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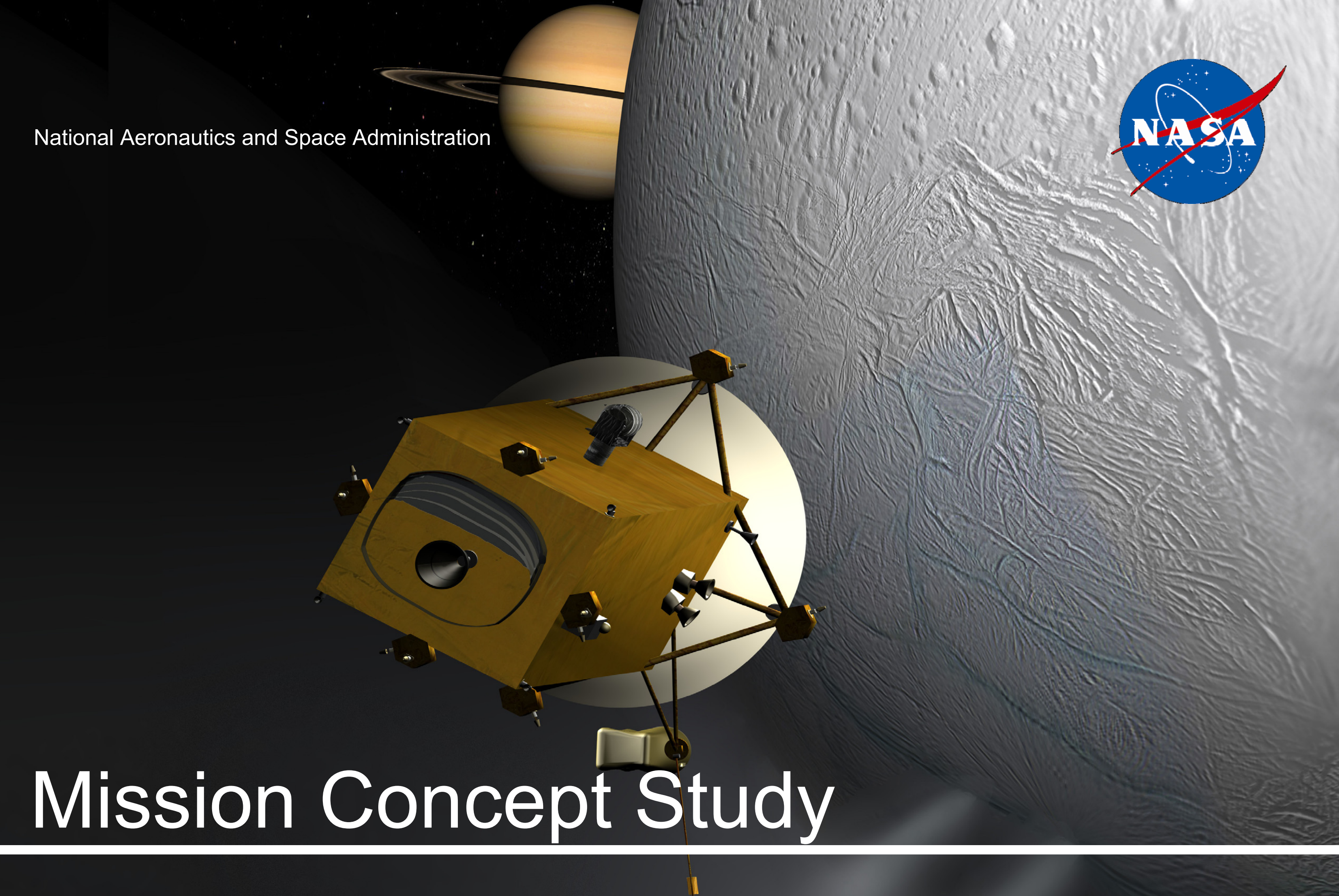

Mission Concept Study

Planetary Science Decadal Survey
Enceladus Orbiter

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May 2010

www.nasa.gov

Data Release, Distribution, and Cost Interpretation Statements

This document is intended to support the SS2012 Planetary Science Decadal Survey.

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Cost estimates described or summarized in this document were generated as part of a preliminary concept study, are model-based, assume a JPL in-house build, and do not constitute a commitment on the part of JPL or Caltech. References to work months, work years, or FTEs generally combine multiple staff grades and experience levels.

Cost reserves for development and operations were included as prescribed by the NASA ground rules for the Planetary Science Decadal Survey. Unadjusted estimate totals and cost reserve allocations would be revised as needed in future more-detailed studies as appropriate for the specific cost-risks for a given mission concept.

Planetary Science Decadal Survey

Mission Concept Study Final Report

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Acknowledgments

This research was carried out at the Jet Propulsion Laboratory, California Institute of Technology, under a contract with the National Aeronautics and Space Administration.

Executive Summary

The Enceladus Orbiter (EO) mission would explore Saturn's moon Enceladus, investigating the following in order of priority:

1. The nature of Enceladus's cryovolcanic activity
2. The internal structure and chemistry of Enceladus
3. The geology of Enceladus
4. The interaction of Enceladus with the Saturn system
5. The surfaces and interiors of Saturn's moons Rhea, Dione, and Tethys
6. Preparation for potential future landing on Enceladus

The Cassini mission discovered Enceladus's plumes and has begun characterizing them. This Enceladus orbiter would provide more extensive characterization, including the following:

- Improved measurements of plume gas and dust composition, and higher-resolution thermal and visible imaging of the plume vents, including conditions at the plume source
- Measurements of tidal flexing, magnetic induction, and static gravity, topography, and heat flow, as well as improved plume and dust composition measurements
- Global high-resolution mapping of surface features

The above would be achieved through a combination of improved instrumentation, much slower south-polar flybys than are possible with Cassini, and an extended period in Enceladus orbit, which would allow uniform, near-global remote-sensing coverage and probing of the interior structure. Instrumentation would include a medium-angle camera (MAC), thermal imaging radiometer, mass spectrometer (MS), dust analyzer, magnetometer, and radio science.

Key operational phases for the mission would include

1. Interplanetary cruise, which would include two Venus flybys and two Earth flybys. This phase would begin with launch and end 8.5 years later with Saturn orbit insertion (SOI).
2. Saturn moon tour, which would include 3 Titan flybys, 15 Rhea flybys, 10 Dione flybys, 12 Tethys flybys, 12 engineering flybys of Enceladus, and 10 scientific flybys of Enceladus. This phase would begin with SOI and end 3.5 years later with Enceladus orbit insertion (EOI).
3. Enceladus orbit, which would include all scientific, mapping orbits of Enceladus. This phase would be 12 months long for the baseline option (Option 1) and 6 months long for a shorter option (Option 2).

The flight system would be nearly identical for both options, with small differences due to the shorter duration for Option 2. The three-axis-stabilized spacecraft would be dual-string with cold spares and include a fixed, 3-meter, high-gain antenna (HGA). Three Advanced Stirling Radioisotope Generators (ASRGs) would provide power. Conventional bipropellant chemical propulsion would provide delta-V. The magnetometer, MS, and dust analyzer would be mounted away from the spacecraft bus and beyond the edge of the HGA by means of a fixed boom.

Aside from ASRGs, no new technology is required for this mission.

The top risks identified for this mission include the following:

1. The planetary protection (PP) cost estimates were based on the assumption that analysis would provide sufficient data to prove that the mission meets the requirements. The current assumption is that if it could be shown that the orbiter could meet the required low probability of impacting a special region (assumed to be 55–90 degrees latitude in the south-polar region), the orbiter would

not need to be sterilized. The launch environment is relatively contaminated and could affect the cleanliness of the spacecraft. The approach is to show by analysis that PP requirements could be met without the use of a biobarrier around the spacecraft before and during launch. If this could not be shown by analysis, there would be implications for mass and cost.

2. Component reliability could be an issue because of the 13-year mission duration. Full redundancy would be the mitigation strategy.
3. Plutonium availability for the ASRGs would be critical. The mission could not fly without an adequate supply. The assumption is that plutonium sufficient for three ARSGs would be available.

1. Scientific Objectives

Science Questions and Objectives

Saturn's moon Enceladus, with its remarkable ongoing cryovolcanic activity, including plumes that deliver samples from a potentially habitable subsurface environment, is a compelling target for future exploration. The Cassini mission discovered the plumes and has begun characterizing them. Among the more salient discoveries are that (1) the plumes originate from the "tiger stripe" fractures of the southern pole (Figure 1-1); (2) the plumes are persistent over time scales of years; (3) the tiger stripe fractures are relatively warm (Figure 1-2); (4) the plume particles create Saturn's E-ring (Figure 1-3); and (5) the plume source contains the basic necessities for biotic activity, including the elements C, H, O, N, warmth, and quite likely liquid H_2O (Figure 1-4).

