ENCELADUS WATER SAMPLE RETURN & OCEAN EXPLORER (ENDURANCE)

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Abstract

Cassini mission revealed the existence of a subsurface ocean on Enceladus via actively jetting geysers at its south pole. Measurements unveiled the presence of essential chemical ingredients for life development. To follow-up on these discoveries, this mission will seek for biosignatures by further investigating its plumes, crust & ocean.

Keywords: Enceladus Sample Return Astrobiology Biosignatures

Acronyms/Abbreviations

Spacecraft (S/C), End-of-Life (EOL), Saturn Orbit Insertion (SOI), Enceladus Orbit Insertion (EOI), Low Earth Orbit (LEO), Chemical Propulsion (CP), Electric Propulsion (EP), Technology Readiness Level (TRL) Thermal Control System (TCS), Electrical Power System (EPS), Gridded Ion Thruster (GIT), Hall-Effect Thruster (HET), Orbit Maintenance (OM), Next Generation Radioisotope Thermoelectric Generator (NGRTG), Multi-Layers Insulation (MLI)

1. Mission directive

The mission directive of this project was to design a space mission of *Sample return from a moon beyond the Asteroid Belt*. It was clarified with the project director David Rodriguez that the spacecraft must return a physical sample either in solid, liquid, or gaseous state to Earth. No constraints on budget nor time allocations were set.

2. Mission statement

From 2005, during the Cassini-Huygens mission (1997-2017), many flybys of Saturn's icy moon Enceladus were performed due to atmosphere suspicions. Close flybys revealed the presence of geysers, actively jetting matter into space from the South Pole region, which allowed Cassini to sample these plumes. These geysers eject a mixture of water and ice particles at a speed of 400 m/s suggesting the existence of an underground liquid ocean beneath Enceladus' icy crust (-200°C surface), which was confirmed about a decade later. From collected data, scientists suggest this liquid ocean is about 10 km deep in the South Pole region, buried under a layer of ice 30-40 km thick and fueled by hydrothermal vents releasing hot, mineral-rich water (see Fig. 2).

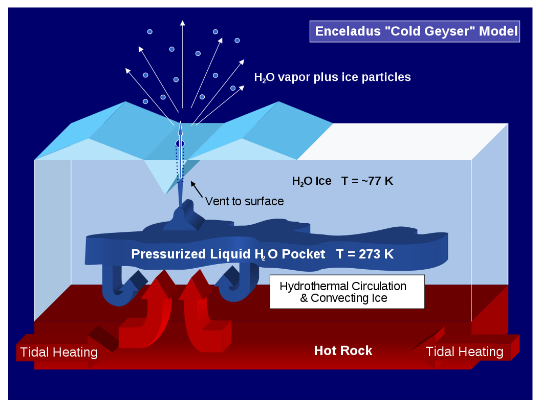

Fig. 2: Enceladus's cryovolcanism model

In 2023, new scientific studies revealed the presence of phosphates [6] and hydrogen cyanide [5], key molecules to the origin of life alongside with life-sparking energy source evidence inside its ocean. These discoveries further enhance Enceladus status as a prime destination for the search for traces of life beyond Earth in our Solar System.

To follow-up on the discoveries of Cassini, ENDURANCE mission & science objectives were established:

1. Investigate Enceladus possible biosignatures through plumes samples capture & return.
   - Characterize plumes chemical composition.
   - Search for biosignatures in the samples' chemical compounds.
2. Further assessing Enceladus ability to sustain life.
   - Characterize Enceladus physical and chemical environment.

- Study Enceladus cryovolcanism.
  - Characterize its internal structures (geysers, crust, ocean & hydrothermal vents) and their interaction.
  - Investigate its formation and evolution.

3. Mission design

In [4], NASA determined that a launch in October 2038 (with a backup window in November 2039) is ideal to maximize south pole lighting during their science phase (2050-2054): this option performs a direct transfer exploiting a Jupiter gravity assist enabling low arrival hyperbolic excess velocity V_∞ at Saturn enabling low ΔV for Saturn Orbit Insertion, thus allowing a higher dry mass. The launch would take place at Kennedy Space Center and may be performed by the Space Launch System (SLS) Block 2 Cargo (7030 kg).

Other options with Falcon Heavy Expendable (FHE) or Ariane 64 exist. However, SLS enables direct trajectory with Jupiter flyby thanks to high characteristic energy $C3$ to shorten transit time to Saturn (7 years). FHE offers low-cost alternative for about the same launch mass but requires an inner cruise to use Venus flybys in place of the high $C3$ (requiring a modified thermal design). Similar Ariane 64 options (launching in 2032 from Kourou) only allow about 3400 kg at launch [1]. In addition, it is most likely such window would be missed having less than 10 years ahead for a complete mission development.

Launcher	Site	Window	Launch mass
Ariane 64	Kourou	2032	3364 kg
SLS B2 Cargo	KSC	2038/9	7030 kg
Falcon Heavy	KSC	2038/9	6600 kg

Table 1: Launch options

Thereby, ENDURANCE is planned to be launched in 2038 from Kennedy Space Center and injected into a direct cruise heliocentric orbit targeting a gravity assist with Jupiter 2 years later, allowing to reach Saturn in 2046 with a hyperbolic excess velocity of only $V_{a_S,\infty} \approx 3$ km/s.

