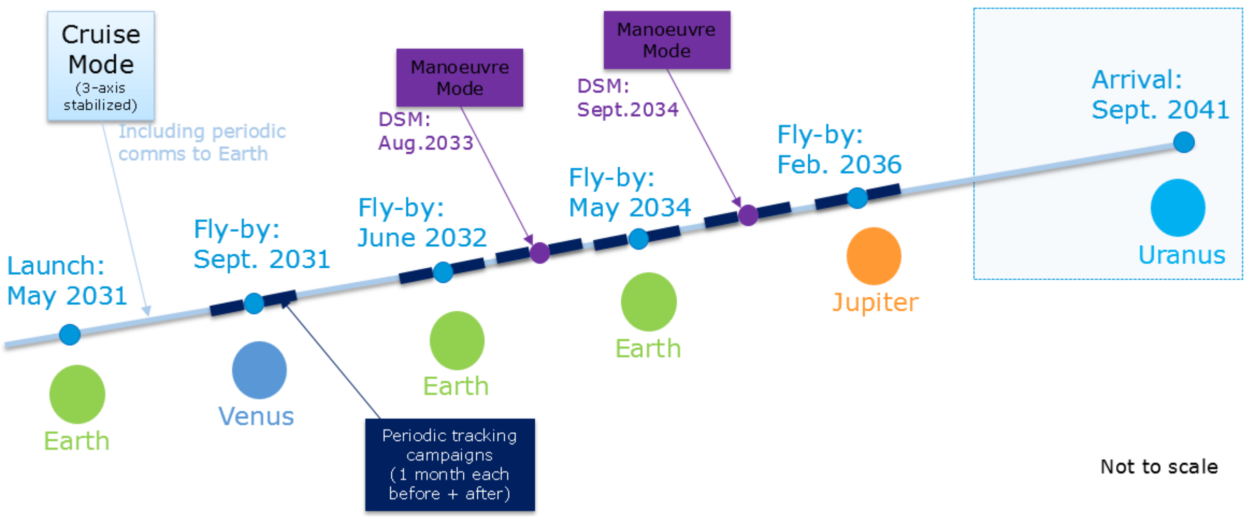


Figure 4: Uranus Mission timeline (first 7.8 years)

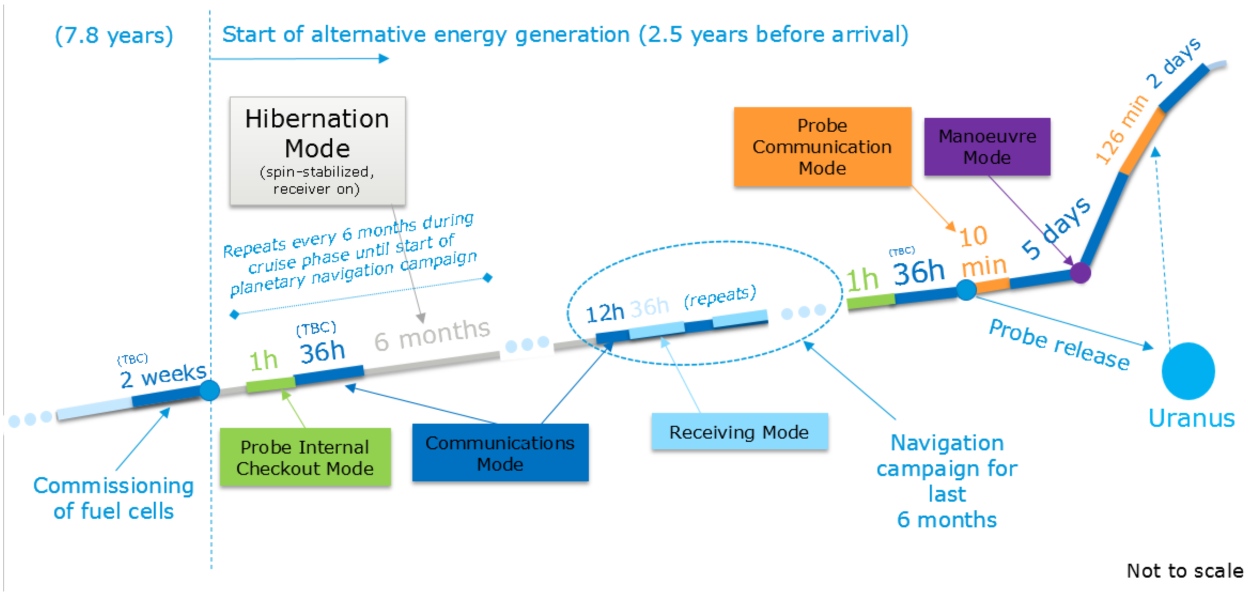


Figure 5 Uranus mission timeline (last 2.5 years)

Regarding the timeline for the last 2.5 years it should be noted that:

- Several values, such as the duration for fuel cell commissioning or the duration for the communication modes, are preliminary and should be assessed in future phases.
- During hibernation there should be a monthly check of spacecraft spin-rate using the gyros, which is not shown in Figure 5
- The carrier baseline design assumes communications with the probe for a 90 minutes descent, but the carrier trajectory described in 7.1 would only allow up to 75 minutes data relay during descent.

7.3 System Baseline Design URANUS

7.3.1 Baseline overview

Carrier Uranus		
Mass (kg) (incl. 20% system margin)	Dry mass (incl. 420kg probe):	3364 kg (including 990 kg of propellant for full cell)
	Propellant mass:	1026 kg
	Wet mass:	4390 kg
Payload	420 kg Planetary probe	

Propulsion	10x Bi-prop thrusters 4x propellant tanks + pressure tank 16x Methane/LOX thrusters 2x fuel cell propellant tanks
AOCS	2x accelerometers 2x star-trackers 6x sun-sensors for survival mode 2x Gyros for safe mode only
Communications	1xHGA, 2xLGA and 1xMGA X-band uplink/downlink: 3.7 kbps@Uranus UHF for probe downlink: 2 kbps
Power	120m2 Solar arrays (2 wings) 5x fuel cells 3x26kg batteries
Data Handling	CDMU (with low power mode for Uranus hibernation) RTU
Structures	464 kg
Thermal	Heaters, MLI, Radiators 78W for propulsion system 12.W heater power for Methane/LOX and dedicated sunshield
Mechanisms	Probe deployment mechanism SADM
Radiation Shielding	15kg

Table 5: Uranus carrier overview

7.3.2 System Budgets

The mass budget for the EPIG spacecraft is provided in Table 6. The detailed mass breakdown considers only the carrier. All dry masses include maturity margin and an additional 20% system margin applied to the total dry mass of the spacecraft. The harness mass has been accounted as 5% of total dry mass of the system. For the planetary entry probe, a total of 420kg (including margins, see section 9) are added on top.

The wet mass accounts for fuel and oxidizer masses for both the fuel cell system and the chemical propulsion system. For the fuel cell system, a 5% margin is added to the masses of Methane and Oxygen. Concerning the chemical propulsion system, the margin is already embedded in the mass calculations of both fuel and oxidizer. The adapter mass is based on a standard Ariane 6 launch adapter.

S/C Mass Budget		Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		4.8
Communications		62.8
Chemical Propulsion		221.6
Data-Handling		34.1
Mechanisms*		66.0
Power		595.4
Structures		464.2
System Engineering		17.7
Thermal Control		76.4
Harness	5%	77.1
Dry Mass w/o System Margin		**1620.2**
System Margin	20%	324.0
Dry Mass incl. System Margin		**1944.2**
Probe		420.0
Dry Mass incl. Probe		**2364.2**
Methane for Fuel Cell Mass		177.4
Methane for Fuel Cell Margin	5%	8.9
Oxygen for Fuel Cell Mass		769.1
Oxygen for Fuel Cell Margin	5%	38.5
Mass incl. fuel		**3358.0**
CPROP Fuel Mass		386.6
CPROP Fuel Margin	0%	0.0
CPROP Oxidizer Mass		622.4
CPROP Oxidizer Margin	0%	0.0
CPROP Pressurant Mass		5.6
CPROP Pressurant Margin	0%	0.0
Total Wet Mass		**4372.7**
Launcher Adapter		115.0
Wet Mass + Adapter		**4487.7**
Target Wet Mass incl. Adapter		5166.0
Below Target Mass by		678.3

***Probe separation mechanisms included in this budget**

Table 6: Mass budget for Uranus case

The power budget for the EPIG spacecraft is provided in Table 7 including a total system margin of 20%.

Row Labels	COM	HIM	CRM	MAM	SAM	PICM	PCOM	RCM
⊟SC (Spacecraft)	318	128	164	393	244	230	229	152
⊟AOGNC	1	0	1	1	6	0	1	1
ACC_AIQ (Accelerometer Innalabs AIQ2030)	0.0	0.0	0.0	0.2	0.0	0.0	0.0	0.0
ACC_AIQ_2030_2 (Accelerometer Innalabs AIQ 2030_2)	0.0	0.0	0.0	0.2	0.0	0.0	0.0	0.0
STR_TermaOH (STR Terma T1 Optical Head)	0.4	0.0	0.4	0.4	0.4	0.0	0.4	0.4
STR_TermaOH_2 (STR Terma T1 Optical Head #2)	0.4	0.0	0.4	0.4	0.4	0.0	0.4	0.4
GYRO_Sireus_1 (GYRO SireusNG10 TAS-UK #1)	0.0	0.0	0.0	0.0	2.5	0.0	0.0	0.0
GYRO_Sireus_2 (GYRO SireusNG10 TAS-UK #2)	0.0	0.0	0.0	0.0	2.5	0.0	0.0	0.0
⊟COM	117	30	39	30	39	30	60	30
HPA_TWTA_X_BEPI_1 (High Power Amplifier TWTA Bepi X #1)	33.7	0.0	3.4	0.0	3.4	0.0	0.0	0.0
HPA_TWTA_X_BEPI_2 (High Power Amplifier TWTA Bepi X #2)	33.7	0.0	3.4	0.0	3.4	0.0	0.0	0.0
RX_1 (Receiver #1)	0.0	0.0	0.0	0.0	0.0	0.0	15.0	0.0
RX_2 (Receiver #2)	0.0	0.0	0.0	0.0	0.0	0.0	15.0	0.0
XPND_TASI_XX_DST_1 (Transponder TASI XX DST #1)	25.0	15.0	16.0	15.0	16.0	15.0	15.0	15.0
XPND_TASI_XX_DST_2 (Transponder TASI XX DST #2)	25.0	15.0	16.0	15.0	16.0	15.0	15.0	15.0
⊞CPROP	0	0	4	240	0	0	0	0
⊟DH	29	6	29	29	29	29	29	29
CDMU_1 (Command and Data Management Unit #1)	20.0	5.0	20.0	20.0	20.0	20.0	20.0	20.0
RTU (Remote Terminal Unit)	9.1	0.9	9.1	9.1	9.1	9.1	9.1	9.1
⊟INS	0	0	3	0	0	31	0	0
Probe (Probe)	0.0	0.0	3.1	0.0	0.0	31.4	0.0	0.0
⊟MEC	0	0	20	0	0	0	0	0
SADM (Solar Array Drive Mechanism [incl SADE])	0.0	0.0	20.0	0.0	0.0	0.0	0.0	0.0
⊟PWR	44	44	20	44	44	44	44	44
PCDU (Power Conditioning & Distribution Unit)	20.0	20.0	20.0	20.0	20.0	20.0	20.0	20.0
SOFC (Solid Oxide Fuel Cell)	4.8	4.8	0.0	4.8	4.8	4.8	4.8	4.8
SOFC_2 (Solid Oxide Fuel Cell #2)	4.8	4.8	0.0	4.8	4.8	4.8	4.8	4.8
SOFC_3 (Solid Oxide Fuel Cell #3)	4.8	4.8	0.0	4.8	4.8	4.8	4.8	4.8
SOFC_4 (Solid Oxide Fuel Cell #4)	4.8	4.8	0.0	4.8	4.8	4.8	4.8	4.8
SOFC_5 (Solid Oxide Fuel Cell #5)	4.8	4.8	0.0	4.8	4.8	4.8	4.8	4.8
⊟TC	127	49	49	49	127	96	96	49
HTR_PRB (Heater Probe)	31.0	31.0	31.0	31.0	31.0	0.0	0.0	31.0
HTR_BATT_1 (Heater battery #1)	5.0	5.0	5.0	5.0	5.0	5.0	5.0	5.0
HTR_FC_TNK_1 (Heater fuel cell tank #1)	12.5	12.5	12.5	12.5	12.5	12.5	12.5	12.5
HTR_CARR_PROP (Heater Carrier Prop)	78.0	0.0	0.0	0.0	78.0	78.0	78.0	0.0
Grand Total	**318**	**128**	**164**	**393**	**244**	**230**	**229**	**152**
Total /w Margin	**381**	**154**	**196**	**471**	**293**	**276**	**275**	**183**

Table 7: Power budget for each system mode – (mean power in watts)

7.3.3 Composite Configuration

The baseline design of the EPIG composite spacecraft for the URANUS option is illustrated in the following figures:

