ELDER Concept Mission
Orbiter Spacecraft to Test Habitability of Enceladus

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The Saturnian moon Enceladus bears potential for unique life formation, but remains unexplored in this area. This report outlines the spacecraft design for a life detection mission to Enceladus, designed to launch in the next few decades. The designed vehicle weighs 2606 kg at launch, has a peak power draw of 691 W, and includes two stages: 1) solar electric propulsive maneuvers up to Venus, followed by 2) chemical propulsive maneuvers to carry the scientific payload to Enceladus. - Juan Salazar

I. Nomenclature

ΔV	=	Spacecraft change in velocity
P_t	=	transmission power
G_r	=	receiver gain
G_t	=	transmission gain

II. Introduction - Logan Kluis

The Enceladus Life Detection Exploration and Reconnaissance (ELDER) mission will search for signs of life on Enceladus and constrain the habitability of life in its subsurface oceans. The mission begins by launching from an Atlas-V and entering a trajectory toward Enceladus that includes Venus, Earth, Jupiter, and Saturn flybys. Once in orbit around Enceladus, the satellite will be collecting data from the plumes that erupt from the moon's south pole. The collected particles are then captured and processed *in situ* and the data is sent back to Earth for further analysis. Photos and video are also sent back to Earth for public outreach and scientific investigation.

The spacecraft team is tasked with choosing and analyzing the systems for propulsion, power, and data transmission and collection. The team also sets the allocations for mass and power to place constraints on the other subsystem teams. Minimum requirements are based off mission objectives and requirements from other subsystem teams.

III. Requirements - Miguel Wagner

In order to ensure the ELDER mission accomplishes its mission the spacecraft design sub-team was responsible for meeting several sub-system requirements. Each of these sub-system requirements corresponds to a system level requirement which in turn corresponds to a mission level requirement. The sub-system requirements imposed by the spacecraft team are the following:

SCD.1-Maximum Wet Mass	:	The SC shall not exceed a wet mass of 6000 kg.
SCD.2-Maximum Dry Mass	:	The SC dry mass shall not exceed 2550 kg.
SCD.3-Maximum Propellant Mass	:	The SC must carry no more than 2800 kg of propellant.
SCD.4-Attachment Points	:	The SC shall ensure no relative motion of attached modules during launch.
SCD.5-Resonant Frequency	:	The SC shall be less than: Axial 5-100 Hz and Lateral 5-100 Hz
SCD.6-Launch Load Survival	:	The SC shall survive launch loads: Axial 8.5 g and Lateral 3 g
SCD.7-Survivable Temperature	:	Components shall stay within their survivable temperature ranges.
SCD.8-Operational Temperature	:	In operation, components shall stay within their operational temp ranges.
SCD.9-Power Source Lifetime	:	Shall allow for an operation time of at least 3x the required mission time.
SCD.10-Maximum Power	:	The spacecraft shall be operational at no more than 880W.
SCD.11-Power Source Selection	:	The power source shall be specified to the demands of the subsystems.
SCD.12-Delta V and Isp	:	Propulsion system shall provide the delta-V and Isp required to reach orbit.

SCD.13-DSN Compatibility	:	Compatibility with the Deep Space Network for data transmission.
SCD.14-Downlink Data	:	The SC shall downlink all relevant scientific, engineering data, and photographic data.
SCD.15-Frequency Bands	:	Return science data at Ka frequency band operating at 34 GHz.
SCD.16-Tracking Accuracy	:	Doppler: 0.1 mm/s over 60 seconds. Ranging: 1-10 m over 60 to 600 seconds.

IV. Background - Juan Salazar

The majority of our design process is based on the detailed examples and principles described in Space Mission Analysis and Design (SMAD), a practical guide to defining spacecraft mission parameters and requirements. In particular, we referenced the chapters on propulsion, control systems, power systems, communications, and structures and mechanisms design to determine our subsystem-level requirements, select components, and obtain mass and power budgets for the spacecraft. Additionally through interactions with experienced professionals including faculty members at MIT and engineers in industry, we learned and implemented different strategies in our component selection, budget analysis, and other areas we would not have pursued otherwise.

V. Spacecraft Design

A. Spacecraft Configuration - Tao Sevigny

The ELDER spacecraft would be an orbiter in the family of JUNO and is shown in Fig. 2. The spacecraft consists of a main bus and a solar electric propulsion (SEP) kick stage. The overall bus would be a cylindrical body of 3.5 m in height, 1.25 m in diameter, made of metal and metallic honeycomb. The spacecraft with solar panels in stowed configuration is 3.75 m in diameter. Dry mass of the proposed spacecraft is estimated at 835 kg. As such, ELDER would fit on an Atlas V 5 m payload fairing or a Falcon 9 payload fairing, with the standard launch vehicle adapter [1, 2].