As Enceladus escape velocity is about 0.24 km/s, if a direct insertion into Enceladus orbit from the hyperbolic trajectory with an appropriate timing was possible, a $\Delta V \approx 3$ km/s maneuver would be necessary. The most efficient solution for this maneuver requires a complex sequence of Saturn's moons gravity assists in order to decrease Enceladus arrival velocity: this *pump-down* technique was used during Cassini-Huygens mission.

Thus, upon arrival at Saturn, SOI maneuver is performed ($\Delta V_{SOI} = 0.85$ km/s) and then begins the complex 4.5 years pump-down sequence ($\Delta V_{pump} = 0.75$ km/s). It proceeds with pump-down flybys of Titan, Rhea, Dione, and Tethys, bringing the spacecraft to a first Enceladus encounter. Next, a series of Enceladus flybys further reduce the excess velocity to $V_{a_E,\infty} \approx 0.2$ km/s. Finally, the spacecraft is inserted ($\Delta V_{EOI} = 0.1$ km/s) in an Enceladus elliptical orbit of 40-60 km periapsis above South Pole ($T \approx 3$ h) to facilitate plumes collection. Science phase begins and is planned to last from 6 months to 1.5 year ($\Delta V_{OM} = 0.3$ km/s) depending on waiting time to provide appropriate phasing conditions for return journey.

The outward journey being clearly defined thanks to available studies [4], a 5% margin [9] was added to their ΔV contributions (see Table 2).

Return trajectory is still uncertain as very little data exists. In any case, one can assume the spacecraft would require a similar gravity assists sequence than at arrival to escape Saturn's gravity (pump-up sequence), mainly using Titan [7]. Then, depending on the achievable escaping conditions, one may perform a direct return trajectory or gravity assists with Jupiter, Earth or Venus. Therefore, it was assumed the inbound journey would require at most twice the ΔV needs up to EOI (100% margin). For the return duration, it was shown in [7] a round trip with Enceladus orbiting configuration requires about 26 years. Thus, it was assumed an inbound journey of less than 15 years was achievable (adding an extra 2-year margin, see Fig. 1). Following Earth Orbit Insertion in LEO, the spacecraft will dock into existing (private) space stations to deliver the samples capsule that will be carried back to Earth soil by humans to avoid reentry under high thermal loads. After samples reception, the spacecraft will be deorbited and will burn in Earth's atmosphere to minimize space debris.

Such deep space missions requiring RTGs (i.e. Cassini) due to low solar power availability, in space disposal measures shall be implemented as these systems cannot reenter

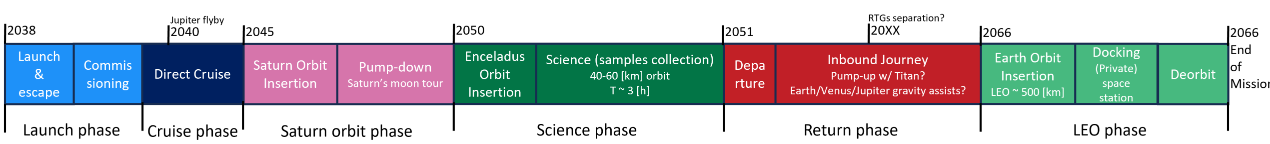

Fig. 1: Timeline & mission phases

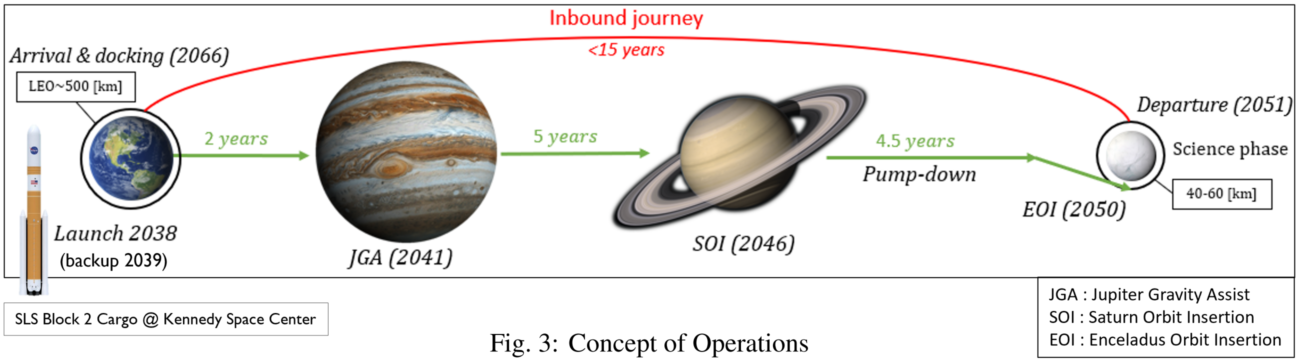

Fig. 3: Concept of Operations

Earth's atmosphere for safety reasons. Therefore, they shall be separated from the spacecraft prior to reentry: as letting them become deep space debris is not an option, then depending on the return trajectory, one may target for either an impact on Jupiter (ideally) or on Venus.