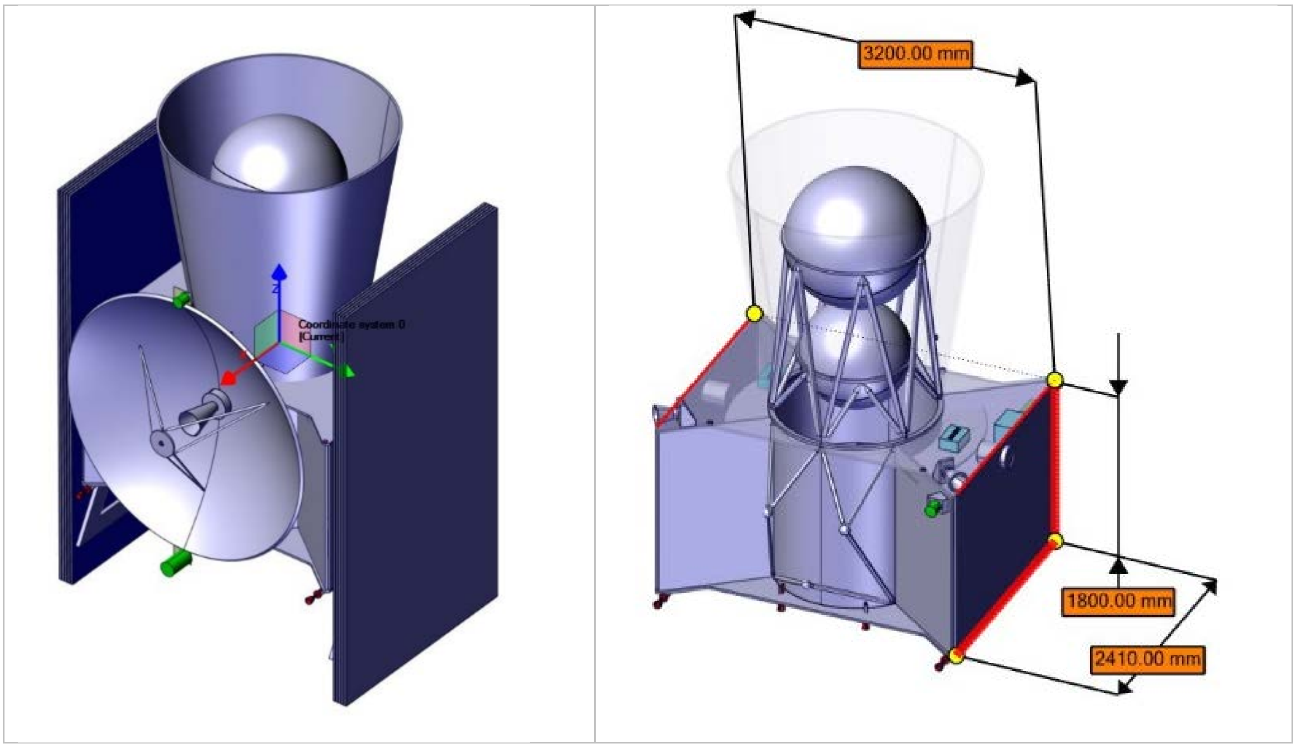


Figure 6: Spacecraft Axis System and dimensions

The lateral dimension of the spacecraft body shown in Figure 6 is driven by the size of the Probe, HGA as well as the height of the Solar Panels. The 420 kg Probe and 40 kg HGA gives static unbalance for the total spacecraft. To balance this problem, the probe is located as close as possible to the central tube and the HGA as far as possible from the central tube. Furthermore, the external appendages of the EPIG spacecraft are shown in the following figure:

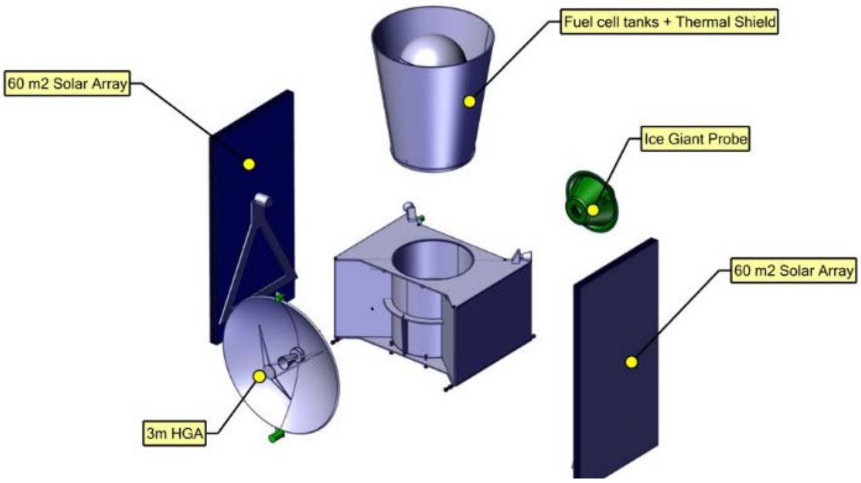


Figure 7: Spacecraft external appendages

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7.3.4 *Alternative Energy Generation Trade-Off*

The key driver for the Uranus mission design was the provision of an alternative means of power and heat generation at high heliocentric distances. These would be required to supplement the low solar power input at distances beyond Saturn.

Given that one of the objectives of the EPIG study was to explore a design without radioisotope power sources, this led to the consideration of chemical-based systems. Two options were considered, namely a combustion process and a fuel cell system; these were combined with several options on the selected chemicals. Note that this trade-off also had major impacts on the thermal and propulsion designs (see 7.3.5.2 and 7.3.6.1).

A key constraint was the transfer to Uranus, which had a dual effect:

- Flybys at Venus (necessary to meet the launcher mass performance, see above) provided the hot case environment for the cryogenics, limiting the choice of certain fuels and driving the carrier configuration.
- Longer durations post-Saturn increasing the amount of time that the alternative energy source must provide power/heat to the system.

The final trade-off selected a solid oxide fuel cell based on LOX/methane for the additional energy source.

Collated below in Table 8 are various plausible sources of heat and power generation. In the upper part of the table are listed the theoretical upper limit of performance of various power sources, in which the mass of associated hardware and fuel tanks is not considered. For the lower three options, details of the practical implementation are known with sufficient maturity to allow the presentation of realistic all-inclusive performance figures. The red part of the table deals with the production of heat. Here the mass-specific energy is presented, together with derived values of mass required to produce 1 W of heat continuously for 1, 5 or 10 years. The purple text considers electrical generation, with informed assumptions made about the realistic heat-to-electricity conversion efficiency of a plausibly applicable power conversion technology.

After thermal analysis (see 7.3.6) and the collation of the spacecraft electrical power budget (Table 7), it was determined that the minimum heat dissipation needed in the main spacecraft body, in order to maintain acceptable temperatures, was at all stages of the mission exceeded by the total dissipation of the electrical units (e.g. avionics etc.). Some additional heating is required, but for non-central equipment such as the probe or tanks. In these cases, electrical heaters for practical purposes must provide the heat. The net result of these considerations is that no direct heat production from the “alternative energy generation” system is needed. All of the required energy is needed in electrical form. Therefore, the purple-text right hand side of the table is the more relevant one for the EPIG trade-off.

			Heat required:	1 W				Elec. power required:	1 W			
			Thermal		kg required					kg required		
			with zero losses		Years of operation			Convers.	Electrical	Years of operation		
	Fuel	Ref / notes	MJ/kg	Wh/kg	1	5	10	Effic.	Wh/kg	1	5	10
Not including tanks, pipes, hardware, burners, engines etc	Hydrazine decomposition	Average case from Bechert, 1984 RRD manual.	1.6	444	20	99	197	0.3	133	66	329	657
	H2 + O2 PEM Fuel Cell	Burke NASA/TM—2003-212730. November 2003. AIAA–2003–5938	13.2	3667	2	12	24	0.5	1833	5	24	48
	LPG Butane + O2 SOFC Fuel Cell	Theroetical enthalpy at STD conditions is 10.7 MJ/kg. Needs to run at high temp (700 degC)	10	2778	3	16	32	0.55	1528	6	29	57
	MMH & NTO bipropellant	NIST+ CEA (Propulsion presentation Session 1)	6.3	1750	5	25	50	0.3	525	17	83	167
	Am-241	Physics	2E+06	6E+08	0.01	0.01	0.01	0.05	3E+07	1	1	1
All-inclusive	Li primary battery TRL9	e.g Saft Li SOCl2, Li SO2	-	400	22	110	219	-	400	22	110	219
	European Am241 RHU TRL4	Univ. Leicester. Final Report of "RHU Prototype Development". G619-012EP, ESA Contract 4000114783	-	-	0.07	0.07	0.07	-	-	-	-	-
	European Am241 RTG TRL 4	Univ. Leicester: "Thermoelectric Converter System for Small-Scale RTGs" T903-006EP. Contract 4200023026	-	-	-	-	-	-	-	1	1	1

Table 8: Alternative Energy Generation Options

As radioisotope power sources are disregarded *a priori*, the theoretical specific energy of H_2 - O_2 reaction is the next best option. However, the very low boiling point of hydrogen means that storage for many years (especially during Venus flyby) would not be possible without major boil-off loss. The hydrocarbon (e.g. methane, butane) – O_2 reaction is almost as high in mass-specific energy, and LOX would be potentially manageable over long timescales with pressurized storage. Various methods for extracting the energy of a hydrocarbon–O_2 reaction are possible, these are further considered below:

- Combustion is the obvious way of reacting hydrocarbons and oxygen for exothermic gain. If the energy were needed in the form of heat, this would be very efficient (but with some inevitable heat loss via exhaust gas). If electrical power is needed, the heat of combustion can be converted to electricity via a heat engine with a generator (e.g. alternator) attached. Combustion produces hot gases – this means that the best type of heat engine to use is most likely one using the Brayton thermodynamic cycle - i.e. a gas turbine connected to a rotary electrical generator. The conversion efficiency would depend on a multitude of parameters but would be almost certainly less than

<50%, and a high-speed mechanical device as a gas turbine is not well suited to multi-year reliability in space.

- A Stirling engine has been developed to high TRL in USA for radioisotope power generators (the NASA - Lockheed Martin ASRG). A European space Stirling engine is under development by TAS-UK, also targeted for application in radioisotope power generators. However, as a closed cycle engine, Stirling is not as well suited to a combustion heat source, is likely to have lower efficiency than Brayton in this context, and will be heavier for the same power output.
- Thermoelectric (Seebeck) power conversion is a reliable and motionless technology for heat-to-electrical power conversion. Space flight heritage exists in the radioisotope thermoelectric generators (RTGs) that have powered NASA missions such as Voyager, Cassini and MSL Curiosity. European space thermoelectric generators are developed to TRL 4 by the University of Leicester for application to Am241-powered RTGs. However, the conversion efficiency of thermoelectric conversion is relative low (e.g. 6%). This would greatly increase the quantity of fuel and oxidizer needed to produce a given amount of energy.
- A fuel cell system is very well suited to the EPIG application. The efficiency is high (50 to 60%), and it is a static, vibration-free device. Successful space heritage of fuel cells exists from Apollo and STS. However, these heritage applications were both alkaline fuel cell systems using H_2 and O_2 reactants. To use hydrocarbon-O_2 as reactants in a fuel cell is certainly possible, but it must be a solid oxide fuel cell (SOFC). The efficiency is very high (up to 60%), but they need to run at high temperature (>700°C in the fuel cell core) which may introduce some difficulty in implementation.

Nonetheless, a SOFC is selected as the most promising applicable technology for "alternative energy generation" for EPIG. The hydrocarbon fuel could be any from the smallest molecule (methane) up to longer hydrocarbons (e.g. kerosene). The trade-off logic and selection of liquid methane ("LNG") as the baseline fuel, as well as the design of reactant storage system, is described in 7.3.5.2.2.

7.3.5 Power subsystem design

7.3.5.1 Solar Array Sizing

A reasonable solar array of 120 m^2 area (two wings of 60 m^2) is achievable without major change from the JUICE or upcoming telecom spacecraft designs. All power required beyond this must be provided by the alternative energy generation.

Figure 8 shows the solar array power available (assuming 10% power conditioning and distribution losses) as a function of mission time, for the Uranus trajectory. Overlaid in red is the total power requirement (the early part of the mission is not shown). The power demand spikes represent the periods of high-power modes (COM and PICM), with the baseline level being the HIM (hibernation) mode power demand. This derives from the operations concept shown in section 7.2.2, and the power consumption in Table 7.