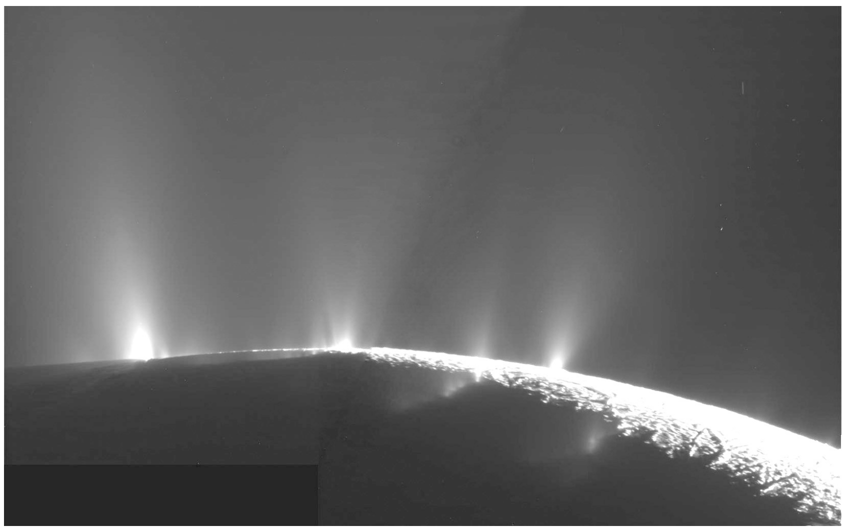

Figure 1-1. Cassini imaging science subsystem (ISS) image shows multiple simultaneous plumes coming from extended "tiger stripe" fractures.

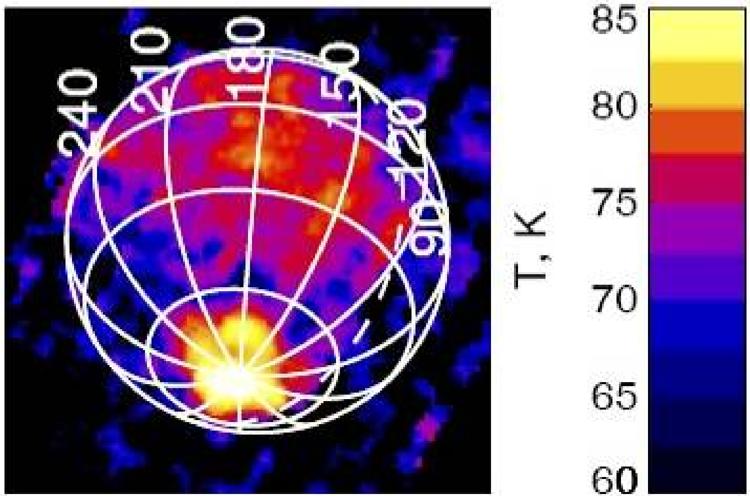

Figure 1-2. Cassini composite infrared spectrometer (CIRS) 9- to16-μm image of Enceladus shows enhanced thermal emission from the south-polar region.

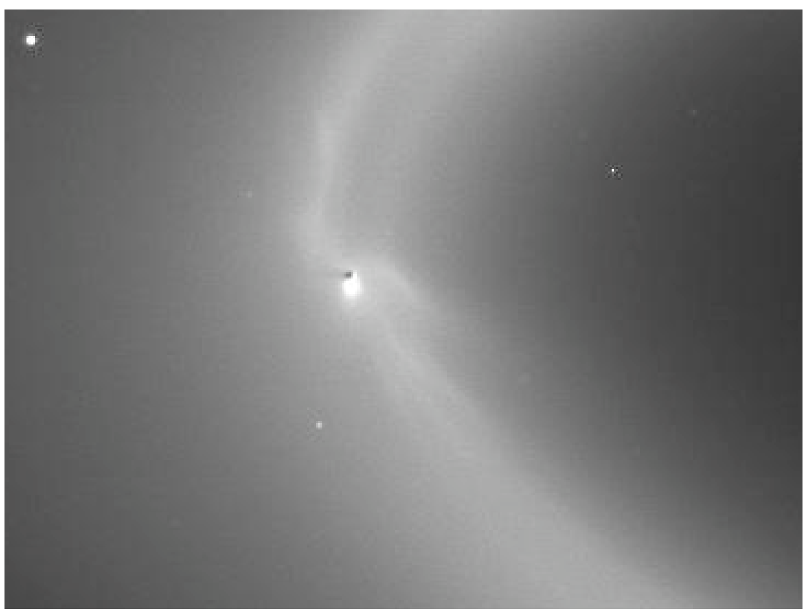

Figure 1-3. Cassini ISS image shows interaction between Enceladus and the E-ring.

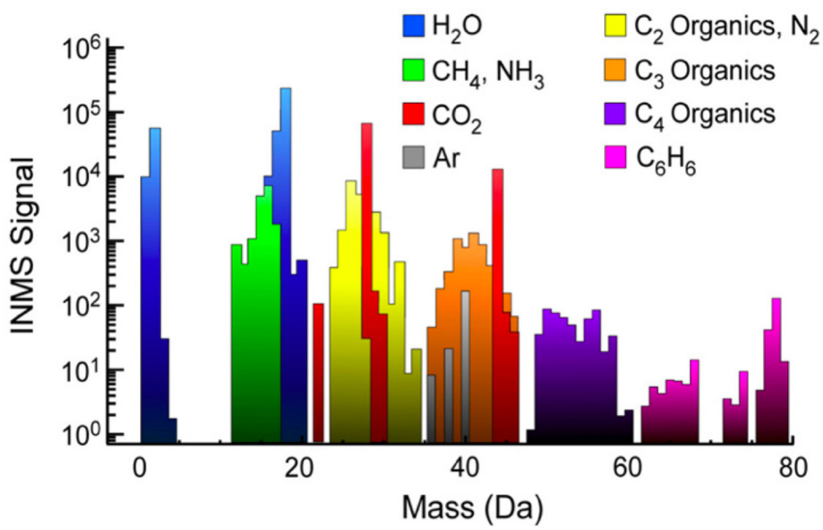

Figure 1-4. Cassini ion and neutral mass spectrometer (INMS) plume mass spectrum shows necessary elements for life, including a wide range of organic compounds.

The specific high-level science objectives for this mission would include the following, in priority order:

1. Investigate the nature of Enceladus's cryovolcanic activity, including the physical conditions at the plume source; the chemistry of the plume source; the possible presence of biological activity; plume dynamics and mass loss rates; and the origin of the south-polar surface features.
2. Investigate the internal structure and chemistry of Enceladus, including its internal structure; the presence, physics, and chemistry of the subsurface ocean; tidal dissipation rates and mechanisms; and chemical clues to Enceladus's origin and evolution.
3. Investigate the geology of Enceladus: The nature and origin of its geologic features and its geologic history.
4. Investigate the interaction of Enceladus with the Saturn system, including its plasma torus and neutral clouds; the E-ring; and its modification of its own surface and that of its neighboring satellites.
5. Pursue other satellite science, specifically investigations of the surfaces and interiors of Rhea, Dione, and Tethys.
6. Prepare for follow-on missions, specifically establishing the characteristics of potential future landing sites on Enceladus.

While Cassini will continue to make major contributions to many of these objectives, it is hampered by the limitations of its instrumentation and orbit. For instance, the low resolution and mass range of Cassini's spectrometer prevent full characterization of plume gases, and Cassini's brief flybys do not allow detailed characterization of Enceladus's gravity field, or measurement of tidal flexing, or extensive high-resolution imaging.

The Enceladus orbiter would advance these objectives beyond what is possible with Cassini by enabling (1) improved measurements of plume gas and dust composition, and higher-resolution thermal and visible imaging of the plume vents to better understand the plume, including conditions at the plume source; (2) measurements of tidal flexing, magnetic induction, and static gravity, topography, and heat flow, as well as improved plume and dust composition measurements, to probe interior structure, particularly looking for and investigating the nature of a possible subsurface ocean; and (3) global high-resolution mapping of surface features to provide a comprehensive understanding of Enceladus's geology. These capabilities would be provided through a combination of improved instrumentation, much slower south-polar flybys prior to EOI than are possible with Cassini, and an extended period in Enceladus orbit, which would allow for uniform, near-global remote-sensing coverage and probing of the interior structure.

The instrumentation used to achieve these objectives would include a medium-angle camera (MAC), thermal imaging radiometer, mass spectrometer (MS), dust analyzer, magnetometer, and radio science. The science traceability matrix (STM) (Table 1-1) shows the objectives that would drive the required measurements, the instrumentation, and the mission design.