Event	**ΔV (km/s)**	**Margin** (%)
Launch & escape	0	
Jupiter Gravity Assist	0	
Saturn Orbit Insertion	0.85	5
Pump-down sequence	0.75	5
Enceladus Orbit Insertion	0.1	5
Orbit maintenance	0.3	5
Inbound journey*	1.7	100
Subtotal	3.7	
Total (with margins)	5.5	

Table 2: ΔV budget

With the various flybys of celestial bodies of this mission concept, one should take advantage to perform science with these encounters (not detailed in this study), in particular of the Saturn's moons during the pump-down/up sequences as they were only visited by Cassini.

4. Systems engineering

This sample return mission is characterized by 5 main functions: go to Enceladus, collect samples, take measurements, protect samples & return to Earth (see Fig. 4).

A list of the associated key requirements from mission-level to subsystem level (high-level only) is provided below:

1. **Functional requirements**
   (a) The sample capture system shall be able to capture Enceladus geysers plumes.
   (b) Samples conservation systems shall protect the samples against the space environment.
2. **Performance requirements**
   (a) The S/C shall carry [750][+/-250]g of plumes.
   (b) S/C shall provide ΔV = 5.5 km/s.
3. **Constraints**
   (a) Launch mass shall be less than 7030 kg.
   (b) The S/C dry mass shall be less than 2010 kg.
   (c) The S/C shall withstand temperatures from 108°C to -164°C.
4. **Interface requirements**
   (a) The S/C docking system shall be compatible with LEO 2060s space stations.
5. **Environmental requirements**
   (a) RTGs shall not reenter Earth's atmosphere.

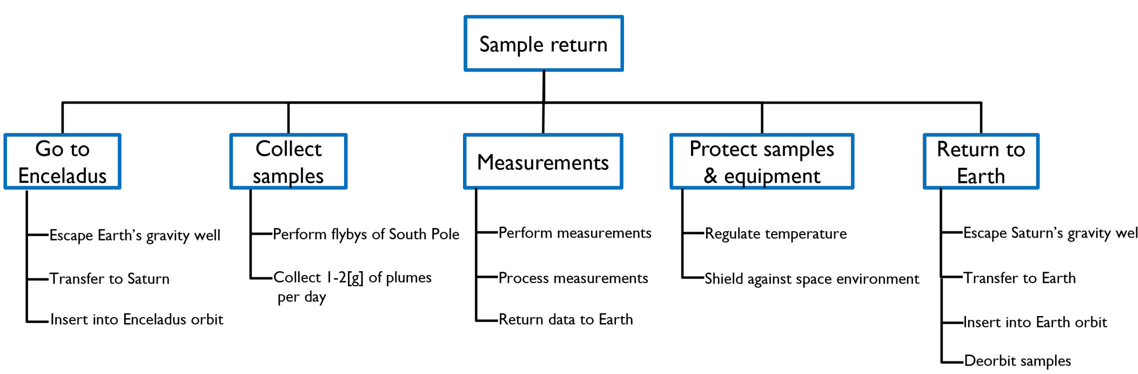

Fig. 4: Functional tree

6. **Others**

   (a) S/C equipment should be reliable for 30 years.

Mission & subsystems designs rely mostly on the return trajectory where major trade-offs occur to find the perfect fit between ΔV, time, mass & power (see Fig. 6 & Fig. 5).

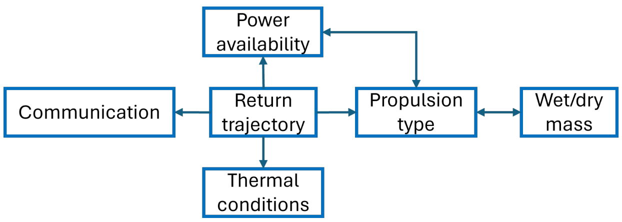

Fig. 6: Map of mission-subsystems design dependencies

The achieved level of success of the mission will depend on these various indicators (by order of importance):

1. Samples quantity returned to Earth: is the requirement verified? (minimum success criterion, Sect. 1)
2. Samples quality: are the samples exploitable?
3. Were data & measurements received? Are they usable?

5. Mission architecture

To design this mission, various conceptual solutions were assessed (see Fig. 5). At first, trajectory types for sample return were evaluated (Free return, Saturn-orbiting & Enceladus-orbiting). An Enceladus-orbiting trajectory was preferred over the others due to time available to collect the samples & to perform science. This option also allows to perform the sample collection at relatively low velocity ($\approx$ 0.2 km/s) compared to the others (> 4 km/s) [7] and many opportunities, allowing to collect a few grams per day.

Next, systems architecture were studied, leading to the selection of an orbiter architecture for increased in-orbit science capabilities & better mass allocation. A previous iteration of the mission with an orbiter with lander was investigated leading an unfeasible mission due to an excessive mass (about 9000 [kg] wet mass). An orbiter with landing capabilities may become feasible in the future ($\Delta V = 0.25$ km/s from orbit to landing) combined with *in situ* resource utilization to produce return journey propellant but its TRL is too low so far.