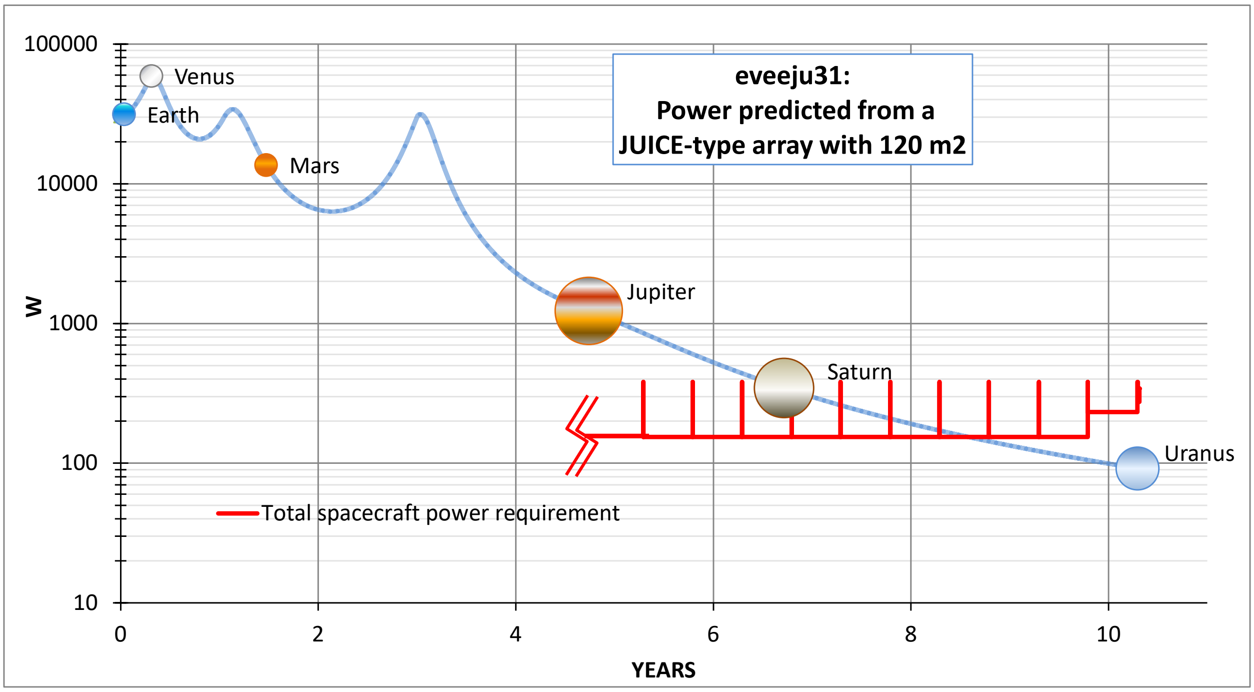


Figure 8: Solar Array Power available at user level (assuming 10% power conditioning and distribution losses) as a function of mission time, for trajectory eveeju31.

It can be seen that, at approximately 7 years after launch (around 9.7 A.U. from the Sun), the power demand begins to exceed the available supply from the array. The initial power shortfall is small in magnitude and short in time, and can be delivered from a reasonably sized secondary battery, with recharging performed afterwards in HIM mode.

However, from 7.8 years (12.3 A.U.) onwards, the energy shortfall is too great to be provided from the battery, not least because the ability to recharge is disappearing as the solar array output falls further towards the baseline HIM mode demand. From this time onward (around 2.5 years before arrival at Uranus), an “alternative energy generation” is required.

7.3.5.2 Fuel Cell System description

In order to estimate the total energy that must be provided by the SOFC (Solid Oxide Fuel Cell) system and therefore the embarked reactant quantity, it is necessary to consider:

- The spacecraft electrical power budget (Table 7)
- The operations concept “timeline” (section 7.2.2)
- The SOFC assumptions and operations (see 7.3.5.2.1)

This results in an electrical energy requirement that is essentially represented by the area between the solar generation and load demand curves, as highlighted by the green dotted circle below in Figure 9. The total required reactant energy (in terms of full reaction enthalpy) is estimated as 2.6 MWh (9.4 GJ).

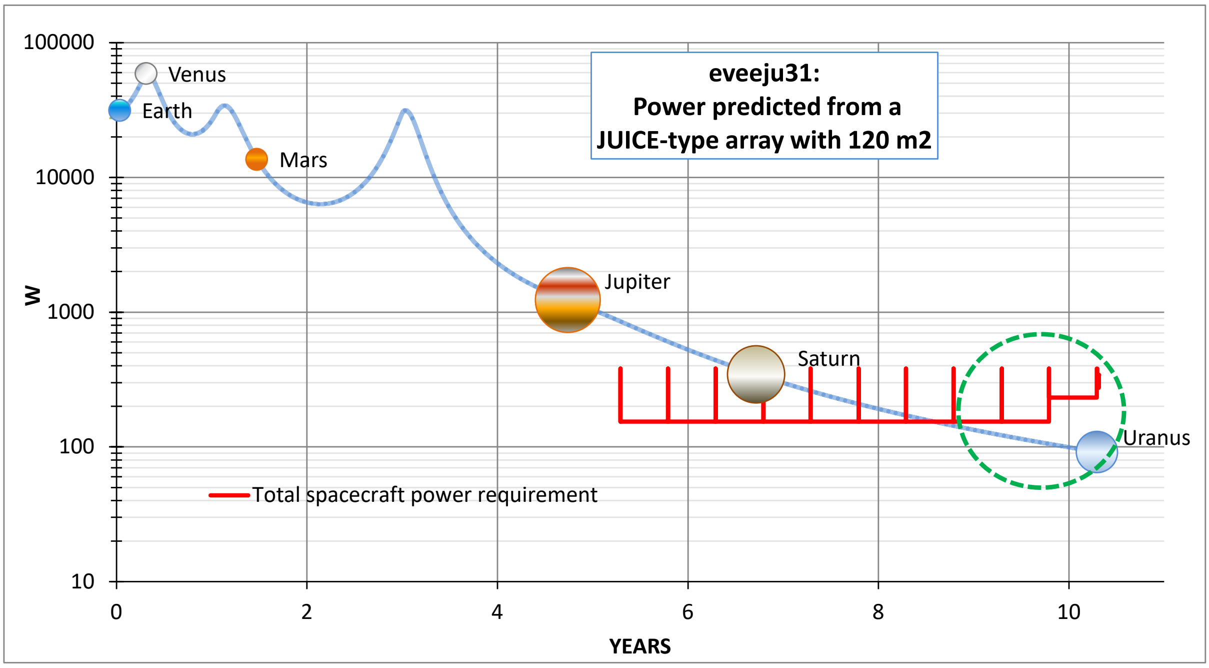


Figure 9: SOFC energy demand (area between red and blue lines in the green-circled zone)

7.3.5.2.1 Fuel cell operations and assumptions

The use of the fuel cell in EPIG is based in the following operational scheme:

- Operational temperature: the SOFC has a high "stack" (core) operating temperature at ~700°C (or more). It is assumed that the fuel cell will be able to maintain high temperature by self-heating after an initial turn-on by electrical heating.
- Throttleability: extreme "throttleability" (operation below nominal power output level) is unlikely because the cell core temperatures would not be maintained. The fuel cell will operate above a minimum power output level than allows maintaining the operating temperature.
- Optimization of consumables: in order to minimize reactant "wastage" by generating more power than required, more flexibility in the total SOFC power level is required than that of the throttleability alone. This leads to an architecture of multiple small fuel cells that can be individually activated to suit power demand as it increases as described in Figure 10. In the case of EPIG, a 4 nominal +1 redundant is selected as the baseline design.

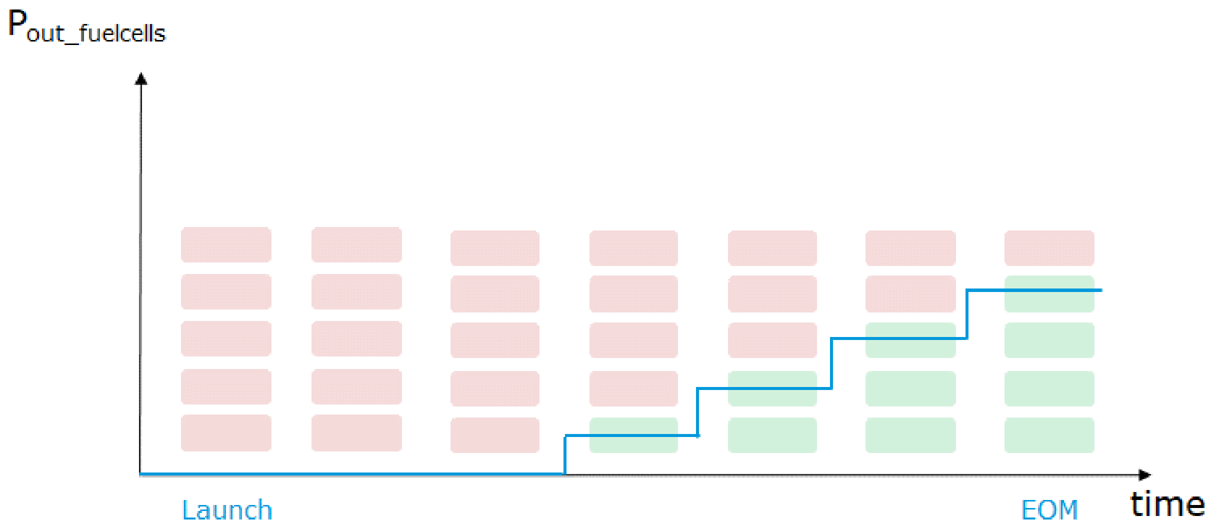


Figure 10: Fuel cell operations during mission lifetime (SOFC usage starts in the last 2.7 years of mission)

In summary, the fuel cells operational concept relies on the following unverified assumptions that would require a dedicated technology development:

1. The fuel cell can operate at or above 50% of its max rating, but not below; otherwise the cell core temperatures would not be maintained.
2. EPIG can embark small SOFC units of <150 W electrical output each, with the possibility to start & stop multiple times, using only the minimum number of units required at any one time.
3. An efficiency of 50% can be achieved (with very small fuel cells).
4. Power demand: 20 W is sufficient to power a common fuel cell management & control unit.
5. Mass per unit: 5 kg per SOFC unit is sufficient mass budget to cover both the SOFCs and their control unit.

7.3.5.2.2 Fuel Cell reactants selection and storage architecture

The trade-off for the propellants to be used for the fuel cell power generation is based on two aspects: first, to maximize the power generation of the propellants per mass required; the second aspect is the consideration about the thermal environment of the tanks during the mission.

The boiling of the propellants has been assessed with respect to the saturation pressure of the propellant. The main propellants chosen were Oxygen with Hydrogen, Methane, Ethane, Propane and Butane. The saturation temperature range for the propellants is increasing in relation to the number of carbons in the chain (hydrogen being at the lowest value in terms of temperature). On the other hand, the number of elements is also decreasing the maximum power possible for the use in a fuel cell (hydrogen has the highest potential power output in this case).

By comparing the possible temperature ranges and as well the potential power generation for the propellants, the decision was taken to choose methane due to the following factors:

- the higher temperature range for the storage (see Figure 13), as well as having mostly the same temperature range as Oxygen,
- the high potential power output for the fuel cell, minimizing the required mass for the hibernation phase.

In order to estimate the amount of propellant need for the fuel cell operations, the following basic assumptions were taken into account:

- Energy Density values provided are based on Enthalpy of Formation at standard conditions (1Bar, 25C)
- Calculated values are adiabatic and stoichiometric.
- No efficiency factor is considered in the cycle on top of efficiencies described in 7.3.5.2.1 and no process\reaction steps are considered.
- Cryogenic candidates are self-pressurized and stored at conditions along the saturation line.

The operation requirements for the FC system are two:

- To generate a total energy of **9.4 GJ** for Heat and Electrical Power (beyond Saturn). The estimated mass of propellants required is around 770 kg of LOX and 178 kg of Liquid Methane (without margins).

- In addition to the generation of energy, a set of boil-off thrusters (Figure 11) have been included in order to provide Delta-V (small delta-v amount) and AOCS propulsion capabilities for the last phase of the mission when the bi-propellant subsystem on-board is no longer operational (detailed discussion in the needs of the boil-off thrusters provided in section 7.3.7).

The architecture for the fluidic chain of the Fuel Cell subsystem is shown in Figure 11.