Science Traceability

The STM (Table 1-1), provided for this study by the EO science team, identifies the major objectives and resource requirements that drove the study, and subdivides the major science objectives into subsidiary objectives.

Objectives concerning *physical conditions at the plume source* would include measurements of temperature and heat flow and chemical equilibria derived from measurement of isotopes, gases, and composition of particles in the plume. The *chemistry of the plume source* and the *presence of biological activity* objectives would rely strongly on the quality of the MS measurements. Objectives concerning *plume dynamics and mass loss rates* and *origin of the south-pole features* would utilize imaging and other coordinated measurements (e.g., dust, mass spectra). All of the observations in the *nature of Enceladus's cryovolcanic activity* science objective would require over-flights of the tiger stripe region.

Most of the observations associated with the *internal structure and chemistry of Enceladus* and with the *geology of Enceladus* would require near-circular orbits and precision navigation to investigate the geophysics of Enceladus, particularly induced magnetic fields associated with conductive oceans and deformation and heating associated with tides. The exceptions are observations of *chemical clues to Enceladus's origin* that would require sampling the plume.

System interaction observations would be addressed in all mission phases. Measurements of the E-ring would provide additional proxy measurements of activity and products of the Enceladus plumes.

Other satellite science would covers opportunities to investigate the geology, chemistry (through MS measurements of exospheric atmospheres), and interior structure of Rhea, Dione, and Tethys during the many slow flybys of these moons in the pumpdown phase of the mission. These objectives would not be mission drivers, but would provide ample opportunity for serendipitous science.

Preparation for follow-on missions would include the detailed surface characterization needed to design potential future Enceladus landers. Landing on the moon would be challenging due to its rugged, small-scale topography and possible deep-surface coating of low-density plume fallout.

Table 1-1. Science Traceability Matrix

#	Science Objective/ Subobjective	Measurement Objectives	Measurements	Instruments	Functional Requirements
1	**Nature of Enceladus's Cryovolcanic Activity**				
1.1	Physical conditions at the plume source	Topography & stratigraphy; Thermal output; vent shape; surface strength; surface roughness; subsurface structure of tiger stripes; cavern size; subsurface lake; particle size distribution and speed; ice temperature	1-m imaging of vent sources, full imaging of active region at 5-m resolution, 3 colors; vis stereo 5/10 m & 2 angles; thermal map 100-m resolution of active region, plume particle sizes and compositions	MAC, thermal imaging radiometer, dust analyzer, MS	10 slow passes over south pole region. Flyby altitudes ranging from 200 km for regional mapping to 50 km for 1-m imaging
1.2	Chemistry of the plume source	Chemical inventory of plume gas and dust species; chemical equilibria; isotopic ratios	Mass spectra 0–500 dalton & 20000/10000/1000 res; dust size 0.01 to 1000 micron & angle dist to 10 degrees & composition	MS, dust analyzer	10 slow passes over south pole region. Some at ~1 km/sec as dust analyzer will not work well at slower speeds
1.3	Presence of biological activity	Organic molecules inventory to high masses	Mass spectra 0–500 dalton & 20000/10000/1000 res, dust composition	MS, dust analyzer	Multiple flybys of south pole region
1.4	Plume dynamics and mass loss rates	Plume structure, ejection rates; particle size vertical structure; particle velocities; time variability (density, particle size, velocity; composition)	Dust size, frequency, velocity; gas density, velocity; high-phase-angle imaging	MAC, MS, dust analyzer	Flybys at multiple altitudes (200/100/50 km). Need high-phase-angle (>150 deg) multicolor imaging at correct ranges for plume imaging. Several flybys with identical geometry to study time variability

#	Science Objective/ Subobjective	Measurement Objectives	Measurements	Instruments	Functional Requirements
1.5	Origin of south-polar surface features	Topography & stratigraphy, temperature distribution of active features	Vis regional map 5-m imaging; stereo, 5- to 100-micron multiband thermal imaging at 100-m resolution of selected areas	MAC, thermal imaging radiometer	Stereo geometries and a range of lighting conditions. Flyby altitudes ranging from 200 km for regional mapping to 50 km for 1-m imaging
2	**Internal Structure and Chemistry of Enceladus**				
2.1	Internal structure	Static gravity, potential Love numbers, magnetic field	Static gravity and magnetic fields to order 12 and precision 10^{-9} m/s^{2,} magnetic field to 0.1 nT. Tidal flexing	Radio science, magnetometer, imaging	65-deg-inclination near-circular orbit (200 km altitude); frequent tracking. Geometry suited for crossover analyses (hi-inclination, near-circular orbit preferred); constant tracking. South-polar flybys to fill in the coverage
2.2	Presence, physics, and chemistry of the ocean	Potential Love numbers, magnetic induction, plume chemistry	Tidal flexing as in (6); Magnetic field orientation to 0.1 degree & 0.1 nT.	Radio science, magnetometer, MS, dust analyzer	At least one week of magnetometer data from low orbit
2.3	Tidal dissipation rates and mechanisms	Long-wavelength global thermal emission, bolometric albedos	Global map 1000 m & phase angles 0, 10, 20, 50, 70; global thermal map 500 m, 5 to 100 microns 6 bands	MAC, thermal imaging radiometer	65-degree-inclination orbit for near-global coverage, south-polar flybys to fill in the coverage
2.4	Chemical clues to Enceladus's origin and evolution	Isotopic and elemental analysis of plume gases and dust grains	Mass spectra 0–500 dalton & 20000/10000/1000 res, dust composition	MS, dust analyzer	Multiple plume fly-throughs
3	**Geology of Enceladus**				
3.1	Nature and origin of geological features and geologic history	Geology, topography, stratigraphy	Vis global map, 20 m & two phase angles, 5 m of selected areas; topography from stereo; static gravity to order 12	MAC, radio science	65-degree inclination orbit for near-global coverage, south- and north-polar flybys to fill in the rest of the surface. Altitudes as low as 100 km for 5-m imaging of selected targets

#	Science Objective/ Subobjective	Measurement Objectives	Measurements	Instruments	Functional Requirements
4	**System Interaction**				
4.1	Plasma and neutral clouds	Spatial distribution, composition, and time variability of neutral clouds, correlation with plume activity	Mass spectroscopy sensitive enough to measure the neutral clouds distant from Enceladus	MS, MAC to monitor plume activity	Multiple passes through the neutral clouds—will happen during the pumpdown/leveraging phase
4.2	E-ring	Variation, composition, and relation to Enceladus activity	Dust density, direction, composition; density variability (image in forward scattering)	Dust analyzer, MAC to monitor plume activity and E-ring structure	Multiple passes through the E-ring—will happen during the pumpdown/leveraging phase
4.3	Modification of the surfaces of Enceladus and the other satellites	Relative ages, surface texture on meter and centimeter scales, exogenic coatings, exogenic impact and ion environment; molecular lifetimes	High-resolution imaging, thermal mapping of the icy satellites	Dust analyzer, thermal imaging radiometer, MAC, MS	Close flybys of Rhea, Dione, and Tethys: 50 km to allow 1-m imaging
5	**Other Satellite Science**				
5.1	Surfaces and interiors of Rhea, Dione, and Tethys	Geology and evolution of surfaces of neighboring satellites, shapes, gravity fields	High-resolution imaging, thermal mapping, static gravity, magnetic properties, exospheric composition of the icy satellites	MAC, magnetometer, radio science, MS	Multiple flybys of each object
6	**Preparation for Follow-on Missions**				
6.1	Nature of potential landing sites	Topography, surface texture, thermal inertia	1-m stereo imaging of selected sites, thermal measurements at multiple times of day	MAC, thermal imaging radiometer	

This matrix describes the linkages between science objectives and how they are achieved. Note that functional requirements are requirements placed by science on the mission concept (e.g., requirements on the spacecraft, trajectory, mission architecture, etc.)

2. High-Level Mission Concept

Overview

As part of NASA's support to the National Research Council (NRC) and its current Planetary Decadal Survey, JPL was assigned the task of developing a core mission and flight system architecture suitable to perform a scientifically viable Enceladus Orbiter (EO) mission responsive to STM requirements formulated by NASA's Science Panel.