Finally, propulsion system types were compared. A chemical propulsion solution leads to a 1220 kg dry mass from Tsiolkovsky equation ($m_{dry,Cassini} \approx 2500$ kg), while an electric propulsion solution requires either 280 m^2 solar panels or 9 RTGs to power the spacecraft at Saturn's distance (10 AU). A combined propulsion system solution was found allowing to increase by almost 800 kg the total dry mass of the spacecraft compared to fully chemical.

The launcher capabilities are presented below:

	Mass (kg)
Launch vehicle lift capability	7468
Total launch mass (5% margin)	7030
Launch adapter mass w/ contingency	100
Available dry mass	2010

Table 3: Launch capability [4] (pp. 22)

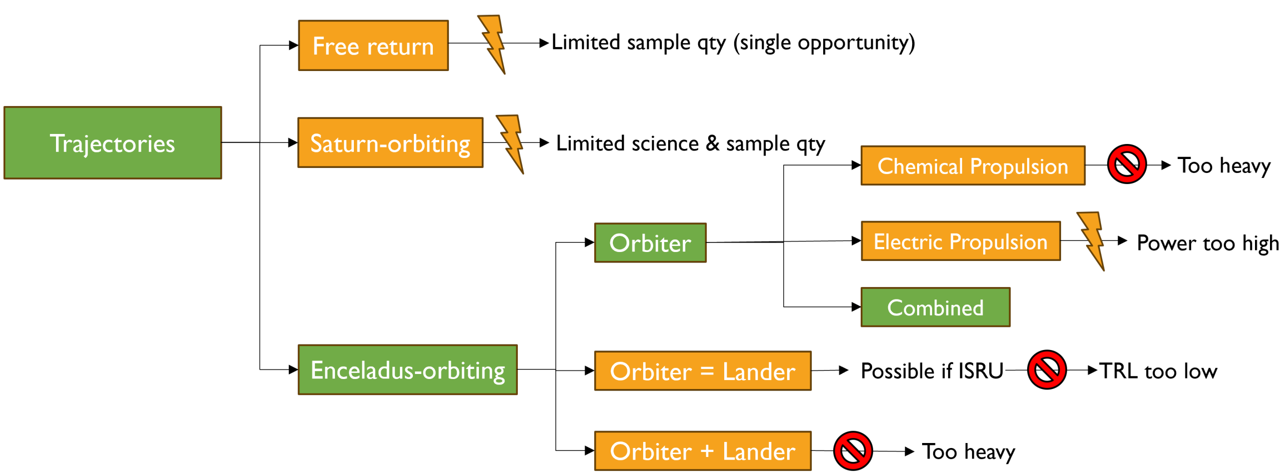

Fig. 5: Alternative conceptual solutions

The system budgets of the spacecraft are the following:

Subsystem	**Mass** (kg)	**Power** (W)
AOCS	66	144 (409)
C&DH	15	33
Communications	98	87
Payload	157	302
Power	406	131
Propulsion	353	945-4725
Structure	375	39
Thermal	69	39
Subtotal	1539	
Harness (10%)	154	
Nominal dry mass at launch	1693	
ESA system margin	20%	
Total dry mass at launch	2032	

Table 4: ENDURANCE subsystems budgets

All the subsystems design can be found in Sect. 6, in particular the detailed power budget with the system modes can be found in Table 13.

Subsystem	**Description/components**
AOCS	• 3-axis stabilized, reaction wheels (4) & HET thrusters • Star trackers (3), IMU (2) & sun sensors
C&DH	MIL-STD-1773 (10 Mbps usable, fault tolerant), temporary storage on SRAM DDR3 4 Gb
Communications	• Ka-band with 45.1 [dB] (uplink, 2000 [bps]) & 19.8 [dB] (downlink, 10 [Mbps]) margins (QPSK) • 2-m diameter parabolic dish HGA (Gain 54 [dB]) • Ground stations → Deep Space Network (34m BWG antennas arrayed)
Configuration	• 4mx4mx4m cube • Deployed & near-Earth configurations (for thermal hot-case)
Electrical power	• 75 [m2] solar panels, triple-junction cells (262 [W] @ 14 years) • 2 NGRTGs (380 [W] @ 14 years) • Secondary NiH_2 2250 [Whr] batteries
Payload	8 instruments (2 CAMs, 2 mass spectrometers, microscope, thermal emission spectrometer, radar & laser altimeter) + sample collection (funnel) & conservation (capsule) systems → 1.1 Tb data
Propulsion	• HTP/TMPDA green bipropellant (Isp 320s) • 4 Xenon Hall-effect thrusters (Isp 1800s), 0.9-4.5 [kW]
Structure & mechanisms	• Aluminum 7075 frame (0.05mx0.1mx4m), honeycomb panels • Antenna steering, solar panels deployment & solar array drive
Thermal	• MLI (aluminized Kapton), heaters (solar panels), heat pipes & radiators

Fig. 7: Spacecraft design summary

It can be seen the available dry mass is extremely close to the total dry mass at launch of the spacecraft. However, as the return trajectory was certainly upper bounded in terms of ΔV needs and ESA margin philosophy was adopted at subsystem & system level, this should ensure the mission is feasible.

To improve future iterations, one should also adopt ESA margin philosophy for power budgets [9], with a 20% margin at subsystem and system level which would induce changes in the EPS design.