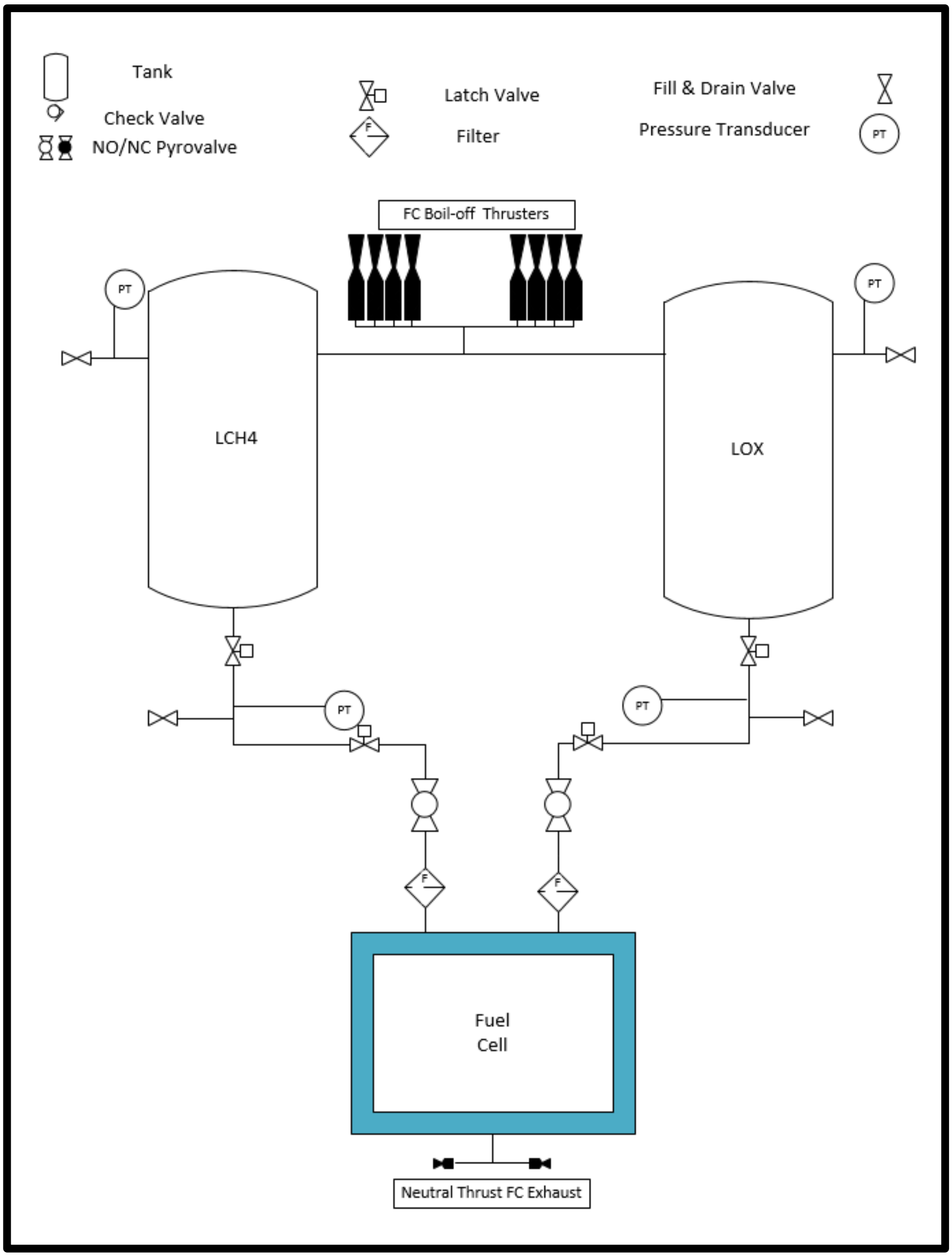


Figure 11: Flow Schematic for FC System

7.3.5.3 Battery & PCDU

In the absence of significant eclipses, the battery sizing case comes from the motivation to minimize SOFC reactant usage. It is the energy difference between the solar generation at 11 A.U. and the energy demand of the peak periods during the cruise phase (1 hour PCIM, followed by 36 hours COM in Figure 5) at 7.3 years after launch. Using the battery for this period delays the start of SOFC operations as far as possible. The battery is sized assuming SAFT VES16 (a current qualified technology) for a total of 8092 Wh.

The mass and dimensions of the PCDU are estimated using TERMA Modular Medium Power Unit as a basic guide. However, the PDCU will be a new development, requiring the functionality to interface with the fuel cell system as well as solar array and battery. The PCDU is also required to have very low quiescent power consumption. A consumption of 20W has been assumed, where 30 to 50W would be considered reasonable for a PCDU of this functionality. In line with JUICE, a power system topology of fully regulated bus with MPPT solar power regulation is baselined.

7.3.6 Thermal Subsystem

7.3.6.1 Fuel Cell Tank Thermal Concept

The fuel cell cryogenic reactants need to be stored for a long duration in tanks during the transfer from Earth to Uranus with minimal mass loss due to boil off. A previous study (RD[10]) investigated a zero boil off (ZBO) concept to maintain hydrogen and oxygen fuel tanks at reasonable pressures using passive thermal cooling. This cooling method included the use of shades, low conductivity supports and spacecraft pointing constraints. For low altitudes over the warmer inner planets, the view factor to the planet can result in significant heat loads to the tanks due to planetary infrared and albedo fluxes so a large cone is needed as a shield. A conical shield surrounds the tanks to limit the exposure to environmental fluxes while the tanks are connected to a radiator that sees cold space.

The initial analysis considered that the methane and oxygen tanks were cylindrical and mounted side by side as shown on the left side of Figure 12. This means the radiators required to passively cool each tank can be designed independently as the oxygen and methane tanks have different boiling points. The shape also means the tanks can have a limited view factor to the hot side of the shield. However, structural and accomodation concerns mean that two spherical tanks mounted directly above a cylindrical core structure are preferred. This concept is shown on the right hand side of Figure 12. The oxygen tank which has the lower boil off temperature is mounted above the methane tank and directly radiates to space. The methane tank is then cooled by a conductive connection to the oxygen tank. This has the disadvantage of coupling the thermal design of the tanks and requires a taller thermal shield.

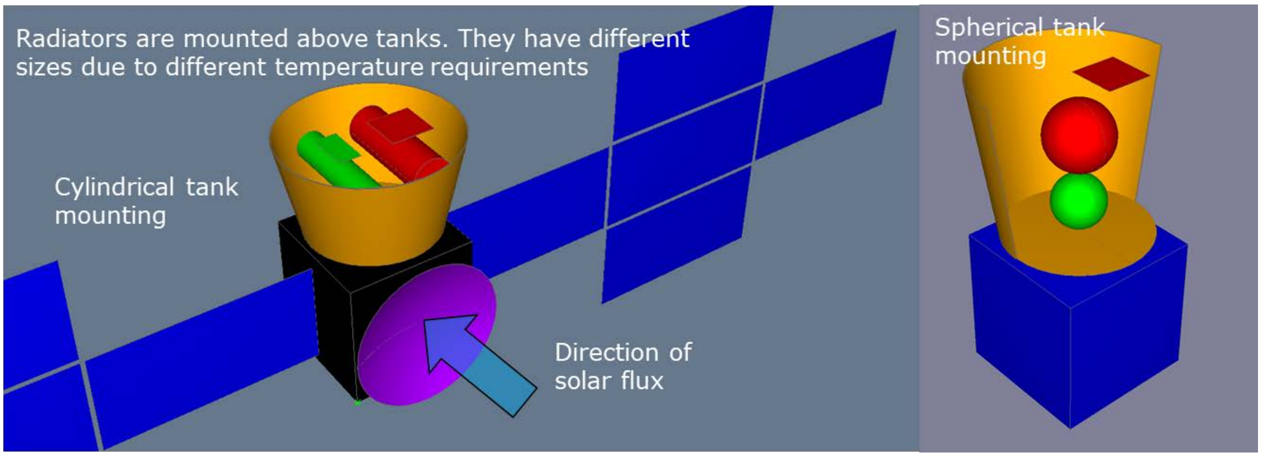


Figure 12: Thermal shielding for fuel cell tanks

In order to supply the reactants to the fuel cell some pressure is needed in the tanks. The pressure versus temperature curves for methane and oxygen are shown in Figure 13. In order to maintain a stable pressure the radiators are slightly oversized and heater power is applied to the tanks in order to achieve the required temperatures and pressure. The actual supply pressure required is unknown but two pressures were assessed, 1 bar and 10 bar. For the 10 bar case, the temperature of the oxygen tank is controlled between -153°C and -150°C and the methane tank is maintained between -123°C and -120°C. For the lower pressure case, the oxygen tank is controlled between -183°C and -180°C while the methane tank is maintained between -163°C and -160°C.

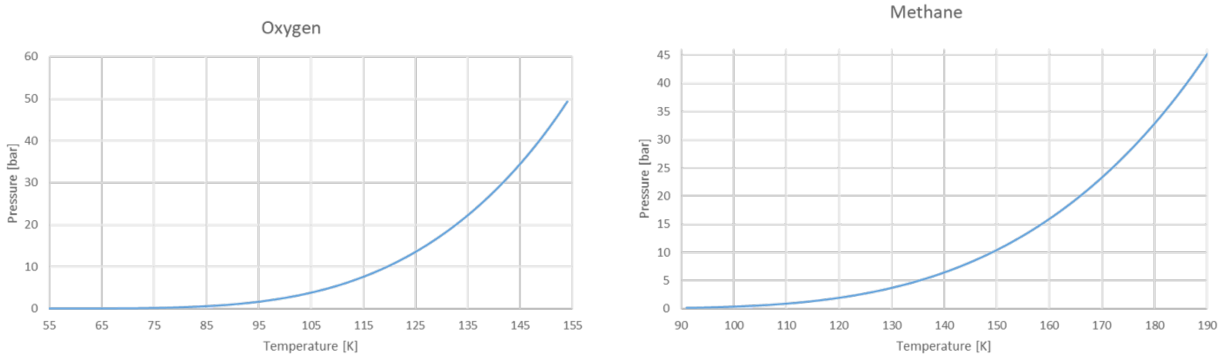


Figure 13: Pressure versus temperature curves for methane and oxygen

To predict the temperature of the shield and the tanks the EVEEJU31 transfer profile was assumed. The spacecraft is assumed to maintain an orientation so that the solar flux comes from one side as illustrated in Figure 12. The absorbed solar flux was applied to the sun facing side of the shield. The shield is also assumed to have a blanket of MLI to reduce the heat load. The tanks are also assumed to be wrapped in MLI. For the methane tank, the radiator size was set as 0.5 m² and for the oxygen tank 1 m². For the MLI on the shield, the performance values were based on the JUICE Helpac blanket.

The temperature predicted for the shield outer surface and for the tanks for both the 10 bar and 1 bar case is shown in Figure 14. The required heater power is highest for the 10 bar case and is predicted to be 12.5 W, this heater power was used for sizing the power system.

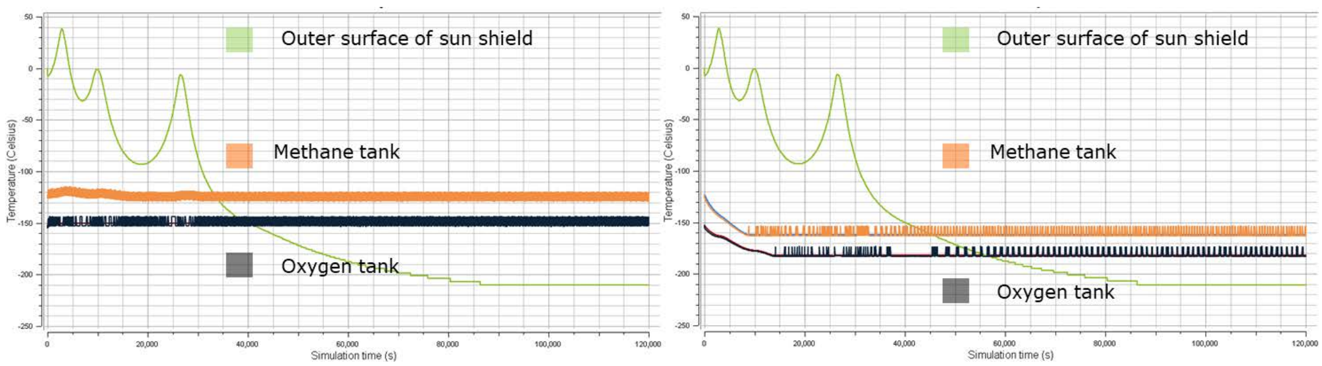


Figure 14: Temperature profile of shield and tanks considering a target of 1 bar and 10 bar respectively

7.3.6.2 Carrier Thermal concept

For the EPIG carrier spacecraft, it is assumed that the avionics and the battery will be connected to separate radiators. For the avionics, the heater power needed to maintain the units above their lower temperature limit was lower than the heat dissipated in the minimum power mode. So no additional heater power is needed in this mode.

While the fuel cell is employed, it will supply varying amounts of power depending on which mode the spacecraft is with communications mode being highest and the hibernation mode being lowest. As the fuel cell is estimated to be 50% efficient, it will generate the same amount of heat as power. Therefore, in order to keep the radiator stable between the lowest and highest power case, a louvered radiator can be employed.

It is assumed that the fuel cell needs to be maintained at 700°C (operational mode) to ensure optimal performance. In order to maintain this temperature over variable power ranges some variable thermal coupling between the radiator and the cell is necessary.

7.3.7 Propulsion

The baseline is a bi-propellant propulsion subsystem, this system is used to perform all delta-v maneuvers up to Jupiter, the final deflection maneuver at Uranus arrival (~10 m/s) and AOCS related control during Uranus operations is performed with boil-off thrusters associated to the Fuel Cell Subsystem (see Figure 11). The reason not to maintain the use of the bi-propellant subsystem up to Uranus is the required heater power to maintain the MMH and MON propellant tanks at the required temperature for almost 2.5 years of mission; it would translate into a very large fuel cell consumables penalty. The disadvantage is the need to mature the technology for the boil-off thrusters. A feasible alternative with higher maturity would be the use of a cold gas system for the Uranus operations with significant less temperature maintenance demands.