The main bus houses the scientific payload. The instrument suite with the mass spectrometer, sub-mm, and camera would be on the nadir-pointing side of the bus. A high-gain antenna (HGA) would sit atop the main bus cylinder and is attached via a gimbal. The bus would hold three tanks: one for oxidizer and two for fuel. The main engine would be located opposite of the HGA, so that the effects of vibration from the thruster minimize changes in performance of the HGA. The advanced Stirling radioisotope generators (ASRG) will be located in the center of the spacecraft above the main thruster. The heat generated by the ASRGs can be used to keep the fuel as operational temperatures. Batteries will be located around the ASRGs and fuel tanks. Having the heavier components like the thruster, ASRGs, and batteries as low as possible in the spacecraft to lower the overall center of gravity of the spacecraft. Having a lower center of gravity reduces the overall bending moment of the launch vehicle interface, and increases the spacecraft launch frequency. The computer will be located above the ASRGs. The payload suite will be placed above the computer, facing in the nadir direction. A sun sensor and two IMUs will be located in the same general area of the payload. Having the computer, payload, HGA, sensors, and power components close to each other reduces the amount of wiring required for the spacecraft [3, Chapter 22, pg. 667].

The SEP kick stage is attached to the main stage by a separable mechanism. There are three hall effect thrusters located at the lowest point of the stage. The stage has one fuel tank for xenon fuel. The stage is powered by five 2 m by 10 m deployable solar panels. The solar panels start folded in the stowed configuration. After launch the solar panels deploy into the operational configuration. The launch vehicle interface is located on the bottom face of the SEP stage.

B. Payload Description and Requirements - Juan Salazar

The spacecraft will carry science instrumentation and fulfill the accommodation and power requirements listed below in Table. 1, along with the pointing requirements. All payload components, beside the magnetometers are housed within the payload module. This module will be located far from the propulsion system to prevent thermal and vibration effects. It will also be located near the data processing and communications components to facilitate the process of electrical routing.

C. Enceladus Operational Modes - Amanda Roberts

Four modes have been planned for operations at Enceladus to maximize data collection for each pass, while ensuring that there is sufficient time planned to relay that data, and keep the batteries charged. The time allocated to each mode is shown in Figure [x].