The major risks of the mission and their mitigation strategies are the following:

- Qualification of equipment for 25-30 years mission (Cassini = 20 years) → redundancy
- Samples alteration (space environment & contamination) → shielding strategy, perform as many on-board analyses as possible
- Green bi-propellant TRL too low → hydrazine/NTO
- RTGs planetary & safety policies → separation prior Earth orbit insertion
- Human operations for 30 years (transmission & hardware/software compatibility) → documentation

6. Baseline design

6.1 Payload components

The payload components of this mission are considered as the various science instruments selected to answer the science objectives (Sect. 2).

- Wide & Narrow-Angle Cameras (OSIRIS-REX): images of Enceladus' surface and detailed views of geysers/cracks
- Laser altimeter (OSIRIS-REX): detailed topography of South Pole and its geysers (high-resolution topographical information)
- Mass spectrometers: samples chemical composition characterization.
  - High-resolution (Europa Clipper): volatile organic and inorganic characterization.
  - Separation (DragonFly): simple and complex molecules characterization, including amino-acids and lipids.
- Microscope: investigation of potential cells & microorganisms in the samples.
- Radar sounder: crust structure characterization.

- Sample collection system: funnel to collect plumes during South Pole flybys.
- Sample conservation system (Hayabusa-2): capsule to protect plumes against space environment (esp. radiations), includes thermal control.
- Thermal emission spectrometer (Europa Clipper): temperature mapping, jet vents physical structure characterization.

Their specifications (mostly retrieved from [4], pp.49-69) are presented below:

Components	**Mass** (kg)	**Volume** (m^3)	**Power** (W)	**Data** (Gb)/(kbps)
Narrow-angle camera	20	0.11	5	159 (4190)
Wide-angle camera	0.4	0.19	2.5	0.6 (4190)
Mass spectrometers	32	0.06	86	24
Microscope	3	0.01	15	0.9
Laser altimeter	7.4	0.02	16.5	3.9 (10)
Radar sounder	12	0.07	25	672 (8000)
Thermal emission spectrometer	3.8	0.01	13	0.6 (1160)
Sample collection system	20	0.5	0	0
Sample conservation system	35	0.05	20	0
Subtotal	131		232	861
Margins	20%		30%	30%
Total	157	1	302	1119

Table 5: Payload components specifications & budgets

Note data volume values were provided for a duration of approximately 0.5 year, but as science phase could last up to 1.5 years, these were scaled up. It can be derived the instruments are working less than 1% of the science phase. Data volume margin allows to include other measurements that may be taken during other phases of the mission.

Field of view requirements of instruments are [4] (pp. 55-56): narrow-angle camera (0.293°), wide-angle camera (44.003°), thermal emission spectrometer (1°) and laser altimeter (0.02°).

6.2 Communications

This subsystem is driven by the science data volume to downlink (payload), by the maximum distance between the spacecraft and Earth (trajectory) and the communication windows.

Cassini was taken as baseline being the only spacecraft that stayed around Saturn & its moons. ENDURANCE will use a 2-meter parabolic dish (most mass-efficient solution for data return to meet science needs) high-gain antenna (HGA) to receive commands (uplink) and to send telemetry & science data back to Earth (downlink). This mission will use NASA's Deep Space Network (34m BWG antennas arrayed in Goldstone, Madrid and Canberra) as ground stations.

The link budget was determined for X-Band (7190 MHz) and Ka-Band (34700 MHz) using the standard procedure from [3] (pp. 533-575), assuming Earth and Enceladus in opposite phase (worst-case). DSN capabilities were provided in [8].

From Table 5 assuming measurements are uniformly distributed across the science phase, it can be derived the daily generated data volume is about 2 Gb per day. Link budget assumed data rates of 2000 bps for uplink (about 4x Cassini needs) and 10 Mbps for downlink, allowing to transmit the daily data volume within minutes. Thus, it seems regardless the communication windows duration, it will be possible to downlink the daily data if the link margin is sufficient.

Using a QPSK modulation ($\frac{E_b}{N_0} \approx 11.2$ dB for $BER = 10^{-7}$), allowing a bandwidth of 10 MHz, the link budget is as follows:

Link	**EIRP** (dBW)	**G/T** (dBK)	**Ls** (dB)	**La** (dB)	**R** (dBHz)	$\frac{E_b}{N_0}$ (dB)	**Margin** (dB)
Up X	109.6	9.9	214.3	3	33	77.8	66.6
DL X	53.5	19.1	214.3	3	70	-7.3	-18.5
Up Ka	103.6	13.2	200.6	45	33	56.3	45.1
DL Ka	67.2	22.2	200.6	45	70	31	19.8

Table 6: Link budget

Therefore, a Ka-band system is chosen for downlink and uplink (19.8 dB & 45.1 dB link margin respectively). Eventually, a dual-band system is possible with X-band for 2-way (commands & telemetry) and Ka-band for science data return. From [3] (pp. 574) and scaling, the subsystem budgets were derived:

Component	**Mass** (kg)	**Volume** (m^3)	**Power** (W)
High-gain antenna	65	2x2x1	0
Others (transmitter, etc.)	17	-	67
Subtotal	82		67
Margins	20%		30%
Total	98	2	87

Table 7: Communication subsystem budgets

As rule of thumb, it was assumed the HGA is 80% of the subsystem mass and uses Cassini's 20 W transmitter (assuming 30% efficiency).