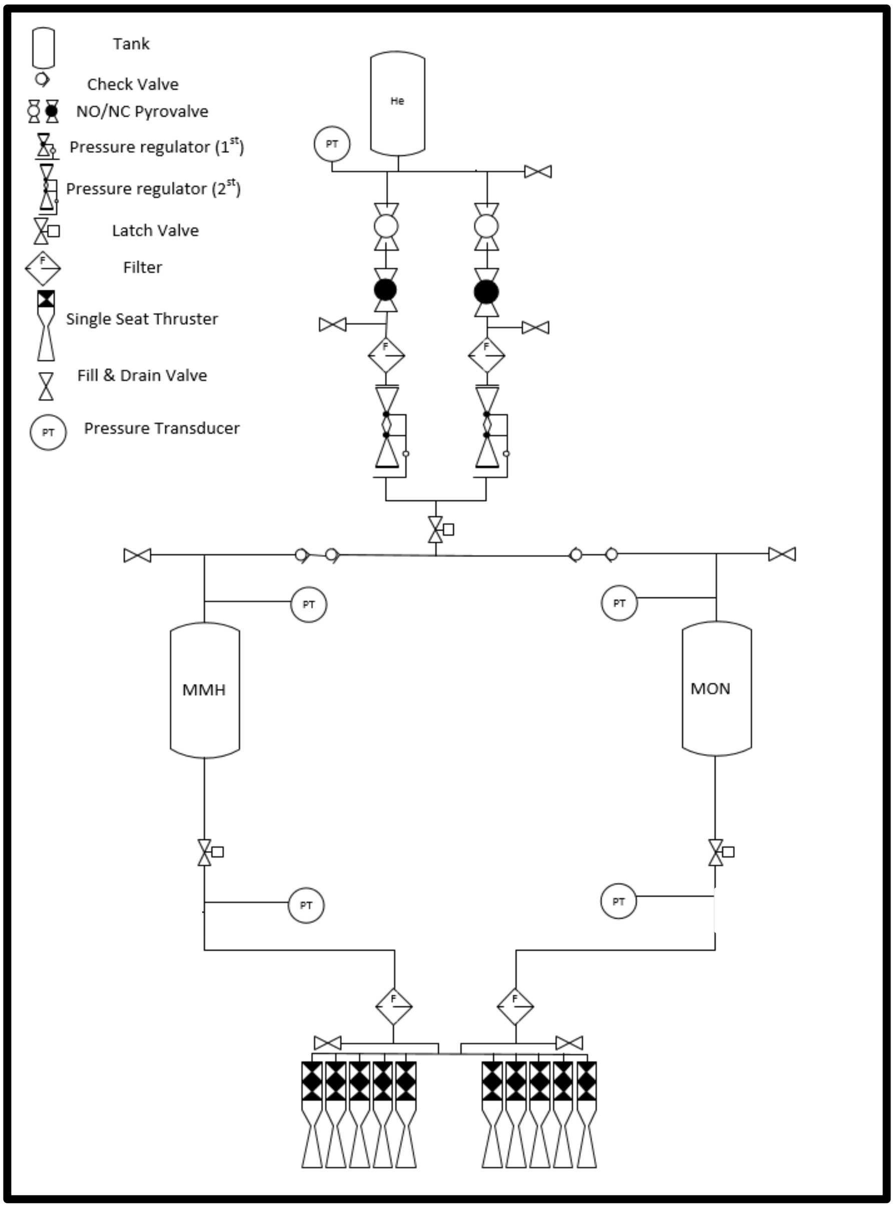


Figure 15: Example Flow Schematic Bi-Propellant Subsystem applicable to EPIG

7.3.8 AOCS

7.3.8.1 General Attitude Concept

The defining constraint for this mission is power - due to the avoidance of RTG and the great distances from the Sun. It is therefore of interest to consider passive attitude control such as a spin-stabilized concept where active attitude control is only required for re-orientation of the spin-axis for delta-V manoeuvres or for Earth communications. Sun-tracking could be achieved passively at other times via introducing dihedral tilt of the solar arrays to form a conical-like shape that has natural stability properties. Slight asymmetric adjustments of solar panel rotation angle about its main axis can also be used to adjust spin rate via the windmill effect. In the outer solar system, a passive attitude concept is necessary for the bulk of the journey simply because continuous powering of sensors and actuators for attitude control is not considered feasible.

However, the fuel cost to re-orient the spin axis (or de-spin, slew and re-spin) is high and outweighs savings from the use of passive control during attitude hold periods. Spin-stabilization is a more favorable solution when the spin-axis does not have to be (frequently) actively re-oriented. Furthermore, the Earth-communication pointing requirement of 0.1 deg in the outer solar system would require extremely stringent mass balancing requirements in order to limit coning motion for a spinning s/c.

For these reasons it was chosen to baseline a hybrid concept with 3-axis stabilization at all times except during hibernation periods (between Earth communications) in the outer solar system. The trade-off is shown in Table 9.

	3-axis stabilisation	**Spin-stabilisation**
Pros	Increased Sun-tracking and Delta-V directional accuracy	Sun-tracking can be achieved passively (with minimal or no fuel cost) via tilting of solar panels to stabilise spin-axis along Sun-line Delta-V burns can be conducted without active attitude control (after aligning spin axis with desired burn vector)
Cons	Frequent power required to operate thrusters and potentially CATbed heaters	Spin-stabilisation implies fixed SADM angle, which means potential power issues in inner solar system during communication periods. Fuel cost high for slewing spin-axis over large angles (slewing for Earth communications or Delta-V burns) Earth-communications pointing in outer solar system may not be possible due to coning motion.

Table 9: Attitude concept trade-off

7.3.8.2 Hibernation Attitude

To minimize power consumption during hibernation periods, spin-stabilization is preferred as discussed in the previous section. However, a spin-stabilized spacecraft will have a fixed spin-axis in the inertial frame. Since Sun and Earth pointing is desired for power and communications, this requires regular shifts of the spin axis.

For the Rosetta mission, the possibility to maintain Sun pointing of the spin axis via passive means was investigated. Adding dihedral tilt to the solar panels and keeping the cog forward (w.r.t. Sun) of the center of pressure the spin-axis will continuously shift toward the Sun with a steady state Sun tracking error less than 10 deg (for the parameters assumed). This is due to the solar pressure torque acting as a stabilizer.

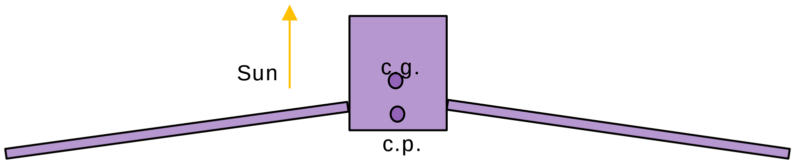


Figure 16: Passive sun-stabilization concept

However, the mission is already budgeting spin-down and 3-axis re-pointing to the Earth every 6-months for communications. Since the Sun is very near to the Earth (< 10 deg), due to relative geometry for S/C in the outer solar system, the Sun tracking is gained for free every 6 months. The drift of the Sun direction relative to the S/C over 6 months has not been determined exactly but is expected to be in the range 6-14 deg. Therefore, there is no strong need to actively slew the spin axis toward the Sun during the 6-month hibernations.

Since the fuel saving of the solar-pressure-stable concept is minimal (due to the operations desire to have 6 monthly communication events) and since it may require a 2 D.O.F. SADM to achieve dihedral solar array tilt (for Sun tracking accuracy <10 deg), the baseline is no dihedral tilt and no reliance on passive Sun tracking.

The baseline spin rate is selected at 0.3 deg/s, this value is more than sufficient to give negligible nutation with the < 10^{-5} Nm solar pressure torques and ~35 000 kg.m^2 spin-axis (X) inertia. The majority of spin 'wobble' will come from coning due to mass imbalance. Higher spin rates may be possible but are not considered necessary.

Monthly rate checks and corrections may still be needed to ensure that the windmill effect does not drive the spin rate too far from target. Windmill torque about the spin-axis is a result of SADM asymmetric alignment errors about their primary rotation axis that causes the spinning s/c to resemble a windmill. This torque could also be used to control the spin rate in a purely passive concept via small regular SADM angle adjustments.

7.3.8.3 AOCS architecture

The functional AOCS needs for the carrier s/c led to the selection of the following AOCS modes:

1. Acquisition & safe mode
2. Inertial pointing mode: used during Sun tracking inner solar system, probe ejection, probe pointing and Delta-V attitude acquisition
3. Hibernation mode: during outer solar system cruise
4. Orbit control mode: used during Delta-V manoeuvres
5. Survival mode: backup safe mode using independent hardware from nominal modes

The use of equipment per mode is reflected in Table 10.

Mode	Acquisition & safe mode				Survival mode				Inertial pointing mode		Hibernation mode		Orbit Control mode
Sub-mode	Rate damp	Earth & Sun acquisition	Earth & Sun point	Cold sky search	Rate damp	Earth search	Earth & Sun acquisition	Earth & Sun point	Attitude acquisition	Attitude hold	Rate control	Idle	Delta-V
Sun sensor						X	X	X					
Coarse rate sensor				X	X	X	X						
Medium Gain Antenna						X	X	X					
Star tracker	X	X	X	X					X	X	X		X
Accelerometer													X
RCS	X	X	X	X	X	X	X	X	X	X	X		X

Table 10: AOCS equipment usage per mode and sub-mode

7.3.9 Communications

The basic architecture of the TT&C subsystems is the following:

- X-band considered for Carrier communications with Ground with 35 W or 65 W TWTA. HGA is necessary to communicate from Uranus (19 AU) to ESTRACK ground stations during the nominal mode.
- Two LGAs accommodated on two opposite sides of the S/C are necessary for the LEOP phase and for the contingency recovery.
- The addition of a fixed MGA (preliminary chosen as 16 cm diameter horn) has been considered and baselined due to increased robustness with a low penalty in mass. The reasons are:
  - Possibility to receive low rate telemetry from Uranus with 70m DSN G/S, or during cruise via ESTRACK G/S array. In both cases with reduced pointing accuracies (that allow saving propellant mass).
  - Emergency communications via ESTRACK G/S with reduced rates (down to 20 bps) up to 8 AU.
- The communication strategy with the probe considered is also via HGA, making a dual-band X and UHF dish antenna. This solution allows having a high-gain antenna in UHF without a dedicated dish, only the addition of a UHF feed is needed.

The baseline design of the communication subsystem foresees an architecture as shown in Figure 17.

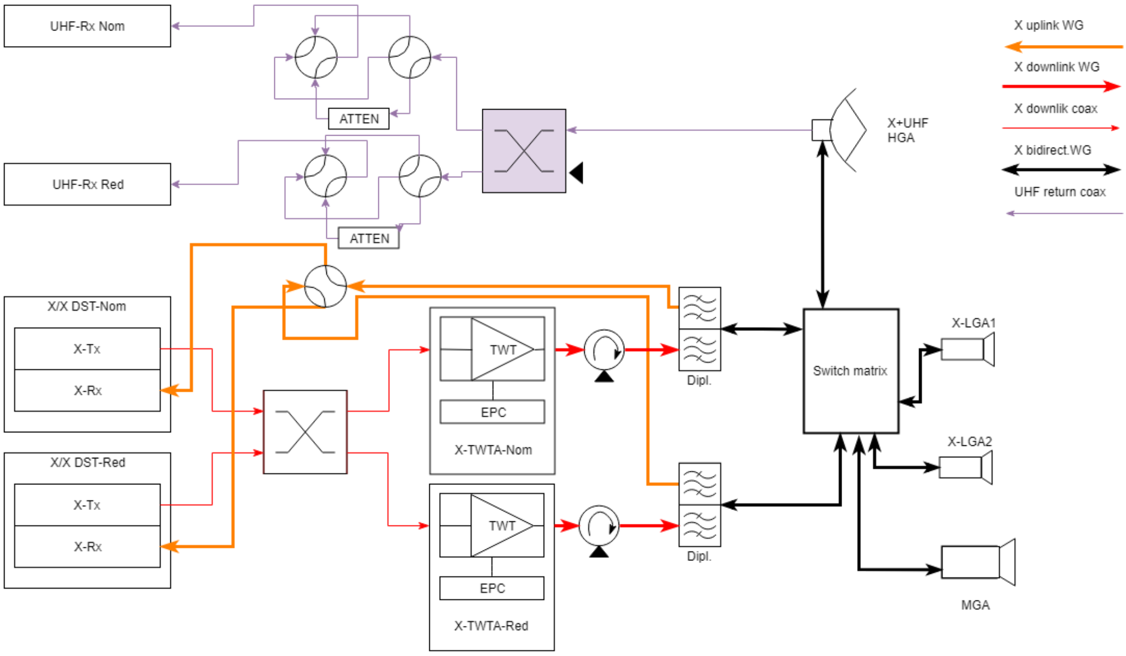


Figure 17: Baseline design for the carrier communication subsystem

7.4 Technology development summary for fuel cell concept based on EPIG Uranus mission proposed architecture

Equipment Name & Text Reference	Technology	Additional Information
Solid Oxide Fuel Cell	Space SOFC for CH_4-O_2 reactants	Small solid oxide fuel cell with low power delivery (< 150 W) suitable for multi-year reliable space operation with CH4 and LOX, investigating throttleability, re-startability (thermal cycling), and counter-flow heat exchange from output to input flow. Also low quiescent power consumption for e.g. control and monitoring.
Solar Array	LILT Solar Array at Saturn (−150°C) or Uranus (−200°C)temperatures	The JUICE LILT Solar Array should be applicable, but so far not tested below 150°C. Suitability needs confirmed, and performance has uncertainty until measured.
PCDU	Low power consumption PCDU including Fuel Cell interface capability	PDCU that interfaces with multiple FCs (possibly through a common controller) as well as the usual SA and battery interfaces. Must be LOW quiescent power consumer.
Long Term Cryogenic Tanks	Tanks capable to store and boil-off cryogenic propellants for the duration of the mission.	See CDF Study CDF-47(A) July 2006
Long Term Cryogenic propulsion equipment	Pipes, Valves, Regulators and any other wetted components capable of withstanding long term cryogenic exposure	See CDF Study CDF-47(A) July 2006
Boil-off thrusters	Cold or hot gas thruster capable of pulsing being fed from the cryogenic boil-offs	LOX/L Methane high thrust non-pulsable engines exists when fed at standard conditions (blowdown/pressure fed)
Thrust Neutral Exhaust	FC system exhaust should not generate thrust	Most cold gas technologies should be able to cope with this requirements but dedicated development is required.