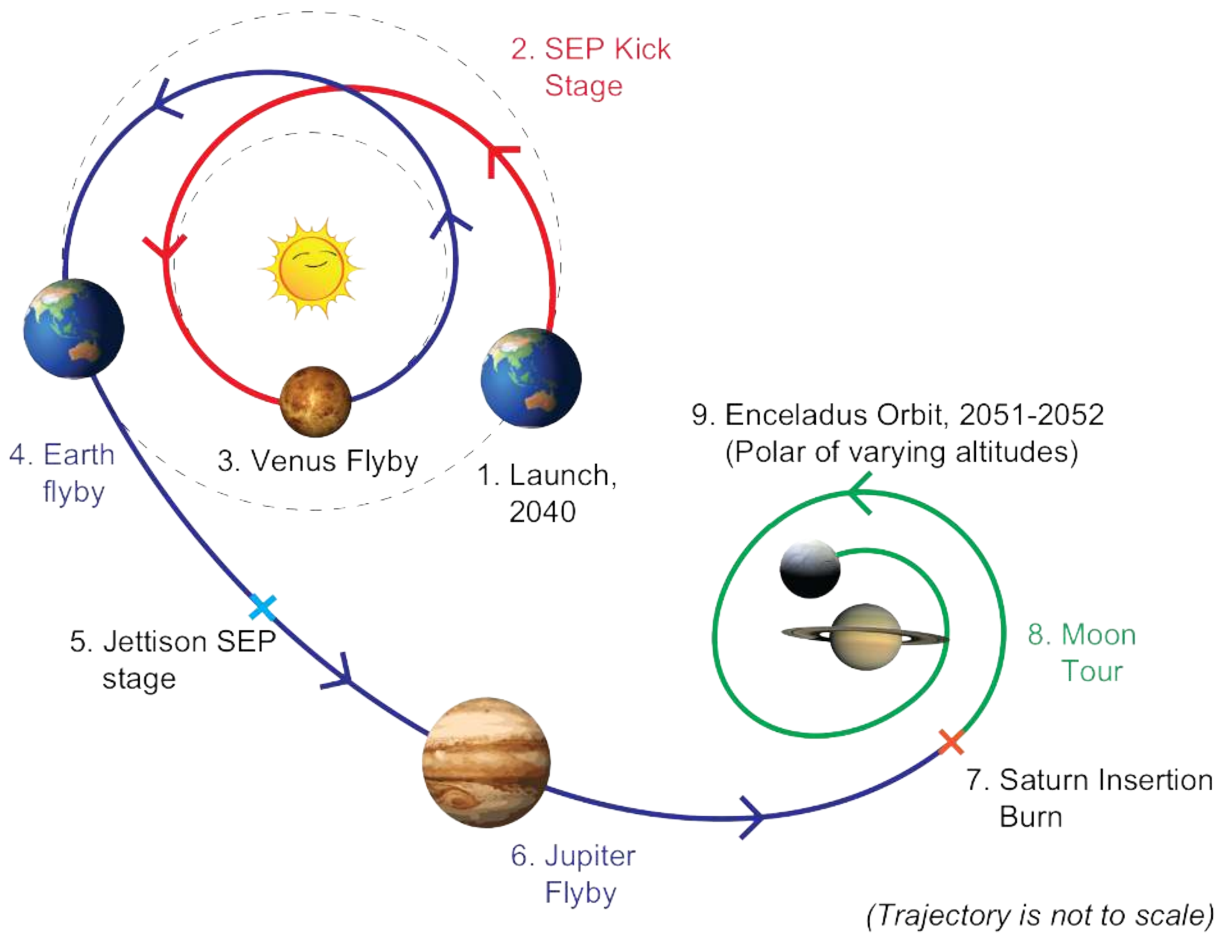


Fig. 1 ELDER mission concept of operations.

Table 1 Payload equipment list. MASPEX is a mass spectrometer, ENIJA is a dust particle mass spectrometer designed for Enceladus, EOA is an organics analyzer, SELFI is a wave spectrometer & radiometer and WAC and NAC are Wide angle and narrow angle cameras.

Instrument	Mass (kg)	Power (W)	Pointing Direction
MASPEX	20.2	45.5	Facing direction of motion
ENJIA	3.5	14.2	Facing direction of motion
EOA	2.5	3.0	Facing direction of motion
SELFI	19.9	43.0	Nadir
WAC & NAC	57.8	55.9	Nadir
Magnetometer	6.0	6.2	Mounted on booms extending from the spacecraft
Laser Altimeter	7.4	16.4	Towards surface
Total	117.3	184.2	

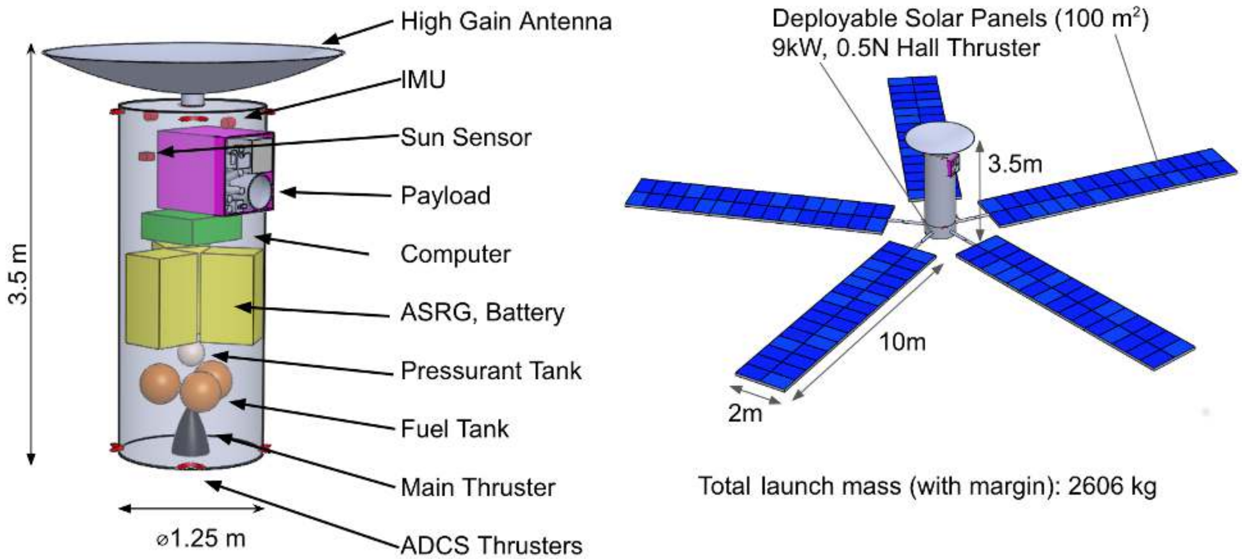


Fig. 2 Overview of spacecraft design. Left, Main stage design showing major components. Right, deployed configuration of spacecraft with Solar Electric Propulsion Kickstage. Magnetometer and mounting boom are hidden from the figures.

1. Data Intensive Mode

The vast majority of data will be collected during passes through Enceladus' southern plumes. The instruments that are operational during a plume pass are determined by the altitude of the pass, which is gradually lowered over the mission lifetime, as seen in Figure 4. A total of twenty orbits are planned for each altitude, with the cameras operational for the higher altitudes, while the mass spectrometer (MASPEX) and other in-situ analysis instruments are only operational for the lower altitudes. The plan of operations for the instrument suite is as recommended by the science team. [include reference?] This mode will be operational for an estimated seven hours/week.