The NASA Science Panel was specifically interested in a mission that would fit within NASA's New Frontiers proposal constraints. Architecture trade-space analyses as well as detailed point designs were to be performed by JPL. To meet this study's needs, the work was divided into two phases, both involving concurrent engineering teams at JPL: (1) an initial examination of the architecture trade space by JPL's Rapid Mission Architecture (RMA) team; and (2) detailed conceptual designs and cost estimates of total mission architectures by JPL's Advanced Projects Design Team (Team X). This arrangement allowed for a free-ranging exploration of possible mission architectures by a team of RMA specialists chosen for their relevant knowledge to the problem, while leveraging the efficiency and experience of Team X with regard to the spacecraft portions of the mission—areas routinely handled by this team. This work was done in close coordination with the Decadal Survey's Satellite Subpanel, with panel members providing active guidance on the design process and decisions to JPL's two study teams.

More than two dozen architectures considered in the RMA study included flyby spacecraft, simple and flagship-class orbiters, landers, and plume sample-return missions. The conclusion was that a relatively simple Enceladus orbiter could provide compelling science for relatively low cost and risk, and this mission architecture was chosen for the more detailed Team X study reported here.

The selected architecture involves a flight system with conventional bipropellant propulsion, three-axis stabilization using reaction wheels for fine pointing control, a 3-meter, fixed X/Ka-band HGA, three ASRGs, and a flash memory card sufficient to capture large amounts of data in short periods of time (e.g., during flybys). The payload would consist of an imaging camera (MAC), thermal mapping radiometer, MS, dust analyzer, and magnetometer. The MS and dust analyzer would be oriented in the direction of the velocity vector, and other instruments would be nadir-pointed. The difference between the two options studied would be the length of time spent in Enceladus orbit: 12 months for Option 1 (baseline) and 6 months for Option 2. The flight system design and payload would be the same for both options.

The MAC would primarily be used to image the Enceladus surface and plumes. It would also be used for optical navigation. The thermal imaging instrument would be used to map the thermal output of the tiger stripe region, measure global heat flow, and measure the thermal inertia of the surface. The MS would be used to measure the composition of the plumes during plume fly-throughs. The dust analyzer would be used to measure size and velocity distribution, and composition, of dust in the Enceladus plume as well as the Saturn E-ring. The magnetometer would be used to measure the intrinsic magnetic field of Enceladus to order 12 and thus help in the search for subsurface oceans or lakes via the weak magnetic induction signature resulting from Enceladus's eccentric orbit through Saturn's magnetic field, as well as the magnetospheric interaction of Enceladus and the other icy satellites. Radio science would use the spacecraft's telecom subsystem to measure static and dynamic gravity field components and thereby infer tidal flexing and internal mass distribution. The mission involves three key operational phases:

1. Interplanetary cruise, which would include two Venus flybys and two Earth flybys. This phase would begin with launch and end 8.5 years later with Saturn orbit insertion (SOI).
2. Saturn moon tour, which would include three Titan flybys, 15 Rhea flybys, 10 Dione flybys, 12 Tethys flybys, 12 engineering flybys of Enceladus, and 10 scientific flybys of Enceladus. The phase would begin after SOI and end 3.5 years later with Enceladus orbit insertion (EOI). The purpose of the tour would be to reduce the spacecraft's energy in its orbit about Saturn. The tour would save the mission over 3000 m/s of propellant compared to direct Enceladus orbit insertion.

3. Enceladus orbit, which would include all scientific, mapping orbits of Enceladus. The phase would begin after EOI and end 12 months later for the baseline option (Option 1) and 6 months after EOI for Option 2.

Enceladus is very close to Saturn; thus any polar orbit would likely be unstable. Orbital parameters would vary substantially from orbit to orbit, but the variation would repeat over the course of ~1.37 days, namely, the orbital period of Enceladus about Saturn. For Option 1, the average orbit would have a 6-hour period, 62-degree inclination, and 184-km altitude for the first 6 months. The altitude would range between 101 km and 267 km, and the orbit would precess by a full 360 degrees in 1.37 days, making it very easy to map the entire planet. After the first 6 months, the spacecraft would transfer into and remain in a lower orbit for the last 6 months. The lower orbit would have a period of 4 hours and an average altitude of 81 km, and its altitude could go as low as 30 km. For Option 2, the spacecraft would have one orbit for all 6 months, similar to the orbit during the first 6 months of Option 1.

During the Saturn tour, there would be one 4-hour DSN pass per day. After EOI, there would be one 8-hour DSN pass per day. Due to limited energy available from the ASRGs, downlink would be for 5 of the 8 hours. (See Key Trades below.) The ground system operations strategy would match the downlink capacity, which would be sufficient to return all science and housekeeping data with margin.

Requirements, constraints, and key parameters for the mission, and flight system are summarized in Table 2-1.

Table 2-1. Mission Requirements and Constraints

Requirements/Constraints	Origin/Comments
Systems Engineering	
Launch in 2023; arrival in the Saturn system in 2031.	RMA selected architecture.
Accommodate JPL Design Principles [1].	
Accommodate NASA planetary protection requirements.	
Class B mission	Consistent with New Frontiers.
Mission design:	
Baseline 13-year mission duration; 3.5 years in Saturn system; 12 months in orbit at Enceladus.	RMA selected architecture.
Option for 12.5-year mission duration; 3.5 years in Saturn system; 6 months in orbit at Enceladus.	
Total postlaunch delta-V of 2685/2660 m/sec for longer/shorter missions.	
Polar flybys available prior to Enceladus orbit insertion, and perhaps from Enceladus orbit after the prime mapping phase.	Fly-throughs important for temporal behavior of plumes, plume sampling. Over-flight important for measuring thermal environment of tiger stripe region.
10 science flybys of Enceladus at the end of the flyby sequence, with ~10 days between flybys.	Flyby cadence chosen to minimize operational complexity.
Orbits at inclination of 62 degrees.	Used for mapping rest of Enceladus: Higher inclinations are not stable. Provides oblique observations of plumes. Phase producing highest data volume.
For Option 1, Enceladus orbits at altitudes ranging from 101 km to 267 km with an average of 184 km for 6 months, followed by 6 months of orbits with average altitude of 81 km, with range possibly going as low as 30 km.	RMA-selected architecture.

Requirements/Constraints	Origin/Comments
For Option 2, Enceladus orbits at altitudes ranging from 101 km to 267 km with an average of 184 km for 6 months	RMA-selected architecture.
Payload	
Accommodate a Science-Panel-defined payload that includes a MAC, thermal imaging radiometer, MS, dust analyzer, magnetometer, and radio science.	Affects flight system pointing, instrument duty cycles, coverage strategy, telecom, and mission design.
Payload data compression is not to exceed a factor of 3 to 5 (i.e., "lossless").	Affects instrument internal design and/or flight system command and data handling subsystem.
Operations	
Accommodate payload data acquisition requirements. For instance, nadir-pointed imaging and dust analyzer and MS in the ram direction during plume fly-throughs. HGA not articulated: radio science/downlink and remote sensing are done sequentially.	Affects flight system configuration and operations strategy.
Flight System	
Radioisotope power system (RPS)	This mission meets Decadal Survey requirement [2] that RPS be enabling.
238 Pu assumed to be available as needed.	
Use ASRGs as opposed to MMRTGs.	Minimize required quantity of Pu.
Use no more than 3 ASRGs.	Minimize cost.
Design for full science after loss of half of an ASRG.	Assume that the other half provides 45% of the power for that ASRG, per the NASA-provided "ASRG Functional Description" document (ID# 912IC002144) [3].
ASRG lifetime 16 years after launch: assumes fueling one year before launch, but still within 17 years total lifetime.	Per agreement with NASA-HQ Decadal Survey POC.
Conventional biprop propulsion	RMA-selected architecture.
Ka-band data downlink.	Decadal Survey ground rule [2].
Downlink data rate ~78 kbps before overhead or compression to an array of two 34-m antennas.	Selected capability
Ground System	
Arraying two 34-m antennas for science return permitted.	Per agreement with NASA-HQ Decadal Survey POC.
70-m and/or arrayed 34-m antennas permitted for critical event coverage.	Decadal Survey ground rule [2].

Concept Maturity Level

Table 2-2 summarizes the NASA definitions for concept maturity levels (CMLs). Following the completion of this study, all options are considered to be at CML 4. The architectures studied were defined at the assembly level and estimated for mass, power, data volume, link rate, and cost using JPL's institutionally endorsed design and cost tools. Risks were also compiled as part of this study.

Table 2-2. Concept Maturity Level Definitions

Concept Maturity Level	Definition	Attributes
CML 6	Final Implementation Concept	Requirements trace and schedule to subsystem level, grassroots cost, V&V approach for key areas
CML 5	Initial Implementation Concept	Detailed science traceability, defined relationships, and dependencies: partnering, heritage, technology, key risks and mitigations, system make/buy
CML 4	Preferred Design Point	Point design to subsystem level mass, power, performance, cost, risk
CML 3	Trade Space	Architectures and objectives trade space evaluated for cost, risk, performance
CML 2	Initial Feasibility	Physics works, ballpark mass and cost
CML 1	Cocktail Napkin	Defined objectives and approaches, basic architecture concept

Technology Maturity

All flight system elements are currently at technology readiness level (TRL) 6 or above: No technology development would be required for the flight system. The TRL of each instrument can be seen in Table 3-1, Instrument Summary. No technology development would be required for the instruments—only engineering development related to mission life. Instrument maturity and heritage support low mission cost-risk.