Table 11: Technology developments associated to fuel cell usage

8 BASELINE ARCHITECTURE SATURN MISSION

The mission architecture for the Saturn option is very similar to the Uranus case with the exception of the absence of need for an alternative energy source that simplifies the spacecraft design significantly. The power provided by the solar array (same size as Uranus design) is sufficient to provide power during the entire mission without the need for a fuel cell subsystem or the need for dedicated Hibernation Mode implementation. The Saturn mission relies on mature technology for the Carrier Spacecraft.

8.1 Transfer options to Saturn

As for the Uranus case, a number of interplanetary transfers were considered; an overview of these is presented in Appendix A. The transfer EEEJS32 was selected as baseline, due to its high available wet mass, low Δv and intermediate probe arrival conditions (relative arrival velocity less than 26 km/s). The back-up trajectory selected corresponds to EEEJS34, details are provided in Figure 19 and Figure 20.

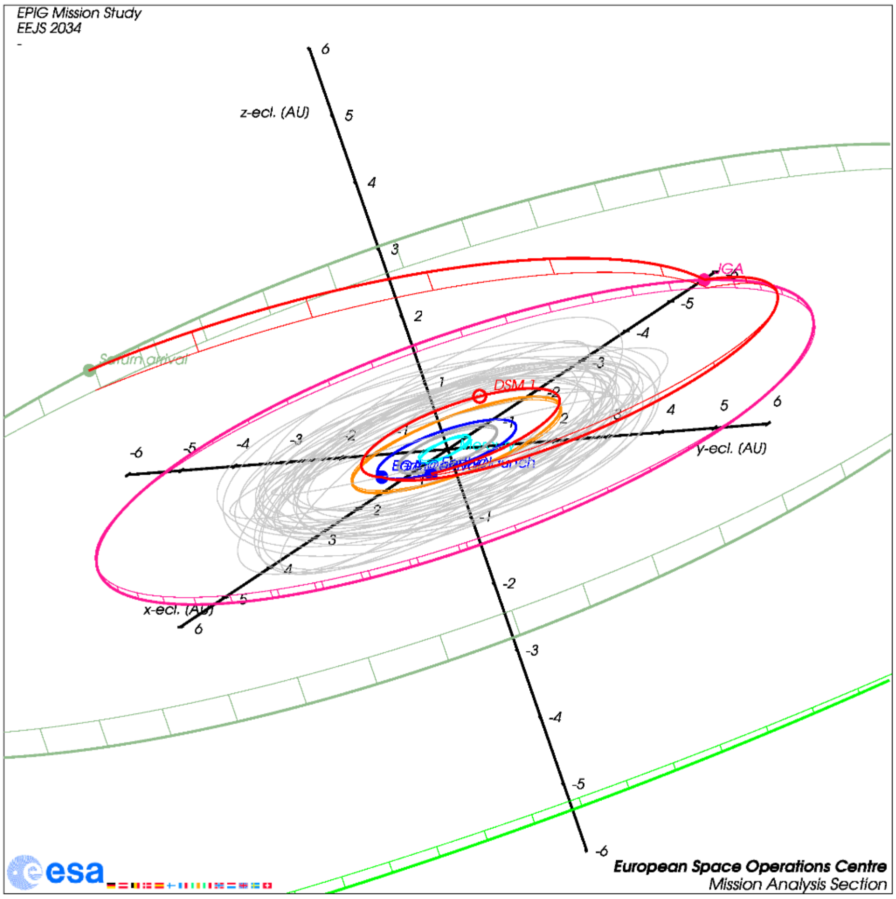
Mission architecture		
Launcher	Ariane 6.4	
Launch Date	Baseline	19/10/2032
	Backup	19/10/2034
Trajectory	Baseline	EEEJS32
	Backup	EEJS34

For the Saturn mission, out of the variety of possible options, the chosen baseline and backup are very similar. Transfer scenarios with moderate Earth escape velocities require a complex sequence of swing-by's that led either to a very long transfer duration or a high Saturn arrival velocity.

A major concern for the design of the Saturn arrival is the placement of the RPC (Ring Plane Crossing) points for both the probe and the carrier. For the probe ring crossings can be avoided altogether if the entry location is chosen judiciously. The carrier will inevitably cross Saturn's Laplace plane twice. The task is to choose the RPC distances such that they lie within radii of minimal particle flux density.

Figure 18: Composite Image of a Cut-Out of the Saturn Ring System

The approach conserved as baseline is to target RPC below the D-rings. The options are to target RPC in the Cassini division or between the F and G rings, as was done for the Cassini-Huygens mission on the day of SOI, 1 July 2004.



Case	EEEJS32	EEJS34
Launch	2032/10/19+LP (5.1 km/s, 20.3 deg)	2034/10/19+LP (5.11 km/s, 21.2 deg)
DSM 1	2033/11/30 (679 m/s)	2035/10/25 (740 m/s)
ESB 1	2034/8/30 (860 km)	2036/8/23 (338 km)
ESB 2	2036/8/30 (997 km)	-
JSB	2040/3/4 (572000 km)	2040/3/16 (543000 km)
Saturn arrival	2045/7/22 (2.68 km/s)	2045/8/21 (2.66 km/s)

Figure 19: EEJS34 Transfer and EEEJS32+EEJS34 Comparison

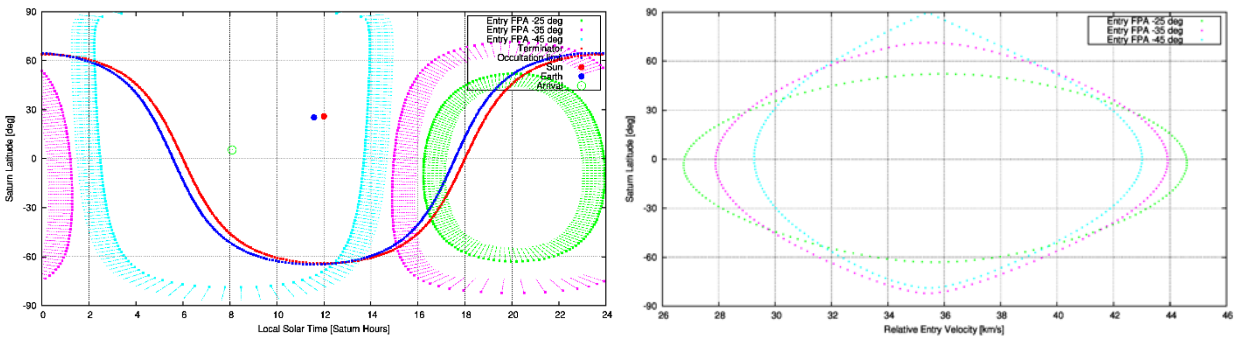


Figure 20: EEJS34 Entry Conditions and Velocity

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8.2 Mission phases

The mission phases are presented in Table 12. It should be noted that for the Saturn option the mission baseline only allows for a 60 min probe descent.

Mission phases	
LEOP	TBD
Interplanetary transfer (total)	12.8 years
Probe coast	5 days
Probe entry+descent	6+60 min
Data relay before EOM	2 days

Table 12: Saturn mission phases

8.2.1 Mission timeline and operational concept

The mission timeline and operational concept for Uranus is presented in Figure 21 and Figure 22.

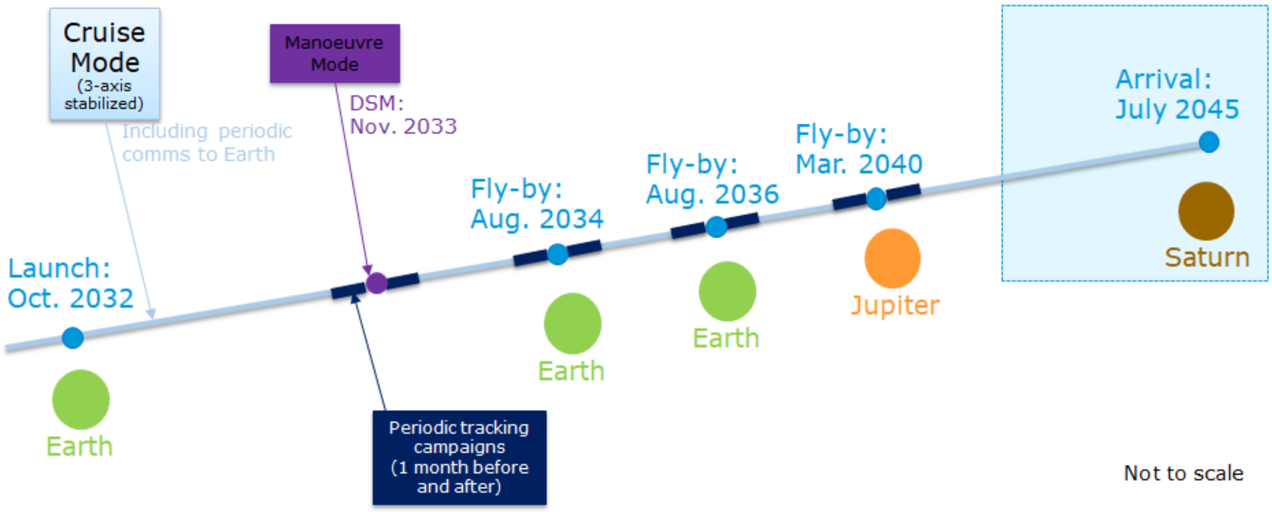


Figure 21: Saturn mission timeline until arrival

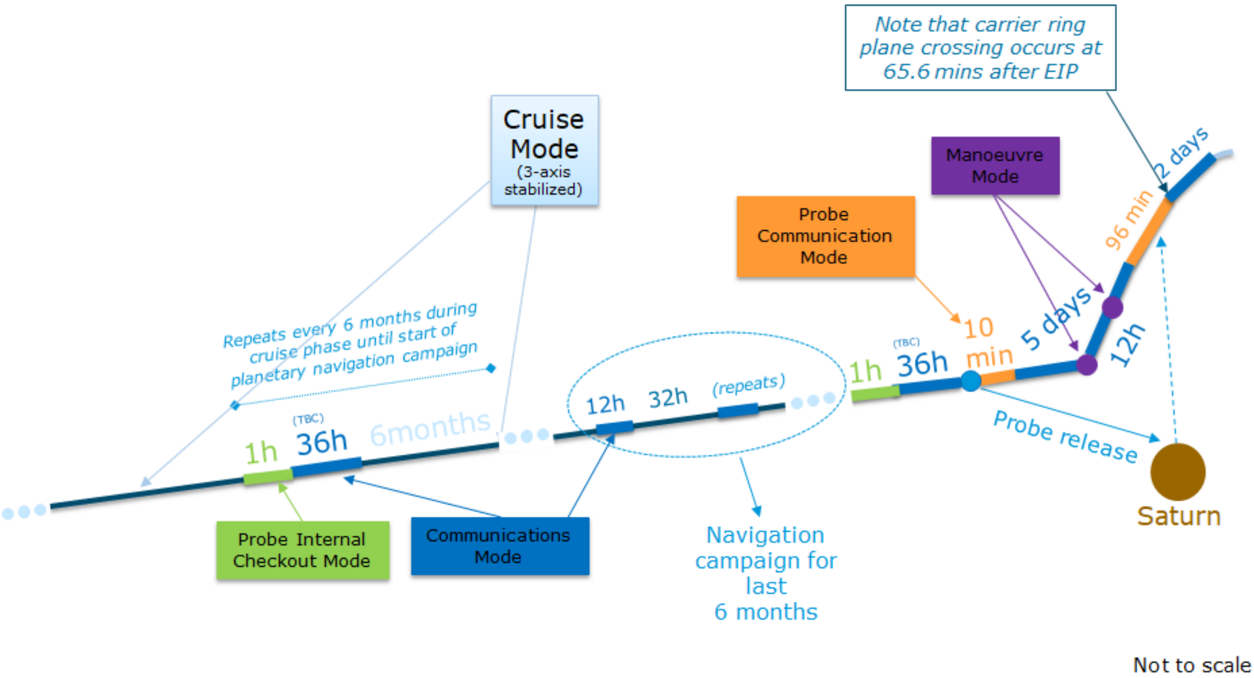


Figure 22: Saturn mission timeline at arrival and probe delivery & descent

8.3 System Baseline Design SATURN

8.3.1 Baseline overview

Carrier Saturn		
Mass (kg) (incl. 20% system margin)	Dry mass (incl. 420kg probe):	1971 kg
	Propellant mass:	794 kg
	Wet mass:	2765 kg
Payload	420 kg Planetary probe	
Propulsion	10x Bi-prop thrusters 4x propellant tanks + pressure tank	
AOCS	2x accelerometers 2x star-trackers 6x sun-sensors for survival mode 2x Gyros for safe mode only	
Communications	1xHGA, 2xLGA and 1xMGA X-band uplink/downlink: 13 kbps@Saturn UHF for probe downlink: 2 kbps	

Power	120m2 Solar arrays (2 wings) 18kg battery
Data Handling	CDMU RTU
Structures	270 kg
Thermal	Heaters, MLI, Radiators 78W for propulsion system
Mechanisms	Probe deployment mechanism SADM
Radiation Shielding	51kg (TBC)
Ring-crossing shielding	2x 18kg whipple shields

Table 13: Saturn carrier overview

As previously mentioned, subsystems such as Propulsion, AOCS, Communications, Data Handling and Mechanisms can be considered almost identical to the Uranus case. The power subsystem relies in Solar Array, battery and PCDU without the need of SOFC subsystem, which also allows reducing the size of the carrier spacecraft and reducing the structural dry mass needs.