2. Downlink Mode

Science data will be relayed to Earth via a high gain antenna while the spacecraft is within view of Earth. Transmission windows are limited by the schedule of the Deep Space Network, but transmission is planned to take place in 4 hour long bursts, over weekly intervals.

3. Course Adjustment Mode

There are two types of course adjustment: firstly, planned altitude lowering maneuvers. These are planned to occur every 20 orbits, until an altitude of 20km has been reached. Secondly, additional adjustment will take place to correct for orbit perturbations.

4. Coasting, Charging and Surface Mapping Mode

The remainder of time that isn't required for plume data collection, downlink, and course adjustment is spent coasting. During this low power mode, the propulsion and ACDS systems are inactive, and most of the payload instruments are in standby. As this brings the total power consumption below that provided by the three ASRGs, the surplus power generated can be used to recharge the batteries. We take advantage of this long coasting period to operate the laser altimeter and perform surface mapping.

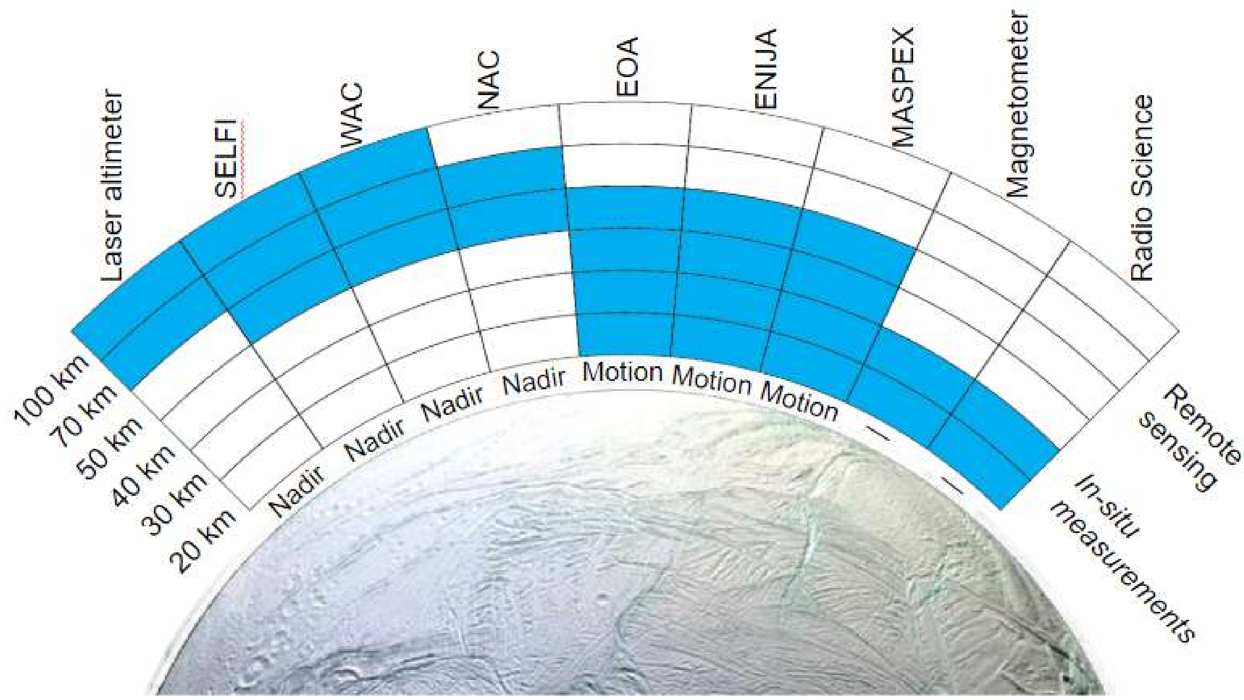


Fig. 3 Operational instruments at each planned orbit altitude. The spacecraft will complete 20 orbits at each altitude.