Under the Decadal Survey guidelines, ASRGs are presumed to be available for the purpose of this study; therefore, no technology development is assumed. The subsystem and overall spacecraft design would be high heritage and based on technology that has already flown. The telecommunications subsystem would make use of a universal space transponder (UST) that, while not currently developed, is planned for flight demonstration on the 2016 ExoMars mission. If the UST has is not developed by the time of this mission, it would be possible to obtain similar functionality with minor modifications to existing small deep space transponders (SDSTs), so the UST would not be a required new development for this mission.

Key Trades

Instrument Trades

Because this study focused on the minimal useful payload, many additional instruments that would add valuable science were not included. For instance, a near-infrared spectrometer would allow remote sensing of plume particle sizes and surface composition, but was considered lower-priority because Enceladus's near-infrared spectrum is dominated by water ice and few additional components have been seen by the Cassini visual and infrared mapping spectrometer (VIMS) instrument. In addition, plume particle sizes could be probed directly by the dust analyzer. An ultraviolet spectrometer would allow remote determination of plume water vapor abundance and distribution via stellar occultations, but many of the same measurements could be made by the MS. A gas chromatograph would provide more detailed information on the nature of organic molecules in the plume, but its mass and power requirements were considered prohibitive. A ground-penetrating radar would also provide unique and valuable information on the subsurface structure, but at prohibitive cost in mass and power.

Particular consideration was given to the inclusion of a laser altimeter, which could provide global topography and also potentially measure tidal flexing of the shape of Enceladus. However, we concluded that the likely amplitude of tidal flexing on Enceladus (~3 meters), would be close to the limit of precision of the altimeter. While an altimeter could help with orbit determination and thus contribute to the measurement of the tidal flexing of Enceladus's gravity field, the gravity flexing could be measured with sufficient accuracy by radio Doppler tracking alone. Global topography could also be measured, though with less global accuracy, by stereo imaging and limb imaging. The laser altimeter was thus not included in the payload.

Energy Balance Trade

A key trade focused on achieving a positive energy balance over 24 hours in Enceladus orbit. The result is as follows: Science was allocated 8 hours to acquire up to 1 Gbit of data; this interval and data volume would be adequate to meet science objectives. A 3-meter X/Ka-band HGA would be used to downlink data at 65 kbps to two 34-meter DSN antennas. The HGA would be used in dual-frequency (X/Ka) mode to support radio science. A positive energy balance would be achieved by limiting downlink to 5 hours, leaving 11 hours for battery recharge. Five hours of downlink at 65 kbps would be long enough to return science and housekeeping data, with margin. So, the baseline allocations would be 8 hours for science, 5 hours for downlink, and 11 hours for recharge in a 24-hour interval.

During this trade, a 4-meter antenna was also considered. With a larger antenna, the telecom subsystem could operate at lower power and achieve the same data rate for a longer interval (e.g., 8 hours) or operate at the same power and achieve a higher data rate for the same 5 hours. In either case, a larger data volume could be returned. Another option was to use the 3-meter HGA in single-frequency mode (Ka-band only) to reduce power and allow for an increased downlink interval. While a single band would not be ideal, radio science would still be possible. Neither of these alternatives was adopted as the baseline, but both could be considered in the future if needed.

3. Technical Overview

Instrument Payload Description

The instrument payload for the EO mission, summarized in Table 3-1, would consist of an imaging camera (MAC), thermal imaging radiometer, MS, dust analyzer, and magnetometer. The MS and dust analyzer would be oriented in the direction of the velocity vector, and other instruments would be nadir-pointed. The instrument characteristics were obtained from the Enceladus Orbiter RMA report [4]. Specific parameters for each instrument are detailed in Tables 3-2 through 3-6. Payload mass and power estimates are given in Table 3-7.

The MAC would be used primarily to image the Enceladus surface and plumes in order to map surface geology and measure the sources and variability of the plumes. It would also be used for optical navigation. The MAC would operate in pushbroom mode, imaging the planet surface in strips that could be stitched together to generate a global map of the Enceladus surface. The MAC has high sensitivity in three to four color bands to enable high-resolution imaging of the tiger stripe region. The MAC would be analogous to the New Horizons multispectral visible imaging camera (MVIC). Like MVIC, it would also include a framing capability for optical navigation.

The thermal imaging radiometer instrument would be used to map the thermal output of the tiger stripe region, measure global heat flow, and measure the thermal inertia of the surface. The thermal imaging radiometer would have six channels between 5 and 100 microns, and would be analogous to the Diviner instrument on the Lunar Reconnaissance Orbiter (LRO). This instrument would operate in continuous pushbroom mode and take data from the nadir orientation, with the occasional blackbody calibration to space. Data would be collocated with the MAC to identify hot spots.

The MS would be used to measure the composition of the plumes during plume fly-throughs. The instrument would have high resolution (>8000) and high mass range (>0–300) and accept samples at Vrel 0.1 to ~7 km/s. EO's MS would be similar to Cassini's INMS.

The dust analyzer would be used to measure size and velocity distribution, and composition, of dust in the Enceladus plume as well as the Saturn E-ring. It would be analogous to the Cassini cosmic dust analyzer (CDA), but with improved mass spectroscopy resolution, and the added ability to measure particle velocities.

The magnetometer would be used to measure the intrinsic magnetic field of Enceladus to order 12 and thus help in the search for subsurface oceans or lakes via the weak magnetic induction signature resulting from Enceladus's eccentric orbit through Saturn's magnetic field, as well as the magnetospheric interaction of Enceladus and the other icy satellites. The nominal sensitivity of the magnetometer would be 0.01 nT.

Table 3-1. Instrument Summary

Instrument Name	# Units	Heritage	Mission Heritage	TRL	Mass CBE (kg)	Mass +cont./Unit (kg)	Op. Power CBE (W)	Standby Power CBE (W)
Medium-angle camera	1	Ralph/MVIC	New Horizons	8+	10.0	13.0	5	2
Thermal imaging radiometer	1	Diviner	LRO	6	12.0	15.6	5	2
Mass spectrometer	1	INMS	Cassini	5	25.0	32.5	40	2
Dust analyzer	1	CDA	Cassini	5	3.0	3.9	10	2
Magnetometer	1	MAG	Galileo	8+	6.0	7.8	3	2

Table 3-2. Medium-Angle Camera

Item	Value	Units
Type of instrument	Optical	
Number of channels	4	
Size/dimensions	0.20 × 0.05 × 0.05	m × m × m
Instrument mass **without** contingency (CBE*)	10	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE + Reserve)	13	kg
Instrument average payload power **without** contingency	5	W
Instrument average payload power contingency	30	%
Instrument average payload power **with** contingency	6.5	W
Instrument average science data rate^ **without** contingency	2,400	kbps
Instrument average science data^ rate contingency	30	%
Instrument average science data^ rate **with** contingency	3,120	kbps
Instrument fields of view (if appropriate)	5.8 × 0.15	degrees
Pointing requirements (knowledge)	0.075	degrees
Pointing requirements (control)	0.29	degrees
Pointing requirements (stability)	0.0115	deg/sec

*CBE = Current best estimate.

^Instrument data rate defined as science data rate prior to onboard processing.

Table 3-3. Thermal Imaging Radiometer

Item	Value	Units
Type of instrument	Optical	
Number of channels	6	
Size/dimensions	0.5 × 0.14 × 0.14	m × m × m
Instrument mass **without** contingency (CBE*)	12	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE + Reserve)	15.6	kg
Instrument average payload power **without** contingency	5	W
Instrument average payload power contingency	30	%
Instrument average payload power **with** contingency	6.5	W
Instrument average science data rate^ **without** contingency	2,400	kbps
Instrument average science data^ rate contingency	30	%
Instrument average science data^ rate **with** contingency	3,120	kbps
Instrument fields of view (if appropriate)	0.38 × 0.19	degrees
Pointing requirements (knowledge)	n/a	degrees
Pointing requirements (control)	n/a	degrees
Pointing requirements (stability)	n/a	deg/sec

*CBE = Current best estimate.

^Instrument data rate defined as science data rate prior to onboard processing.