8.3.2 System budgets

The mass budget for the EPIG spacecraft (Saturn option) is provided in Table 14. The detailed mass breakdown considers only the carrier. All dry masses include maturity margin and then an additional 20% system margin applied to the total dry mass of the spacecraft. The harness mass has been accounted as 5% of total dry mass of the system. For the planetary entry probe, 420kg are added on top of the dry mass including system margin.

No fuel cell subsystem is required for the Saturn design so the wet mass accounts for the chemical propulsion system. The adapter mass is based on a standard Ariane 6 launch adapter.

S/C Mass Budget		Mass [kg]
Attitude, Orbit, Guidance, Navigation Control		4.8
Communications		62.8
Chemical Propulsion		96.0
Data-Handling		34.1
Mechanisms*		66.0
Power		499.5
Structures		270.9
System Engineering		105.1
Thermal Control		59.6
Harness	5%	59.9
Dry Mass w/o System Margin		**1258.8**
System Margin	20%	251.8
Dry Mass incl. System Margin		**1510.5**
Probe		420
Dry Mass incl. Probe		**1930.5**
CPROP Fuel Mass		298.3
CPROP Fuel Margin	0%	
CPROP Oxidizer Mass		480.3
CPROP Oxidizer Margin	0%	
CPROP Pressurant Mass		5.6
CPROP Pressurant Margin	0%	
Total Wet Mass		**2714.7**
Launcher Adapter		115.0
Wet Mass + Adapter		**2829.7**
Target Wet Mass incl. Adapter		3384.0
Below Target Mass by		554.3

***Probe separation mechanisms included in the budget**

Table 14: EPIG Saturn Mass Budget

The mass budget for the EPIG spacecraft is provided in Table 15. The definition of the modes is equivalent to the description provided in 7.2.1; there is no need for Hibernation Mode in the Saturn case.

Row Labels	P_mean COM	SAM	PICM	PCOM	CRM_SAT	MAM_SAT
SC (Spacecraft)	281.2	207.6	193.5	192.9	220.5	425.8
AOGNC	0.8	5.8	0.0	0.8	0.8	1.2
ACC_AIQ (Accelerometer Innalabs AIQ2030)	0.0	0.0	0.0	0.0	0.0	0.2
ACC_AIQ_2030_2 (Accelerometer Innalabs AIQ 2030_2)	0.0	0.0	0.0	0.0	0.0	0.2
STR_TermaOH (STR Terma T1 Optical Head)	0.4	0.4	0.0	0.4	0.4	0.4
STR_TermaOH_2 (STR Terma T1 Optical Head #2)	0.4	0.4	0.0	0.4	0.4	0.4
GYRO_Sireus_1 (GYRO SireusNG10 TAS-UK #1)	0.0	2.5	0.0	0.0	0.0	0.0
GYRO_Sireus_2 (GYRO SireusNG10 TAS-UK #2)	0.0	2.5	0.0	0.0	0.0	0.0
COM	117.3	38.7	30.0	60.0	30.0	30.0
HPA_TWTA_X_BEPI_1 (High Power Amplifier TWTA Bepi X #1)	33.7	3.4	0.0	0.0	0.0	0.0
HPA_TWTA_X_BEPI_2 (High Power Amplifier TWTA Bepi X #2)	33.7	3.4	0.0	0.0	0.0	0.0
RX_1 (Receiver #1)	0.0	0.0	0.0	15.0	0.0	0.0
RX_2 (Receiver #2)	0.0	0.0	0.0	15.0	0.0	0.0
XPND_TASI_XX_DST_1 (Transponder TASI XX DST #1)	25.0	16.0	15.0	15.0	15.0	15.0
XPND_TASI_XX_DST_2 (Transponder TASI XX DST #2)	25.0	16.0	15.0	15.0	15.0	15.0
CPROP	0.0	0.0	0.0	0.0	3.5	231.5
DH	29.1	29.1	29.1	29.1	29.1	29.1
CDMU_1 (Command and Data Management Unit #1)	20.0	20.0	20.0	20.0	20.0	20.0
RTU (Remote Terminal Unit)	9.1	9.1	9.1	9.1	9.1	9.1
INS	0.0	0.0	31.4	0.0	3.1	0.0
Probe (Probe)	0.0	0.0	31.4	0.0	3.1	0.0
MEC	0.0	0.0	0.0	0.0	20.0	0.0
SADM_Saturn (Solar Array Drive Mechanism 1dof Saturn [incl S/	0.0	0.0	0.0	0.0	20.0	0.0
PWR	20.0	20.0	20.0	20.0	20.0	20.0
PCDU_SAT (Power Conditioning & Distribution Unit Saturn)	20.0	20.0	20.0	20.0	20.0	20.0
TC	114.0	114.0	83.0	83.0	114.0	114.0
HTR_BATT_1 (Heater battery #1)	5.0	5.0	5.0	5.0	5.0	5.0
HTR_PRB (Heater Probe)	31.0	31.0	0.0	0.0	31.0	31.0
HTR_CARR_PROP (Heater Carrier Prop)	78.0	78.0	78.0	78.0	78.0	78.0
Grand Total	**281**	**208**	**193**	**193**	**220**	**426**
Total with 20% margin	**337**	**249**	**232**	**231**	**265**	**511**

Table 15: EPIG Saturn power budget

8.3.3 Baseline configuration for Saturn mission

The spacecraft design of the Saturn carrier is derived from the design of the Uranus carrier; nevertheless, the Saturn carrier will not be equipped with a fuel cell subsystem as the Uranus carrier. The planetary probe of 420 kg can then be accommodated on the top panel to have symmetrical spacecraft configuration. The probe is directly supported to the central tube by means of trusses. The final spacecraft design of Saturn option is illustrated below:

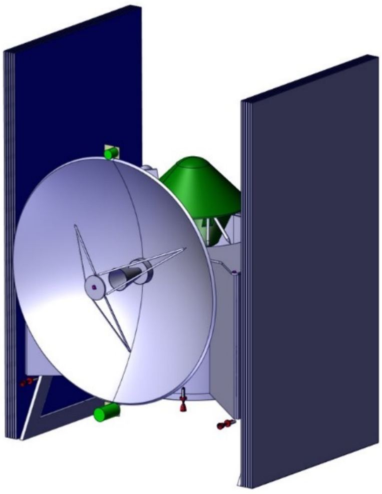


Figure 23: EPIG Saturn configuration

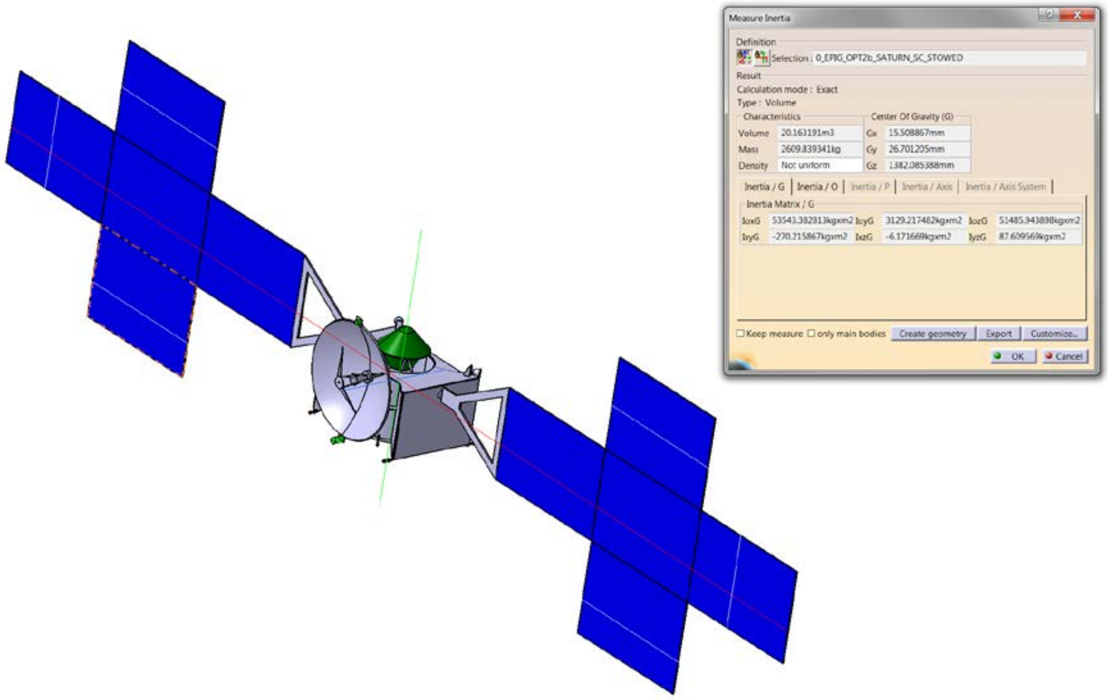


Figure 24: EPIG Saturn deployed configuration

9 COMMON PROBE CONCEPT

9.1 Probe overview

The probe design was almost identical to the one presented in the CDF Ice Giants study (RD[1]). An overview of the CDF Ice Giants probe design is included in Table 16. Note that the probe design was slightly customized for EPIG as follows:

- A power and electrical data connection was included between the probe to provide all energy and communications required for thermal control, probe checkouts, etc.
- No RHUs were included
- Minor modifications in probe thermal concept to account for absence of RHUs.
- The total probe mass was scaled up from 342 kg to 420 kg. This increase in allocation accounts for the following two aspects:
  - Additional mass reserve to address the observation from the CDF Ice Giants study (RD[1]) that the available probe payload mass is likely too low to meet the science requirements. Scaling the payload mass would have a direct consequence on the pressure vessel and TPS sizing.
  - Finally, the Saturn probe would likely experience higher atmospheric entry velocities, and as such require a heavier TPS. Based on a previous CDF study (RD[1]), this may lead to a TPS for Saturn at least 20% heavier than for Uranus.

Keeping the total probe mass constant for both mission options was considered beneficial from a comparison perspective.

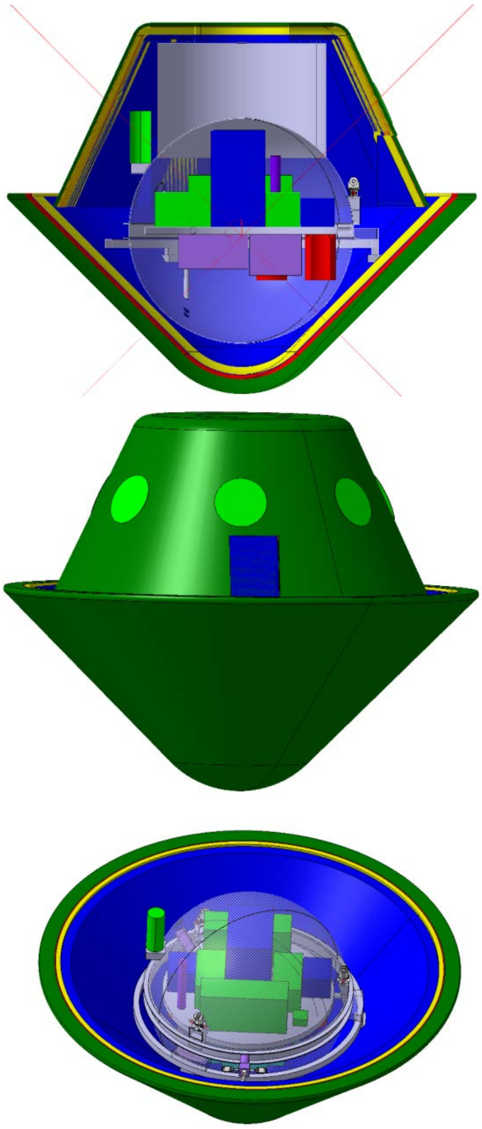
Probe		
Mass (Incl 20% system margin)	Mass w/o TPS&TC: 191 kg TPS&TC: 151 kg Mass with TPS: 342 kg	
Ballistic coefficient	Projected area: 1.43 m^2 (diameter 1.35 m) Front shield area: 1.99 m^2 Cd: 1.07 BC: 228 kg/m^2	
Payload	• Atmospheric Structure Instrument • Camera-Radiometer • Mass Spectrometer • Photometer • USO-Doppler	
EDS	2 subsonic parachutes: pilot (M=0.8) and main	
TPS	Front shield: 51.9 mm thickness, 129 kg Back shield: 31.4 mm, 19.9 kg	
GNC	2 redundant IMUs 2 parachute deployment switches	
Mechanisms	Back and Front Shell Separation Mechanisms Parachute Swivel Mechanism Mortar parachute pyro cutter Spin Eject Mechanism [Probe side]	
Communications	UHF redundant chain Patch antenna on the backshell Helix antenna during descent	
Power	4 x 3 kg batteries PCDU	
Data Handling	CDMU including timer	
Structures	61.3 kg of structures (of which 28.5kg are in the descent module)	
Thermal	31 RHUs, MLIs, Front shield radiator, pressure vessel insulation	

Table 16: Overview of CDF Ice Giants probe (from RD[1])

In addition, the probe descent profile was modified in order to account for the shorter coast phase (5 days instead of the 20 days baselined in RD[1]). As such, the nominal operations profile of the probe is as follows:

Event / phase	Timing	Comment
Probe released from carrier	5 days before planetary arrival	7 days may allow more margin from an operational perspective to allow for contingency operations in the event of failures (particularly of manoeuvres)
Checkout 1	10 min directly after probe release	To check system health. Note that it was identified during the CDF Study that such a checkout would not be possible with the current design, given that the carrier cannot slew to point the HGA at the probe within such a timeframe. Options to address this in future work could include: • Delaying the checkout until the carrier could slew towards the probe. This would likely take approximately 1.5 hours. • Including an antenna to receive the probe UHF signal on the same carrier face as the probe. • Removing Checkout 1 (as there is no way to telecommand the probe should issues be identified)
Coast of probe to planet	From end of Checkout 1 until just before atmospheric entry	Most units OFF (except wake-up timers)
Checkout 2	30 min before arrival	Short health check and to calibrate GNC
Entry	~6 min (for Uranus, Saturn TBC)	
Descent	60-90 minutes from 1 to 10 bars	Note that the 90 min here refers to the science requirement, to which the parachute for the Ice Giants probe was designed. Due to the carrier velocity (and for Saturn, the ring constraints) the actual observability from the carrier is limited to 75 min for Uranus, and to ca. 65 min for Saturn.