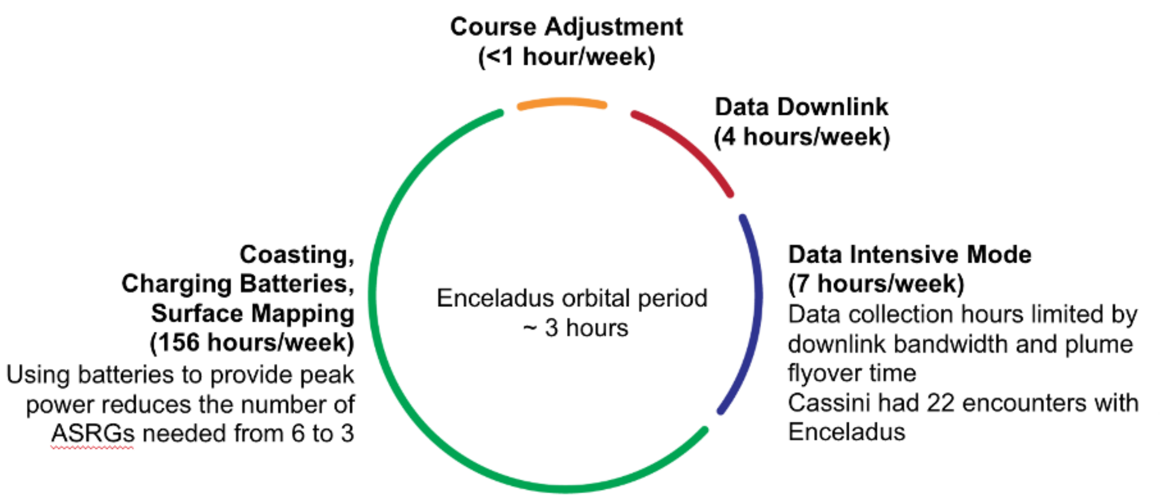


Fig. 4 The four operational modes during primary mission at Enceladus.

D. Main Stage Propulsion System - Devansh Agrawal

The main propulsion system is the propulsion system to be used for the flyby tour, Saturn insertion, moon tour, Enceladus operations, and spacecraft disposal. The trajectory analysis team has estimated the total ΔV requirement, and accounting for a small margin, the spacecraft was designed to deliver a $\Delta V = 2000$ m/s for the maximum expected mass of the vehicle.

For reliability and due to the flight heritage, the Aerojet HiPat 445N thruster was chosen [4]. It is a liquid bi-propellant system using MMH/NTO with a specific impulse of 323 s. Operating the thruster draws about 46 W of power.

The propellant is stored in three titanium tanks (two for the fuel, one for oxidizer), each of 70 cm internal diameter and 1.1 mm wall thickness, allowing the tanks to be pressurized to 20 bar (operational) and 40-bar (burst pressure). Helium is used as a pressurant, stored initially at 275 bar, dropping to 20 bar at end of life. It is stored in a titanium tank of 42 cm diameter, and 9.4 mm wall thickness.

E. SEP Kickstage Propulsion System - Devansh Agrawal

The trajectory chosen requires an electric propulsive burn to lower the orbit from Earth to Venus in the first leg of the mission. The AeroJet BPT-4000 Hall thruster has been chosen as a suitable thruster [5]. Each thruster produces 254 mN of thrust at 2020 s of specific impulse and requires a 400 V, 4.5 kW power supply. To allow for single failure, three thrusters have been specified of which only two will fire at a time, producing 508 mN of thrust and requiring 9 kW of power. The power conditioning units, and Xenon Flow Controllers are also available from Aerojet [5]. While these specific components have been chosen, they have only been tested for a cumulative 5,600 hours, and not for the extended continuous burns that the trajectory requires. This will require further flight qualification, or the design and flight qualification of custom thrusters.

The total ΔV allocated to the SEP stage is 7000 m/s, which sizes the Xenon tanks required. 776 kg of Xenon is needed, which is housed in a single Titanium tank of 39 cm diameter, and 1 mm wall thickness.

F. Communication System - Miguel Wagner

In order to transmit data, our spacecraft must have compatibility with the Deep Space Network. The science instrumentation team requires 500 Megabytes of data for 120 orbits of collection. This data can be compressed at an 8:1 ratio. For the creative communications team, 30 minutes of high definition video is required. This comes out to a total of $1.23 * 10^{11}$ bits of data or 15.4 GB to be transmitted over the course of a year. Due to Deep Space Network scheduling constraints, the spacecraft will transmit in 4 hour bursts in weekly intervals which comes out to a data rate of 164 kbps (kilo-bits per second). The 34 m Antenna will be used to receive data transmissions and will operate at Ka band frequency at 34 GHz. As a result of the link budget analysis outlined below, our spacecraft requires 73.7 dB of signal strength to achieve the desired data rate of 164 kbps. This 73.7 dB metric was the driver for our antenna sizing trade. The curves on the graph in 5 show all possible combinations of antenna size and transmission power that produce our desired data rate. The engineering decision was made to size the antenna at 2.5 m in diameter, which was ideal for storage and assembly. This corresponds to 58.9 W of transmission power required from the spacecraft. In order to achieve this 58.9 W, an efficiency factor of 50%. Three traveling wave tube amplifiers (TWTAs) each supplying 40 W of power to the signal were used to achieve the required 117.8 W.

The link budget equation:

$$\frac{E_b}{N_0} = \frac{P_t G_t G_r}{(4\pi d/\lambda)^2 k T_s R} \tag{1}$$

was used in order to create a link budget analysis. $(4\pi d/\lambda^2)$ is equal to the free space path loss where d is distance at 10 AU and λ is found using our frequency band. The noise is equal to kT_s, and E_b/N_0 represents the Energy per bit to Noise ratio. This must be at least 5 dB to ensure that the data we transmit can be properly decoded. This ratio, the noise factor and the data rate are the drivers for the analysis.