Table 3-4. Mass Spectrometer

Item	Value	Units
Type of instrument	Particles	
Number of channels	8000	
Size/dimensions	0.5 × 0.2 × 0.2	m × m × m
Instrument mass **without** contingency (CBE*)	25	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE + Reserve)	32.5	kg
Instrument average payload power **without** contingency	40	W
Instrument average payload power contingency	30	%
Instrument average payload power **with** contingency	52	W
Instrument average science data rate^ **without** contingency	3.4	kbps
Instrument average science data^ rate contingency	30	%
Instrument average science data^ rate **with** contingency	4.42	kbps
Instrument fields of view (if appropriate)	100	degrees
Pointing requirements (knowledge)	5	degrees
Pointing requirements (control)	5	degrees
Pointing requirements (stability)	n/a	deg/sec

*CBE = Current best estimate.

^Instrument data rate defined as science data rate prior to onboard processing.

Table 3-5. Dust Analyzer

Item	Value	Units
Type of instrument	Particles	
Number of channels	n/a	
Size/dimensions	n/a	m × m × m
Instrument mass **without** contingency (CBE*)	3	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE + Reserve)	3.9	kg
Instrument average payload power **without** contingency	10	W
Instrument average payload power contingency	30	%
Instrument average payload power **with** contingency	13	W
Instrument average science data rate^ **without** contingency	0.3	kbps
Instrument average science data^ rate contingency	30	%
Instrument average science data^ rate **with** contingency	0.39	kbps
Instrument fields of view (if appropriate)	4 × 40	degrees
Pointing requirements (knowledge)	5	degrees
Pointing requirements (control)	5	degrees
Pointing requirements (stability)	n/a	deg/sec

*CBE = Current best estimate.

^Instrument data rate defined as science data rate prior to onboard processing.

Table 3-6. Magnetometer

Item	Value	Units
Type of instrument	Fields	
Number of channels	3	
Size/dimensions	n/a	m × m × m
Instrument mass **without** contingency (CBE*)	6	kg
Instrument mass contingency	30	%
Instrument mass **with** contingency (CBE + Reserve)	7.8	kg
Instrument average payload power **without** contingency	3	W
Instrument average payload power contingency	30	%
Instrument average payload power **with** contingency	3.9	W
Instrument average science data rate^ **without** contingency	0.3	kbps
Instrument average science data^ rate contingency	30	%
Instrument average science data^ rate **with** contingency	0.39	kbps
Instrument fields of view (if appropriate)	n/a	degrees
Pointing requirements (knowledge)	0.1	degrees
Pointing requirements (control)	5	degrees
Pointing requirements (stability)	n/a	deg/sec

*CBE = Current best estimate.

^Instrument data rate defined as science data rate prior to onboard processing.

Table 3-7. Payload Mass and Power

	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Medium-angle camera	10.0	30	13.0	5	30	6.5
Thermal imaging radiometer	12.0	30	15.6	5	30	6.5
Mass spectrometer	25.0	30	32.5	40	30	52.0
Dust analyzer	3.0	30	3.9	10	30	13.0
Magnetometer	6.0	30	7.8	3	30	3.9
Total Payload Mass	**56.0**	**30**	**72.8**	**63**	**30**	**81.9**

Flight System

The flight system would consist of a single orbiter that would enter Enceladus orbit approximately 13 years after launch. In the 3.5 years prior to EOI, the spacecraft orbit would be changed through multiple flybys of Saturn's moons Titan, Rhea, and Dione, as well as Enceladus. Ten of the Enceladus flybys would be used for science observations, as would the subsequent 6 months (Option 2) to a year (Option 1) of Enceladus orbit.

The spacecraft would be nearly identical for Option 1 and Option 2 because the only difference between the two options would be the duration of science operations. The instruments included in the payload would be the MAC, thermal imaging radiometer, dust analyzer, MS, and magnetometer. The spacecraft would be dual-string with cold spares and would include the following: a rectangular bus; a fixed, 3-m HGA, three ASRGs for power; conventional bipropellant chemical propulsion; a fixed boom on which the MS and dust analyzer would be mounted, a deployable boom (hinged at the end of the fixed boom), on which the magnetometer would be located, and an articulated cover for the main engine. The spacecraft would be three-axis-stabilized. The total ionizing dose (TID) for the mission would be relatively low and would not require the use of a radiation vault, though some spot-shielding for electronic components could be required.

Table 3-8 and Table 3-9 show the spacecraft mass summary for each of the options. Because Option 2 would involve only six months of Enceladus orbit operations, compared to a year for Option 1, the amount of propellant needed for Option 2 would be smaller. This would reduce slightly the mass of the structures and propulsion subsystems in comparison to Option 1, but the spacecraft would be identical otherwise. The mass contingency policy shown is based on the subsystem-level contingency factors. Each subsystem designer provided a contingency factor based on the assumed subsystem heritage and complexity. The total subsystem contingency is computed based on the sum of all subsystem masses. A systems contingency factor is additionally applied to ensure that the total mass contingency is 43%. The 43% mass contingency corresponds to a 30% mass margin, which is consistent with JPL Design Principles [1]. The overall mass contingency is shown in the mass equipment lists in Appendix D.

Table 3-10 shows the identical power modes for the two options. The Team X power contingency policy is to add 43% contingency to the total power for each power mode. Unlike mass contingency policies, the power contingency policy does not have subsystem engineers adding contingency for power at the subsystem level.

Figure 3-1 shows the spacecraft in its deployed configurations, and Figure 3-2 shows the flight system functional block diagram.

Table 3-8. Flight System Mass and Power: Option 1

	Mass			Average Power		
	CBE (kg)	**% Cont.**	**MEV (kg)**	**CBE (W)**	**% Cont.**	**MEV (W)**
Structures & Mechanisms	302	30%	393	See details in Flight System Power Modes table		
Thermal Control	63	29%	81			
Propulsion (Dry Mass)	136	27%	173			
Attitude Control	40	10%	44			
Command & Data Handling	21	15%	24			
Telecommunications	72	18%	85			
Power	118	30%	153			
Total Flight System Dry Bus Mass	**752**	**27%**	**953**			

Table 3-9. Flight System Mass and Power: Option 2

	Mass			Average Power		
	CBE (kg)	**% Cont.**	**MEV (kg)**	**CBE (W)**	**% Cont.**	**MEV (W)**
Structures & Mechanisms	301	30%	391	See details in Flight System Power Modes table		
Thermal Control	63	29%	81			
Propulsion (Dry Mass)	135	27%	171			
Attitude Control	40	10%	44			
Command & Data Handling	21	15%	24			
Telecommunications	72	18%	85			
Power	118	30%	153			
Total Flight System Dry Bus Mass	**750**	**27%**	**949**			

Table 3-10. Flight System Power Modes: Options 1 and 2

Subsystem/ Instrument	Power (W)									
	Launch	**Cruise**	**Maneuvers**	**Science w/ MS (Flyby)**	**Science w/o MS (Flyby)**	**Telecom**	**Science w/ MS (Orbit)**	**Science w/o MS (Orbit)**	**Recharge**	**Safe**
Instruments	0.0	0.0	0.0	63.0	23.0	10.0	63.0	23.0	10.0	0.0
Structures and mechanisms	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0
Thermal control	13.5	15.9	15.9	15.9	15.9	15.9	15.9	15.9	15.9	15.9
Propulsion	45.3	45.3	310.3	45.3	3.3	3.3	3.3	3.3	3.3	45.3
Attitude control	33.7	39.1	39.1	92.1	92.1	58.4	58.4	58.4	58.4	39.1
Command and data handling	46.4	46.4	46.4	46.4	46.4	46.4	46.4	46.4	46.4	46.4
Telecommunications	15.0	25.0	76.0	15.0	15.0	135.0	15.0	15.0	15.0	15.0
Power	28.2	29.6	48.9	39.5	34.9	36.5	32.7	30.5	29.8	29.0
TOTALS	**182.1**	**201.3**	**536.6**	**317.2**	**230.6**	**305.5**	**234.7**	**192.5**	**178.8**	**190.7**
System contingency %	43%	43%	43%	43%	43%	43%	43%	43%	43%	43%
Subsystems with Contingency	**260.4**	**287.8**	**767.3**	**453.6**	**329.7**	**436.8**	**335.6**	**275.3**	**255.7**	**272.7**
Power mode duration (hours)	3.0	24.0	1.0	1.5	10.0	5.0	1.0	7.0	11.0	24.0