Table 17: Probe timeline

9.1.1 Modified probe coast phase

Given the constraint that the system should not consider radioisotope sources for heat or power, an early trade-off was performed to estimate the impact on the planned probe coast duration. This phase covers the time from release of the probe from the orbiter, up until the start of its atmospheric entry. In RD[1], this had been set at 20 days. However, in this case the minimum temperatures of the units had been maintained via 31W of RHU power inside the probe. For EPIG, no such RHUs were considered; as such, the temperature of the probe units would decrease significantly during the coast.

As discussed in 9.1.2, a shorter probe coast of 5 days seems to be feasible given the current assumptions for a probe without RHUs. This would lead to the battery reaching its minimal operative temperature (with margins) just before the required activation before entry. Such a strategy would however require either a thermal boost just before probe release, or improvements in e.g. the MLI performance (as discussed later). Feasibility of a probe coast significantly longer than 5 days is considered unlikely.

Alongside the thermal considerations of the probe, two additional criteria had to be assessed:

- Δv required for the carrier trajectory change after probe deployment
- Time available for carrier operations (including maneuvers)

The Δv increased with the inverse of the coast duration. For a 5-day probe coast, the Δv required was thus four times higher than for the 20-day case. Nonetheless, the absolute value was considered low enough such that the impact on the system was considered low.

The carrier must also be operational and perform ground communications and a maneuver after the probe release. 5 days was considered a reasonable amount of time to allow for all these operational needs.

9.1.2 Modified probe thermal concept

The probe design from the Ice Giants study (RD[1]) found that the temperature prediction was very sensitive to variations in MLI performance and internal probe dissipation. In order to reduce this sensitivity, the radiator was sized in order to get a reasonable temperature change when heat dissipation is varied. With this design, 31 W of internal heat is needed in order to maintain the probe above 10°C. This heater power was supplied by RHUs mounted inside the probe. For the EPIG study, the carrier must supply this heat electrically. Upon arrival at Uranus the probe will separate from the carrier and coast until atmospheric entry. During this coast phase there will be no heater power available and the probe will drop in temperature. In the course of the EPIG study, a goal was set of achieving a 5 – 7 day coast without exceeding the probes lower temperature limits.

The thermal prediction of the Ice Giant probe is shown for a 7-day period on left side in Figure 25. After 5 days the payload bay is predicted to be below -50°C and after 7 days it has nearly reached -70°C, this is well below the limits of the payload.

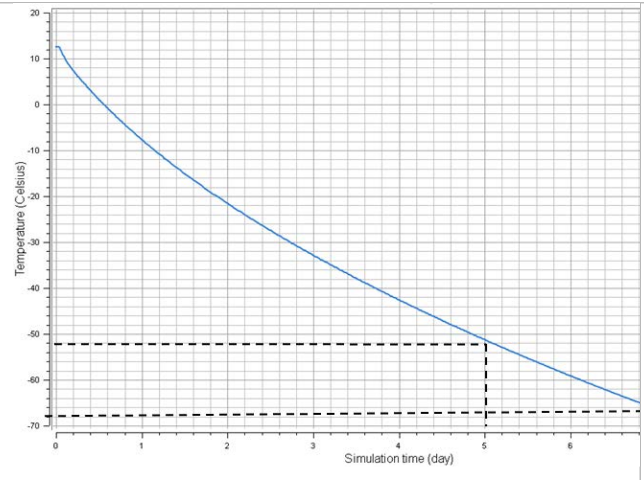
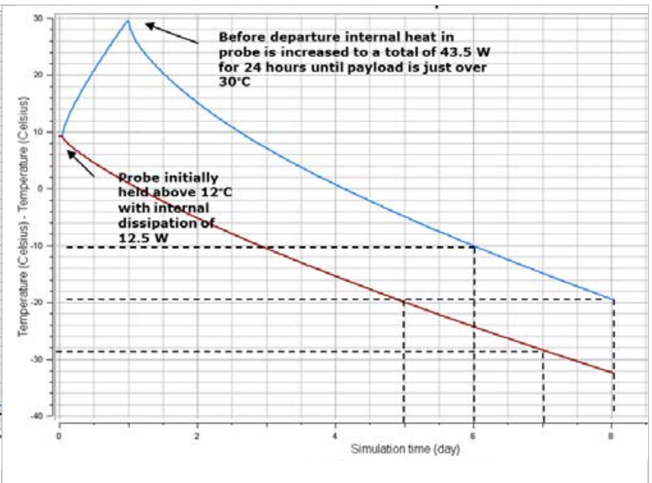
	
Prediction of probe cool-down after release (Ice Giants probe concept)	Prediction of modified probe cool down with and without heat boost

Figure 25: Probe temperature evolution during coasting phase

An assessment of what changes could be achieved in the probe cool down rate by modifying the MLI and radiator properties was carried out. The modified probe MLI is based on the JUICE design, this allows reducing the internal dissipation required to keep the modified probe at 10°C to a value of 12.5 W. The impact of pre-release heat boost on the probe was also assessed; this increased the heat dissipated in the probe payload from 12.5 W to 43.5 W for 24 hours before release. The predicted temperature profile of the modified probe with and without a heating boost is shown in the right side of Figure 25. The heat boost increased the payload bay to 30°C after 24 hours. The boost allows the payload to reach -10°C after 5 days and -20°C after 7 days, both compatible with the required temperature of the avionics.

10 CONCLUSION

- A mission to Saturn (Carrier with deployment of an Entry Probe) based only on Solar Power is considered possible for an Ariane 64 launch.
- A similar mission to Uranus would require the additional use of SOFC system to provide electrical power and heat beyond ~9 AU, requiring some dedicated technology development, but could, based on preliminary estimations still fit an A64 launch.
- A similar concepts to Neptune could not be established, mainly because of the significant increase for SOFC fuel demand, which would lead to launch mass beyond the A64 launcher capability.

APPENDIX A SUMMARY LAUNCH WINDOW ASSESSMENT (2030-2045)

The attached excel file to this document (Launch Windows 2030_2045.xlsx) provides an extensive list of launch opportunities to Saturn, and Uranus covering the period 2030-2045.

For each planet, it is considered the following optional fly-by trajectory sequences:

- Earth-Earth-Jupiter-Saturn/Uranus
- Earth-Venus-Earth-Earth-Jupiter-Saturn/Uranus
- Earth-Venus-Earth-Jupiter-Saturn/Uranus
- Earth-Venus-Venus-Earth-Jupiter-Saturn/Uranus

For each launch opportunity, it is provided the following information:

- Launch Date
- Escape velocity
- Declination
- Launch mass considering Ariane 64 (preliminary estimate)
- Total delta-v
- Total trajectory time
- For each fly-by: date, V infinity and altitude
- Arrival at the planet: date, V infinity, declination

Regarding the A64 payload capabilities, the preliminary performances presented Figure 26 have been assumed (no data available for escape velocities higher than 5.5 km/s). For declinations larger than +5° or smaller than -5°, it has been assumed that the spacecraft is launched into a parking orbit. For these cases, it has been assumed that the A64 performance is approximately 90% of a direct escape (declination 0°) with the same escape velocity. These performances must be considered preliminary and subject to significant evolution in the frame of the Ariane 64 design consolidation. Additional margins are suggested to be used.

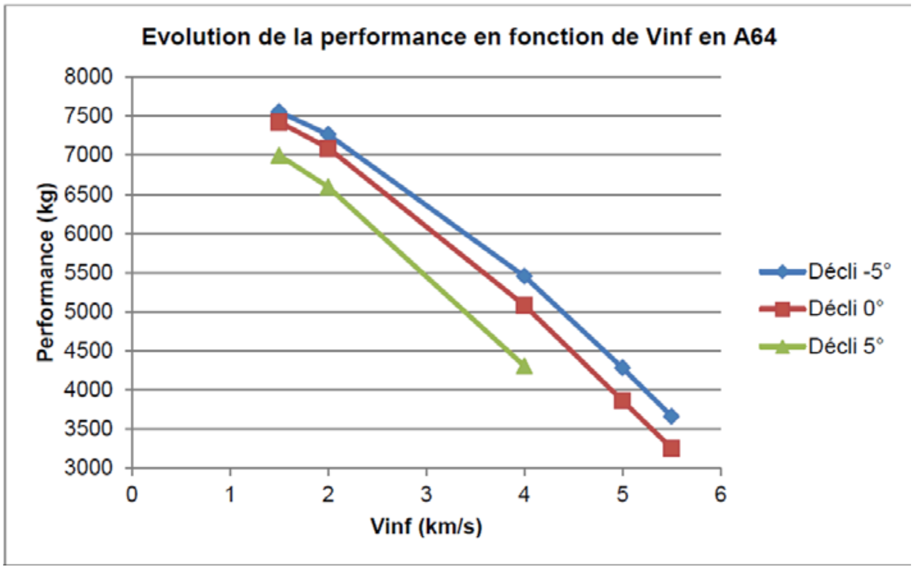


Figure 26: Ariane 64 preliminary performance

From the total list of trajectories to both Uranus and Saturn, the following subset of trajectories were traded-off during the EPIG CDF Study as potential candidates for the mission.

#	Case	Launch	JGA altitude [km]	Delta-v [m/s]	A64 performance [kg]	Duration [y]	Arrival	Backup (yes/no)	Relative arrival velocity (km/s)[1] for FPA of -35deg	Probe entry in daylight / Earth line-of-sight at FPA - 35deg	Multi-mission possibility (U + S)	Radiation compatibility
U1	EEJU31	2031/6/12	35318	1060	3944	9.3	2040/9/13	Yes	23.7 - 26.6	Yes	Yes	TBD
U2	EEJU32	2032/7/20	935558	1047	3233	9.5	2042/2/11	Yes	22.3 - 25.2	Southern latitudes only	Yes	TBD
U3	EVEJU31	2031/2/14	143804	808	4849	11.5	2042/8/11	TBD	TBD	TBD	TBD	TBD
U4	EVEEJU31	2031/5/23	794763	695	5166	10.3	2041/9/8	2030	22.7 - 25.7	Southern latitudes only	TBD	Yes
U5	EEEJU43	2043/8/31	343313	721	3321	12.3	2056/1/12	*Yes?*	21.3 – 24.4	Southern latitudes only	TBD	TBD
U6	EEJU45	2045/8/31	350164	696	3296	10.3	2056/1/27	*Yes?*	21.3 – 25.4	Southern latitudes only	TBD	TBD

Table 18: Uranus interplanetary transfer options

#	Case	Launch	JGA altitude [km]	Delta-v [m/s]	A64 performance [kg]	Duration [y]	Arrival	Backup (yes/no)	Relative arrival velocity for FPA -25deg (km/s)[1,2]	Probe entry in daylight / Earth line-of-sight at -25deg	A64 launcher performance – S/C wet mass (kg)	Radiation compatibility	Multi-mission possibility (U + S)
S1	EEJS31a	2031/6/12	254892	1030	3958	9.7	2041/2/12	Yes (S1-5)	31-43	Partially		TBD	Yes
S2	EEJS31b	2031/4/1	424021	583	3144	9.5	2040/10/25	Yes (S1-5)	30-44	No		TBD	TBD
S3	EEJS32a	2032/7/22	19656	810	3258	8.1	2040/9/16	Yes (S1-5)	30-44	No		TBD	Yes
S4	EEJS32b	2032/7/24	610512	972	2880	10.3	2043/11/3	Yes (S1-5)	29-43	No		TBD	TBD
S11	EEEJS32	2032/10/19	572000	679	3384	12.8	2045/7/22	**Yes (S5)**	**27-45**	Partially	503	Yes	
S5	EEJS34	2034/10/19	543020	740	3371	10.9	2045/8/21	Yes (S11)	27-45	Partially		TBD	TBD
S6	EVEJS31	2031/2/16	443488	797	4806	9.5	2040/8/25	Yes (S8)	30-44	No		TBD	TBD
S7	*EVEJS32*	*2032/11/6*	*361000*	*939*	3708	*9.0*	*2041/11/5*	*Yes (S6+8)*	*TBD*	*TBD*		*TBD*	*TBD*
S8	EVEJS34	2034/2/17	533944	720	4328	9.3	2043/6/19	Yes (S6)	30-43	No		TBD	TBD
S9	EVEEJS31	2031/5/23	225093	179	5166	9.6	2040/12/3	Yes (S10)	30-44	No		TBD	TBD
S10	EVEEJS32	2032/7/27	1236424	0	5259	9.1	2041/11/9	Yes (S9)	30-44	No		TBD	TBD

Table 19: Saturn interplanetary transfer options