G. Main Stage Power System - Devansh Agrawal

The main stage power supply must be able to provide the power required by all systems operating concurrently. At Saturn, the solar irradiance is about 1% of that at Earth, and therefore the use of solar panels to power systems at Saturn is prohibitively massive and costly. Advanced Stirling Radioisotope Generators (ASRGs) were chosen instead, for their reliability and power per unit weight [6].

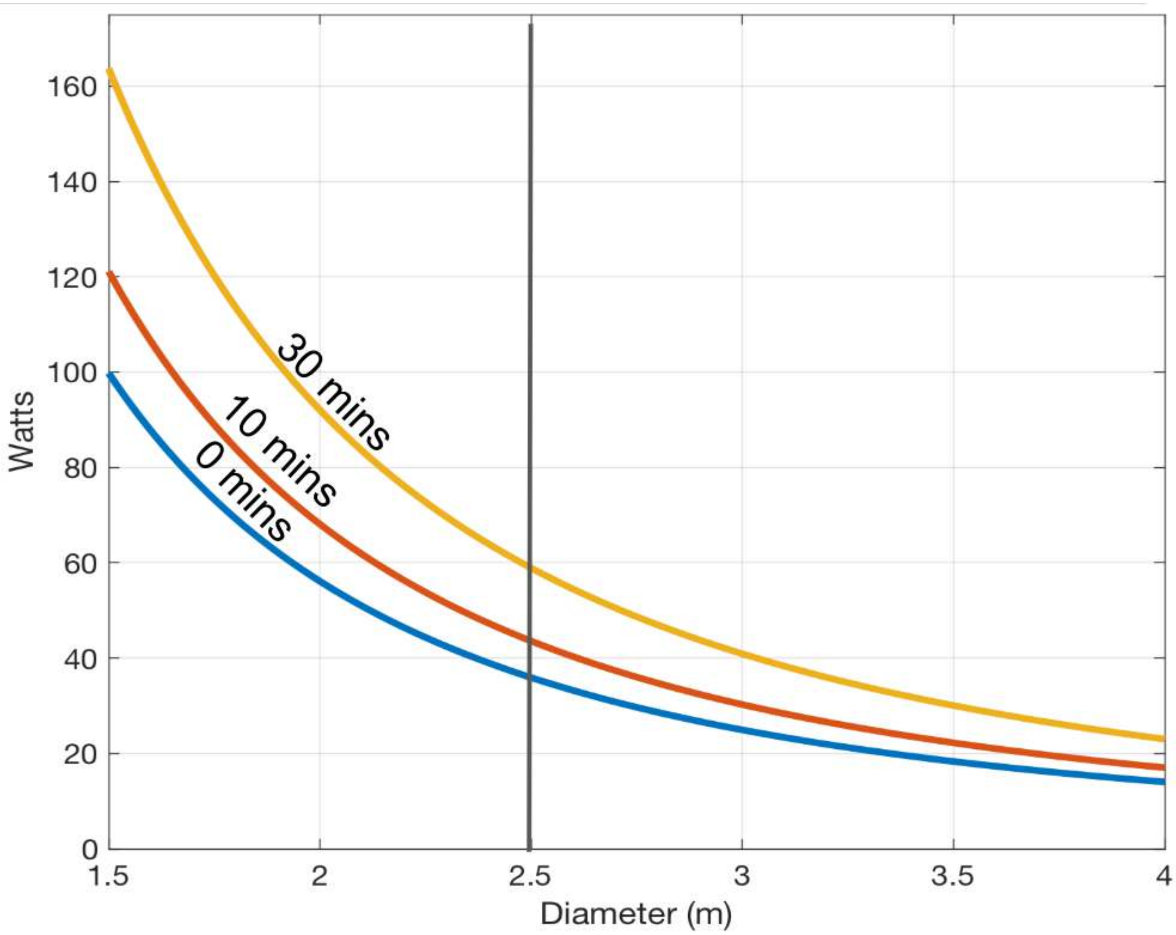


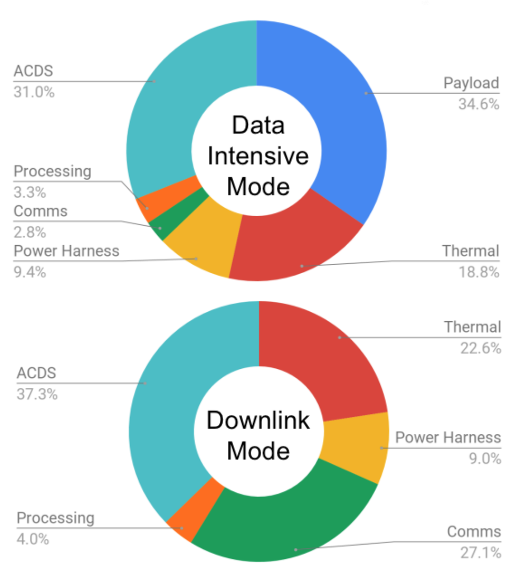
Fig. 5 Curves show combinations of Antenna Diameter and Transmission Power required to transmit science data plus (n) minutes of video at 164 kbps