6.3 Command & Data Handling

This subsystem is driven by the science instruments data budgets (payload) and communication windows for storage capacity.

- Data bus speeds requirement: max 0.8 Mbps single channel (radar sounder)
- Data storage requirement: 4 Gb (2 Earth days without communication).

From the methodology presented in [3] (pp. 395-407), the following architecture was derived.

- Upgraded Cassini on-board computer (large size S/C)
- Bus electrical interfaces: CAN Bus (≈ 1 Mbps) or MIL-STD-1773 (10 Mbps usable, fault tolerant)
- Memory type: drivers (temperature, radiation, 30 years operations) → SRAM (non-volatile)
- Complexity (command/telemetry): long duration mission → radiations (shielding), lifetime & reliability key → complex (worst-case)
- Budget (derived from [3], pp. 405) for complex

	Volume (m^3)	**Mass** (kg)	**Power** (W)
Command & DH	0.016	12.5	25
Subtotal	0.016	12.5	25
Margins		20%	30%
Total	0.02	15	33

Table 8: Command & Data Handling budgets

6.4 Propulsion

This subsystem is driven by ΔV needs (5.5 km/s), launch mass (7030 kg) and propulsion type/power supply.

With the mission settings, it was shown in Sect. 5 a combined CP/EP system architecture was necessary (see Fig. 5) where EP is used for orbit maintenance at Enceladus & on the return trajectory below 5 AU (≈ 1 kW can be generated from solar panels at this distance).

Event	**DV** (km/s)	**Propulsion**
Launch to EOI	1.785	Chemical
Orbit maintenance	0.315	Electric
Enceladus → 5 AU	1.7	Chemical
5 AU → Earth OI	1.7	Electric
Total	5.5	

Table 9: Alternative ΔV budget with propulsion type

This distribution should allow to put the spacecraft into a "fast" route thanks to CP (instead of long duration EP thrusting to reach cruising velocity) while EP allows to progressively reduce spacecraft velocity to arrive on Earth with a very low hyperbolic excess velocity $V_{a,\infty}$ for smooth capture.

Psyche was taken as reference for EP design. It is the first interplanetary spacecraft to use solar-powered Hall-effect thrusters for propulsion and orbital maneuvering and its propulsion system is able to deliver $\Delta V > 8$ km/s ($\Delta V_{EP} \approx 2.1$ km/s) for this mission. As the limiting factor for EP is the power supply at Enceladus level for orbit maintenance, HETs were chosen due to greater thrust-to-power ratio than GITs and less intensive power supply (also smaller, simpler, even though lower I_{sp} efficiency). The spacecraft will have 4 SPT-140 HETs (I_{sp} = 1800s, 0.9-4.5 kW throttleable) while using once at a time, providing up to 280 mN of thrust.

For the chemical propulsion system, traditional hydrazine/NTO ($I_{sp} \approx$ 320s) would be traded for greener bipropellant options such as high-test peroxide (HTP) as oxidizer and isopropyl alcohol (IPA) or TMPDA as fuel ($I_{sp} \approx$ 320s, 200-500 N thrust, O/F ratio = 5:1). These options are non-toxic, allow safe on-ground operations and have lower contamination risk. Currently, these technologies are still under development and their in-orbit storability duration is still to confirm.

This architecture leads to the following propellants and propulsion budgets respectively:

Propellants	**Mass** [kg]
High-test peroxide	3670
TMPDA	730
Xenon	285
Subtotal	4685
Margin	5%
Total	4920

Table 10: Propellants budget

Component	**Volume** (m^3)	**Mass** (kg)	**Power** (W)
HTP tank(s)	2.5	130	0
TMPDA tank	0.9	50	0
L-XTA Xenon tank	0.3	44	0
AJ R-4D thruster	-	10	0
HETs (SPT-140) (4)	-	60	900-4500
Subtotal		294	900-4500
Margins		20%	5%
Total	3.7	353	945-4725

Table 11: Propulsion budgets

6.5 Attitude and Orbit Control (AOCS)

For this mission, AOCS is driven by the scientific instruments pointing accuracy requirements and the mission duration (redundancies). As some scientific instruments require high pointing accuracy to perform measurements ($FoV_{min} = 0.02°$, laser altimeter), the spacecraft will use a 3-axis stabilization method. Reaction wheels were preferred over small thrusters for propellant savings & better pointing performance.

To optimize the combined propulsion architecture, orbit maintenance ($\Delta V_{OM} = 0.315$ km/s) and reaction wheels desaturation are performed by the hall-effect thrusters assuming it can be done with 0.9 kW input power, during this phase science instruments are turned off (except samples conservation system).