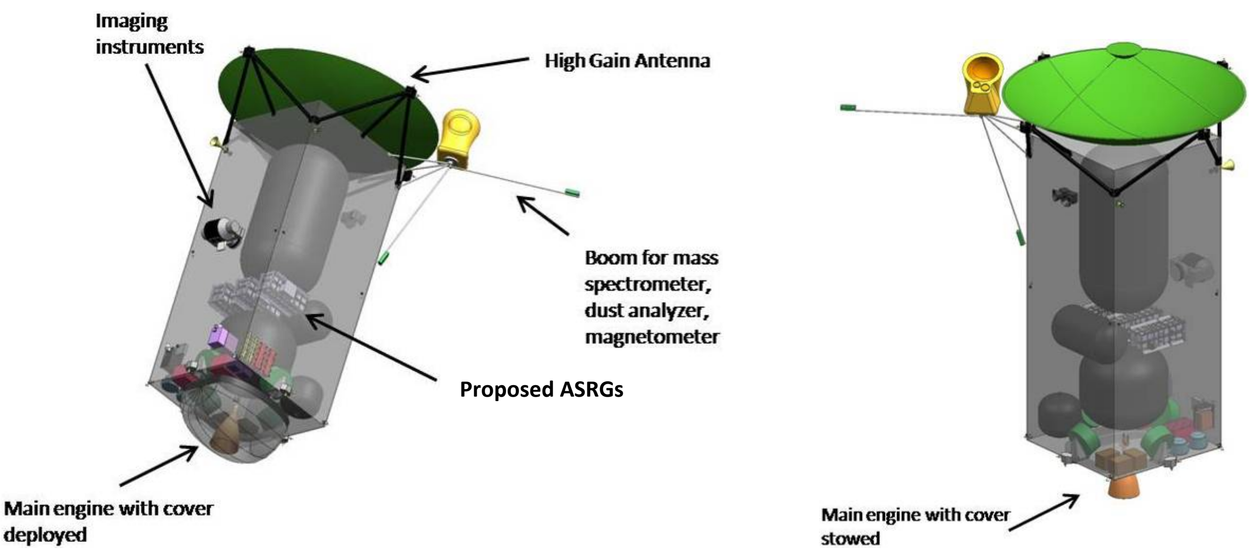

Figure 3-1. Artist's Concept for Flight System Configuration

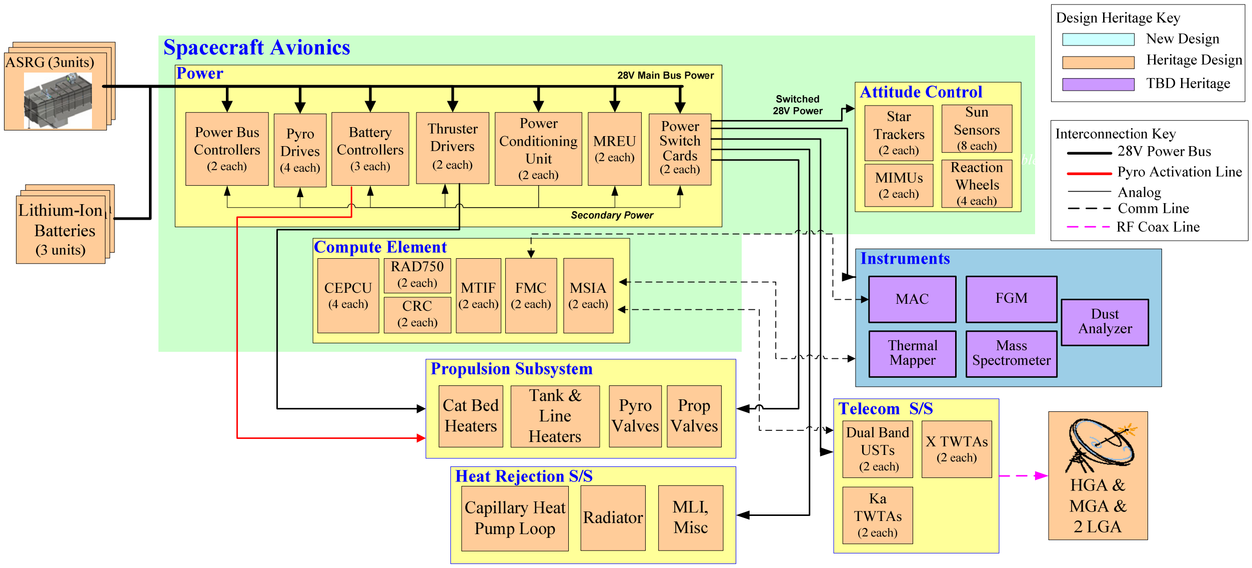

Figure 3-2. Flight System Functional Block Diagram—Options 1 and 2

Subsystem Descriptions

Table 3-11 provides detailed subsystem characteristics. Descriptions of each of the subsystems are provided below.

Structures

The structure of the spacecraft would consist of a rectangular bus (1.7 m × 1.7 m × 4 m) enclosing the four propulsion tanks and three ASRGs located between the two largest propulsion tanks. A 3-m HGA would be fixed on the upper surface of the bus, with thruster clusters located on tripods at the four upper corners of the bus so as to extend beyond the HGA. Thruster clusters are also located at the four lower corners of the bus. A fixed boom would support the MS and dust analyzer, and a deployable boom would be hinged at the end of the fixed boom to support the magnetometer. A main engine cover to protect the main engine during spacecraft E-ring passes would be deployed and stowed utilizing a single-axis actuator.

Attitude Control

The attitude control subsystem (ACS) would be a dual-string design with cold spares, providing three-axis stabilization with reaction wheels, star trackers, Sun-sensors and inertial measurement units (IMUs). The ACS design would be driven by the pointing requirement for the HGA to within 0.05 degrees of Earth, and the 0.035 mrad/sec stability requirement for the MAC. Sixteen 0.9-N thrusters would be used for momentum unloading and for small orbit-correction maneuvers, and four 90-N thrusters would be used for orbit maintenance maneuvers and thrust-vector control during main-engine burns. Star-trackers would provide inertial attitude knowledge, and IMUs would provide relative attitude knowledge between the inertial attitude knowledge updates by star-trackers. Sun-sensors would be used for initial attitude determination after launch and for Sun search in the event of safing. Reaction wheels would provide precise pointing required for the HGA as well as the MAC. To save power, IMUs would be turned off upon reaching Enceladus orbit. All stellar attitude determination would be used at that time.

Propulsion

The propulsion subsystem would provide three-axis control during cruise and tour of Saturn’s moons, momentum wheel unloading, and main-engine burn-control authority. The bipropellant propulsion system would include one 890-N main engine used for all deep space maneuvers, orbit insertions, and orbit pump down; four 90-N engines used for main-engine-control authority; and sixteen 0.9-N attitude control engines in two redundant branches used for three-axis control and momentum wheel unloading. There would be a single hydrazine fuel tank and a single NTO oxidizer tank, which would be composite, overwrapped with titanium liners. The design would include propellant management devices (PMDs) for propellant expulsion, as well as one fuel pressurant tank and one oxidizer pressurant tank, which would also be composite, overwrapped with titanium liners.

Thermal

The thermal design for the mission would be driven by the requirement to maintain the propulsion module within specified temperature limits while minimizing heater power so as to keep the spacecraft power to a minimum. As a result, the thermal design would use both active and passive thermal control. The dissipated energy from the three ASRGs located between the two large propellant tanks would be used to heat the propellant tanks and electronics. A capillary-pumped heat pipe (CPHP) would transfer the thermal energy from the ASRGs to the tanks. A thermal radiator would be required to provide variations in the thermal energy rejection needed during different phases of the mission. Twenty radioisotope heater units (RHUs) would be used in the bus and to heat the thruster clusters. Multilayered insulation (MLI), thermistors, and temperature sensors would also be used for thermal control.

Power

The power subsystem would consist of three ASRGs, providing a total of 275 W of power at 13 years after launch. Three Li-ion batteries, with a total capacity of 90 A-hr, would be included for energy balancing. To ensure positive energy margin over a 24-hr period, battery sizing would be based on an operational sequence of 5 hours of telecommunications and 11 hours of battery recharge, followed by consecutive science modes (8 hours total) with and without the MS enabled. The design would have dual-string power electronics with a “cold” second electronics string that would not be brought online unless there were a fault in the primary string. Excess generated energy would be shunted into space using shunt radiators provided by the thermal-control subsystem. The battery system would be sized to provide maximum depth of discharge of 70% fewer than 100 times in the mission on the “prime” batteries; the system would include one additional battery to meet single-fault-tolerance requirements. The design would be robust to the loss of a single piston in one of the ASRG units with no degradation of science. It is assumed that the power subsystem would be tightly monitored, and operations would be replanned as necessary to be consistent with available power output.