Table 2 Link Budget Parameters

Parameter	Value	Unit
Antenna Size	2.5	m
Antenna Gain	56	dB
Receiver Gain	78.9	dB
Transmission Power	58.9	dB
Transmission Loss	2	dB
Receiver Loss	1	dB
Free Space Path Loss	306.6	dB
E/N Ratio Required	5	dB
Data Rate	164	kbps
Noise	-214.14	dB

System		Data Intensive Mode (7h)	Downlink Mode (4h)	Trajectory Correction Mode (<1h)	Charging, Surface Mapping (156h)
	Payload	184	0	20	50
	Thermal	100	100	100	100
	Power Harness	50	40	40	25
	Comms	15	120	15	15
	Processing	18	18	18	35
	ACDS	165	165	165	0
	Propulsion	0	0	46	0
Estimated Power Draw (W)		**532**	**443**	**404**	**225**
Margin		30%	30%	30%	30%
Estimate + Margin (W)		**691**	**575**	**525**	**293**

	BOL:	EOL:	
Power Required from ASRGs	394	**317**	W
Power Delivered by 3 ASRGs	420	338	W
Required Battery Capacity (allows 2 weeks of operation without charging)		7658	Wt-hours



Average weekly power = **317 W**

Fig. 6 Bottom up power requirement estimates for each of 4 operational modes, ASRG power supply capacity, and required battery capacity. Pie charts only highlight two operational modes.

The main stage is powered by three ASRGs. Considering a decay over 19 years (launch to mission completion is expected to be 12 years), the end-of-life (EOL) power output of three ASRGs is 338 W (Fig. 6). The maximum expected operational power draw is 691 W during data collection. The extra power required will be supplied by a Lithium-ion battery with a capacity of 7658 Wt-hours. Such a battery allows us to operate the spacecraft for two weeks without any charging and was deemed sufficient to account for safety modes and operational uncertainties. Battery degradation was considered, but since the number of charge cycles was relatively small, on the order of 200 cycles over the mission duration, it hardly reduced the battery capacity. Long term battery storage during the transfer to Saturn was difficult to obtain accurate estimates on, and not considered.

The cost of the spacecraft is likely to be dominated by the cost of the ASRGs. A JPL reviewer suggested the cost of one ASRG is on the order of \$100m, but the marginal cost of adding a second and third ASRG is not nearly as high. For a Discovery class mission, this therefore represents a significant fraction of the mission budget.

H. SEP Kickstage Power System - Devansh Agrawal

The Hall thrusters chosen require 9 kW of continuous power. This high power however is only required with the solar range is less than 1.6 AU. To provide this high power, we have chosen to use GaAs solar panels. Considering the degradation, inefficiencies and pointing capabilities, 100 m^2 of solar panels are required. During the transfer, the power required for nominal spacecraft operation is still provided by the ASRGs but the hall thrusters and associated power conditioning unit and valves are powered by the solar panels. These estimates were performed by using parameters from [3, Chapter 21].

I. ADCS - Tao Sevigny

The Attitude Determination and Control System (ADCS) chosen is a configuration of 16 1 N thrusters to provide small corrections and accomplish the necessary slew rate. The primary driver for the required slew rate was from communications, where the spacecraft would have to slew at a rate to rotate 2 π radians in one Enceladus orbit, one hundred times. SMAD was referred to when choosing a control system [3, Chapter 19, pg. 579-586]. The actuators do not include reaction wheels because the necessary precision can be accomplished by thrusters, and reaction wheels used in deep space have the chance to seize due to environmental effects. ADCS sensors chosen are three sun sensors and two IMUs. These sensors meet the required accuracy needed for our pointing accuracy. The primary driver for the pointing requirement is the payload, which is 1.0 %. SMAD was referred to when choosing sensors [3, Chapter 19, pg. 579-586].

VI. Mass Budget - Tao Sevigny

System		Estimate (kg)	Margin	Estimate + Margin (kg)
	Propulsion	51	33%	68
	Payload	117	29%	152
	Computer	2	30%	3
	Structures	219	20%	263
	Thermal	53	20%	63
	Communications	51	30%	66
	ADCS	6	21%	7
	Power	195	30%	254
Main Dry Sum		**694**	**26%**	**876**
Propellant		**559**	*	**559**
Main Wet Sum		**1253**	**14%**	**1434**

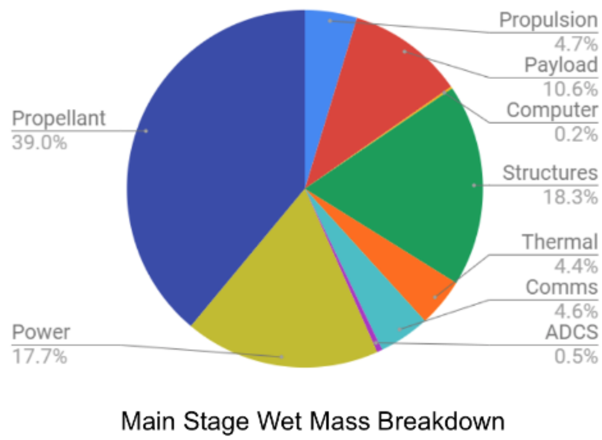


Fig. 7 Bottom-up main stage mass estimates.