For such long duration mission, a fully redundant subsystem is necessary: attitude and rate knowledge are obtained from two box-redundant star trackers (ST) and one internally redundant inertial measurement unit (IMU). Control is maintained via three reaction wheels, with a fourth available as a spare. As life expectancy of reaction wheels is 15-20 years, it may be taken over by HETs for the last 5-7 years if necessary. From [3] (pp. 354-380) and [4] (pp. 127), AOCS budgets are established below:

Component	**Volume** (m^3)	**Mass** (kg)	**Power** (W)
Reaction wheels (3+1)	0.05	26.4	66 (270)
Sun sensors	0.003	4.5	3.3
Star trackers (2+1)	0.03	7.8	11.2
IMU (1+1)	0.017	16	30
Subtotal		54.7	110.5 (314.5)
Margins		20%	30%
Total	0.1	66	144 (409)

Table 12: AOCS budgets, peak values in brackets

6.6 Electrical power

Electrical power subsystem (EPS) design was driven by the solar power availability throughout the mission (trajectory) and the system modes (esp. longest eclipse phase and hall-effect thrusters use).

The spacecraft has 6 system modes: post-launch (PL), chemical propulsion cruise (C CP), science (Sci), orbit maintenance (OM), eclipse (Ecl) and electric propulsion cruise (C EP) (see Table 13).

It can be seen if the spacecraft orbit maintenance phase around Enceladus is covered by EPS design, then all the other phases are also covered since EP cruise mode is only used below 5 AU where solar panels can provide sufficient power to the propulsion system. Indeed, orbit maintenance requires a high power consumption as HETs are fired. It is assumed subsystems (except AOCS & propulsion) are set to minimal power consumption during this mode.

Subsystem	**PL** (W)	**C (CP)** (W)	**Sci** (W)	**OM** (W)	**Ecl** (W)	**C (EP)** (W)
AOCS	409	144	144	409	144	144
C&DH	33	33	33	33	33	33
Communications	87	87	87	0	0	87
Payload	0	26	302	26	26	26
Power	131	131	0	0	0	131
Propulsion	0	0	0	945	0	945-4725
Structure & mech.	39	39	39	0	0	39
Thermal	0	39	39	39	39	39
Total	699	499	644	1442	242	1444-5224

Table 13: EPS modes

Currently, it is not feasible to rely on solar panels only as power supply beyond Jupiter (5 AU). At Jupiter, the mean solar irradiance is still about 50 W/m^2: JUICE has 85 m^2 solar panels producing about 850 W. At Saturn's distance (10 AU, 15 W/m^2) it would produce only 255 W or require about 170 m^2 of solar panels to produce the same amount of power. Therefore, RTGs are necessary to complement solar power supply similarly as Cassini. These devices are well suited for such deep space mission, being independent of Sun and unaffected by S/C orientation & shadows. Most importantly, they generate about 4400 W of thermal power per unit which can be recovered partially to heat the spacecraft in such cold environment (see Table 16). However, RTGs implementation required a proper disposal strategy considering planetary policies for return as they shall not reenter Earth's atmosphere.

From trajectory, the longest eclipse phase is due to Saturn's shadow when orbiting Saturn or Enceladus and its duration $T_{eclipse} \approx 180$min is given in [2] (pp. 206). Enceladus orbiting period being only $T_{orb} \approx 180$min, eclipse due to Saturn is effectively the design case. Potential eclipses when leaving Earth and Jupiter flybys are assumed negligible as these bodies are not orbited.

From this knowledge, the following EPS architecture was determined applying the methodology presented in *Lecture 06* (or [3] pp. 407-427):

- Solar panels based on Psyche: 75 m^2, triple-junction cells ($\eta = 0.35$, 0.5% per year degradation [3] pp. 412), producing 262 W at 10 AU at 14 years. Provides 938 W at 22 years (5 AU) and 22'500 W at 28 years (1 AU).
- 2 NGRTGs (380 W at 10 AU at 14 years) to guarantee enough power with solar panels for science phase, also providing enough power on its own for eclipse mode. Provides 328 W at 22 years (5 AU) and 294 W at 28 years (1 AU).

- Secondary NiH_2 batteries sized to bridge the gap during orbit maintenance & recharged with excess (C_r = 2250 Whr, $DOD = 40\%$, T_e = 180 min).
- A PCDU unit for power distribution and regulation with maximum power point tracking (MPPT) for controlling generated power (most used in interplanetary), uses 4-7% of total power.

Power budget was sized to ensure sufficient power supply at 14 years (10 AU) as it is the lowest point of power supply and where latest orbit maintenance mode occurs. Beyond this point it was assumed the spacecraft is on its return trajectory and the global power production gradually increases (from solar panels) despite RTGs and solar cells degradation.

Component	**Volume** (m^3)	**Mass** (kg)	**Power IN** (W)	**Power OUT** 14 years (W)
Solar panels (75 m^2)	3.75	150	0	262
NASA NRTGs (2)	0.32	111.4	0	380
Batteries	0.04	39	0	800 (OM)
PCDU w/ MPPT	0.05	38	101	0
Subtotal		338.4	101	642 (1442)
Margins		20%	30%	
Total	4.2	406	131	642 (1442)

Table 14: EPS Budgets

6.7 Thermal

Thermal subsystem design is driven by the equipment temperatures requirements (see Table 15) and the hot & cold cases of the mission (trajectory/eclipse)

Equipment	**Operational** (°C)	**Survival** (°C)
Avionics baseplates	-20 to 60	-40 to 75
Batteries	10 to 30	0 to 40
Hydrazine fuel	15 to 40	5 to 50
Solar arrays	-150 to 110	-200 to 130
Antennas	-100 to 100	-120 to 120
Reaction wheels	-10 to 40	-20 to 50

Table 15: Typical temperature requirements

It was assumed scientific instruments already have in-house temperature control devices (Peltier, heaters, etc.) accounted in their power budget.