System		Estimate (kg)	Margin	Estimate + Margin (kg)
	Propulsion	45	20%	54
	Power	227	33%	303
	Structural	30	30%	39
	SEP Dry Sum	**302**	**31%**	**396**
	Propellant	**776**	*	**776**
Kickstage Wet Sum		**1078**	**9%**	**1172**

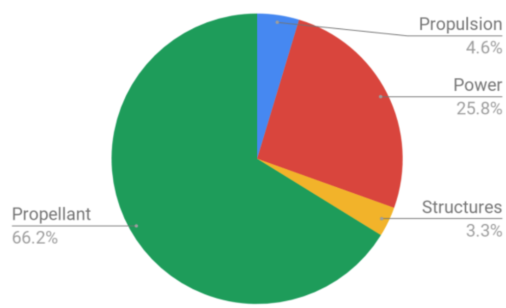


Fig. 8 Bottom-up SEP kickstage mass estimates.

The total wet mass of the ELDER spacecraft is 2606 kg. The current estimate of the main stage propellant mass is 559 kg. The dry mass of the main stage is 876 kg. The current estimate of the SEP kick stage propellant mass is 776 kg. The dry mass of the SEP kick stage is 396 kg. An estimate of each subsystem of the spacecraft was made and a margin was given for each system determined by the maturity of the components and the current phase of the design. The table in Fig. 7 shows the breakdown of the mass by each spacecraft subsystem.

The initial mass estimates were made using SMAD as a reference [3, Chapter 14, pg 422]. The mass estimates were made using the recommended percentages for mass for each subsystem and the maximum launch mass. The mass estimates were refined as the fidelity of the design increased.

VII. Critical System Risks - Logan Kluis

The spacecraft team has identified three main risks that would be catastrophic to the success of the mission. Figure 10 describes both the risk and the planned mitigation for the given risk.

System	Estimate (kg)	Margin	Estimate + Margin (kg)
Main Dry Sum	694	26%	876
Main Propellant	559	*	559
Main Wet Mass	**1253**	**14%**	**1434**
Kickstage Dry Sum	303	31%	396
Kickstage propellant	776	*	776
Kickstage wet mass	**1079**	**9%**	**1172**
Total Launch Mass	**2331**	**12%**	**2606**

Fig. 9 Bottom-up mass estimates for entire spacecraft.

Risk	Risk Mitigation Strategy
Separation - Separating the kick stage from the main stage could fail due to a hardware malfunction, preventing operation of the main engine	**Separation** - Incorporate redundant separation systems and heritage hardware in the design, and conduct extensive ground tests
Data Collection/Transmission - Radiation exposure along the trajectory could degrade instruments, decreasing the possible science	**Data Collection/Transmission** - Ensure the instrumentation suite is protected by sufficient radiation shielding
ASRGs - A limited number of ASRGs results in a high demand that could limit the availability of such technology in the ELDER timeline	**ASRGs** - Plan backup power sources similar to ASRGs that could supply sufficient energy for the mission

Fig. 10 Spacecraft Risks and Mitigation

VIII. Conclusion - Amanda Roberts

According to our analyses, the spacecraft design presented here meets the stated requirements for the defined life detection mission to Enceladus. Our design is well within the maximum wet mass, with a margin of 130% on the spacecraft launch mass. While the ASRG power output decays by 19.5% over the mission life, the system has been sized such that the end of life power output is still sufficient to support the planned operations at Enceladus. The propulsion system provides more than enough ΔV to support the maximum expected spacecraft mass. Further, our calculations show that the communications system can support the transmission of 500 Megabytes of science data, and 30 minutes of high definition video.

The highest priority next step that should be taken is the design of the thermal control system. The current design has not performed the analysis to select thermal protection components to keep all spacecraft components within their operational temperature ranges. Additionally, the performance of finite element vibration and load analysis is essential to ensure that the spacecraft survives launch conditions, and the loads during engine firing. As these steps are performed, it would be necessary to continue refining the power, mass, and link budgets to ensure that the design does not exceed the requirements.

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To see all the calculations we used in designing the spacecraft, see the spreadsheet at https://docs.google.com/spreadsheets/d/1tBLZ3z_fyF5JPZ3pnsFsMWUE6pXYiSfMhrd1mIRAn5g/edit?usp=sharing

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