To determine the hot and cold cases, knowing NGRTGs provide a total excess heat of 8800 [W], it was assumed half of it can be recovered from radiation & redistributed with pipes. Moreover, it is assumed $\approx 20\%$ of all power inputs are also converted into heat, then $P_{int} \approx 4500$ [W].

Applying the methodology presented in *Lecture 10* (or [3] pp.428-458) with a few possible surfaces, the following thermal cases were obtained:

Surface	**Earth**	**Jupiter**	**Saturn Enceladus**	**Eclipse**
Goldized Kapton	270	144	139	136-143
Polished Aluminum	108	22	18	16-21
Aluminized Kapton	122	56	54	52-55
Solar Panels	66	-124	-163	-164

Table 16: Thermal analysis (°C) for various surfaces

Eclipse also corresponds to deep space travel beyond Jupiter (where mean solar irradiance becomes negligible). From Table 16, aluminized kapton was selected as MLI with hot case at Earth during post-launch $T_{hot} \approx 122$°C and cold case at Saturn/Enceladus/Eclipse $T_{cold} \approx 52$°C. In reality, both cases are certainly colder (RTGs assumptions too wide), thus aluminized kapton MLI provides extra margin. However, in the event a Venus gravity assist for the return trajectory, then the hot case has to be redefined and so the cooling strategy.

Based on this knowledge, the thermal architecture/strategy and its budgets were defined:

- Solar panels $\rightarrow$ Kapton heaters
- RTGs $\rightarrow$ radiators for early/late journey (below 2 AU)
- Active & passive heat pipes to conduct RTGs recovered heat and for finer temperature control (e.g batteries, antenna, fuel, reaction wheels, etc.)
- As absorptivity α increases over time $\rightarrow$ adjust S/C orientation $\rightarrow$ shadows from antenna + solar panels (also during hot cases)

Component	**Volume** (m3)	**Mass** (kg)	**Power** (W)
MLI (Aluminized Kapton)	0.01	32	0
Radiators	0.4	15	0
Heat pipes (active/passive)	0.1	10	5
Kapton heaters	≈ 0	0.5	25
Subtotal		57	30
Margins		20%	30%
Total	0.5	69	39

Table 17: Thermal budgets

6.8 Structures & mechanisms

Structure is driven by the volume/envelope of the spacecraft and its necessary deployable/steerable equipment.

From volume budgets $V_{tot} \approx 16$ m^3, a 400% margin (harness, yet unknown equipment, constraints) was applied leading to a total spacecraft volume of $V_{S/C} \approx 64$ m^3. It corresponds to a cube with $L = 4$m (launcher payload bay diameter is at least 7m for an available volume of 988 m^3).

Following the procedure presented in *Lecture 09* (or [3] pp. 459-497), assuming an aluminum 7075 frame completed with honeycomb panels, it was found an optimal frame of 0.05x0.1x4m bars were necessary to withstand the launch loads strengthened with 48 m^2 honeycomb sandwich panels ($\rho_{panels} \approx 1$ kg/m^2).

In addition, solar panels require deployment and drive mechanisms to orient themselves as perpendicular as possible to Sun's irradiance. The high-gain antenna also requires a steering mechanism to orient towards Earth.

Component	**Volume** (m^3)	**Mass** (kg)	**Power** (W)
Frame (Al 7075)	0.08	224	0
Honeycomb panels	1.44	48	0
Antenna steering	0.01	10	20
Solar panels deployment	0.01	10	0
Solar array drive	0.03	20	10
Subtotal		312	30
Margins		20%	30%
Total	1.6	375	39

Table 18: Structure & mechanisms budgets

6.9 Spacecraft configuration

In its main configuration, the spacecraft is as follows:

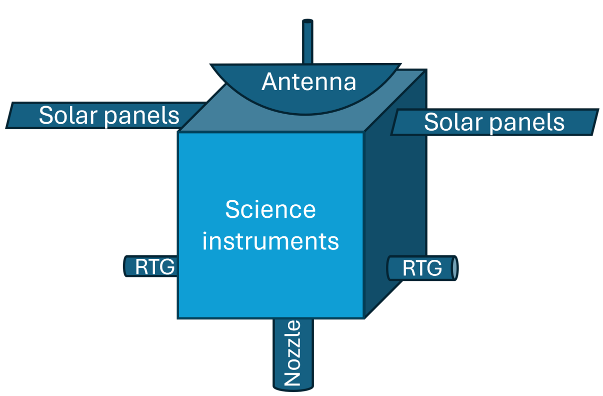

Fig. 8: Journey configuration

with HETs system is located at its back.

Overall, the spacecraft has 4 configurations:

- Launch: solar panels are folded.
- Journey: solar panels are fully deployed.
- Communication: HGA is oriented towards Earth.
- Thermal: if necessary during hot cases (may occur below 2 AU), the spacecraft is oriented to maximize shadow with the high-gain antenna and solar panels.

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