Command and Data Handling

The command and data handling (C&DH) subsystem would be a dual-string system with cold spares. The science data volume for the Enceladus flybys would drive storage requirements. The C&DH design assumes build-to-print Multi-Mission System Architecture Program (MSAP)–heritage components with no new development required. The standard MSAP architecture would be included in the design, except for the standard nonvolatile memory board, which would be replaced with a higher-capacity flash memory card to provide sufficient data storage. The architecture would include a RAD750 processor; critical relay control card (CRCC); MSAP telecommunication interface card (MTIF), serial interface assembly (MSIA), motor control and interface card (MCIC) for interface with the IMU, remote engineering units (MREUs); and a 96-GByte flash memory card.

Telecommunications

The telecommunications subsystem would be required to support a two-way X-band link with Earth during the entire mission for command uplinks and engineering telemetry downlinks. The subsystem would need to support a Ka-band downlink of 65 kbps using two arrayed 34-m DSN stations for science data return, as well as simultaneous X-band up and down and Ka-band up and down for precision Doppler tracking for gravity science. The telecommunications system for all options would be a fully redundant X-band and Ka-band system. The design would consist of one 3-m X/Ka-band HGA, one X-band medium-gain antenna (MGA), and two X-band low-gain antennas (LGAs) along with two Ka-band 50-W traveling-wave tube amplifiers (TWTAs), two X-band 25-W TWTAs, and two X/Ka up/down USTs. The telecommunication design assumes that the spacecraft would stop taking science during downlinks and point the body-fixed HGA within 0.05 degrees of the ground station. The UST is a next-generation space transponder, which is currently being developed for other missions, such as the 2018 ExoMars mission. The UST would not be a required new development for the EO mission because if the UST is not developed in time for this mission, modifications could be made to existing SDSTs to achieve the same functionality.

Table 3-11. Subsystem Characteristics

Subsystem Parameters (as appropriate)	Value/Summary, Units	
	Option 1	**Option 2**
General		
Design life, months	156	150
Structure		
Structures material (aluminum, exotic, composite, etc.)	Aluminum, Titanium, Composites	
Number of articulated structures	1 (Main engine cover)	
Number of deployed structures	2 (Magnetometer boom, ME cover)	
Aeroshell diameter, m	N/A	
Thermal Control		
Type of thermal control used	Capillary-pumped heat pipe (for ASRG waste heat), radiator, RHUs, MLI	
Propulsion		
Estimated delta-V budget, m/s	2881	2835
Propulsion type(s) and associated propellant(s)/oxidizer(s)	Bipropellant hydrazine, NTO main engine	
Number of thrusters and tanks	(1) 890-N main engine (4) 90-N engines for main engine control authority (16) Attitude control thrusters (1) Hydrazine tank (1) NTO tank (1) Hydrazine pressurant tank (1) NTO pressurant tank	
Specific impulse of each propulsion mode, seconds	Biprop: 315	
Attitude Control		
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis	
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial	
Attitude control capability, degrees	0.05 deg	
Attitude knowledge limit, degrees	0.025 deg	
Agility requirements (maneuvers, scanning, etc.)	Requirements for imaging/MAC—control 1 mrad, knowledge 0.5 mrad, stability—0.035 mrad/sec	
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	1-axis gimbal for main engine cover	

Subsystem Parameters (as appropriate)	Value/Summary, Units
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	Sun sensors (radial accuracy <±0.5 deg) Star trackers (pitch/yaw accuracy within ±18 arcsec) IMUs (bias stability within ±0.015 arcsec/sec) Reaction wheels (25 Nm momentum storage capability, 0.2 Nm torque capability)
Command & Data Handling	
Housekeeping data rate, kbps	2
Data storage capacity, Mbits	96 GB flash memory
Maximum storage record rate, kbps	1000
Maximum storage playback rate, kbps	65
Power	
Type of array structure (rigid, flexible, body mounted, deployed, articulated)	N/A (3 ASRG)
Array size, meters × meters	N/A
Solar cell type (Si, GaAs, Multi-junction GaAs, concentrators)	N/A
Expected power generation at beginning of life (BOL) and end of life (EOL), watts	480 W BOL, 375 W EOL (13 years)
On-orbit average power consumption, watts	driving power mode—437 W telecom (5 hrs)
Battery type (NiCd, NiH, Li-ion)	Li-ion
Battery storage capacity, amp-hours	30

Concept of Operations and Mission Design

The Enceladus Orbiter mission would involve several key operational phases:

1. Interplanetary cruise, which would include two Venus flybys and two Earth flybys. This phase would begins with launch and end with SOI.
2. Saturn moon tour, which would include 3 Titan flybys, 15 Rhea flybys, 10 Dione flybys, 12 Tethys flybys, 12 engineering flybys of Enceladus, and 10 scientific flybys of Enceladus. The phase would end with EOI.
3. Enceladus orbit, which would include all scientific, mapping orbits of Enceladus.

These phases will be described in the following sections.

Interplanetary Cruise

The interplanetary cruise for the EO mission is illustrated in Figure 3-3. The nominal launch date would be January 28, 2023; a 21-day launch period would be constructed around this date. The trajectory would require a launch C_3 of 11.7 km^2/s^2. The spacecraft would fly by Venus twice and Earth twice before reaching Saturn on July 29th, 2031. One large, deterministic maneuver would be required during the cruise; otherwise only small trajectory correction maneuvers (TCMs) would be required. Table 3-12 summarizes information about the cruise.

There are numerous ways that the Enceladus Orbiter could launch from Earth and reach Saturn. Figure 3-4 shows a comparison of many interplanetary options to reach Saturn, including options that implement solar electric propulsion as well as chemical propulsion. The trajectory studied here is labeled for comparison.

When the spacecraft would arrive at Saturn, it would perform an insertion maneuver to place it into a 6-month orbit. This would require approximately 990 m/s of ΔV, the largest maneuver to be performed in the mission.

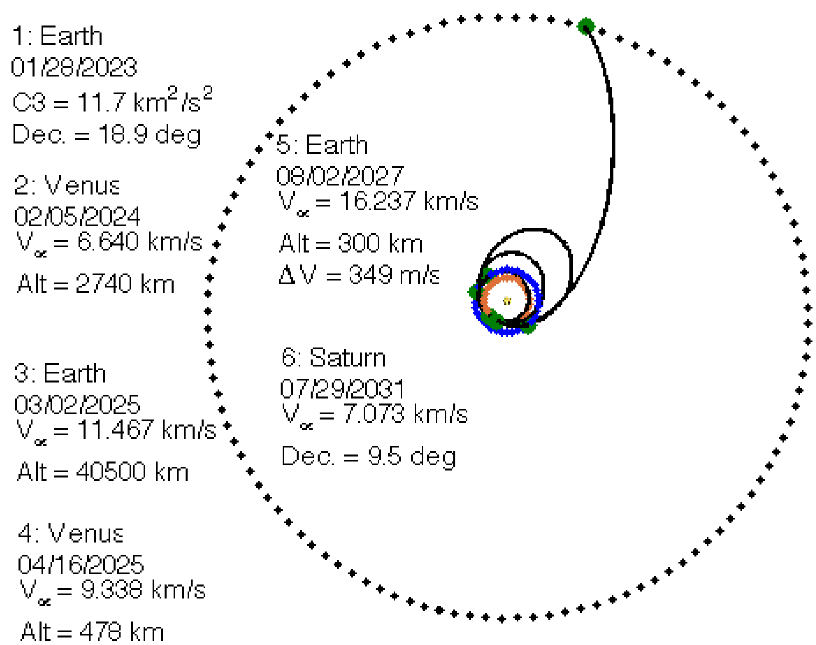

Figure 3-3. Interplanetary cruise for the Enceladus Orbiter mission would include two Venus flybys and two Earth flybys.

Table 3-12. Key Events, Timeline, and ΔV Budget for Interplanetary Cruise

Event	Date	L + mos.	ΔV (m/s)	Comments
Launch	1/28/2023	-	50	Launch error correction, C3 = 11.7 km^2/s^2; DLA = 18.9°
Venus flyby	2/05/2024	12.4	5	Flyby altitude = 2740 km, statistical ΔV
Earth flyby	3/02/2025	25.5	5	Flyby altitude = 40,500 km, statistical ΔV
Venus flyby	4/16/2025	27.0	5	Flyby altitude = 478 km, statistical ΔV
Earth flyby	8/02/2027	54.9	355	Flyby altitude = 300 km, statistical ΔV + Earth bias + deterministic ΔV
Saturn arrival	7/29/2031	103.5	5	Statistical ΔV during cruise
Saturn orbit insertion	7/29/2031	103.5	988.8	V_∞ = 7.073 km/s. This insertion burn includes gravity loss expectations.
Total ΔV used during interplanetary cruise:			1413.8 kg, which is 49.1% of the total 2880.6 kg budget	

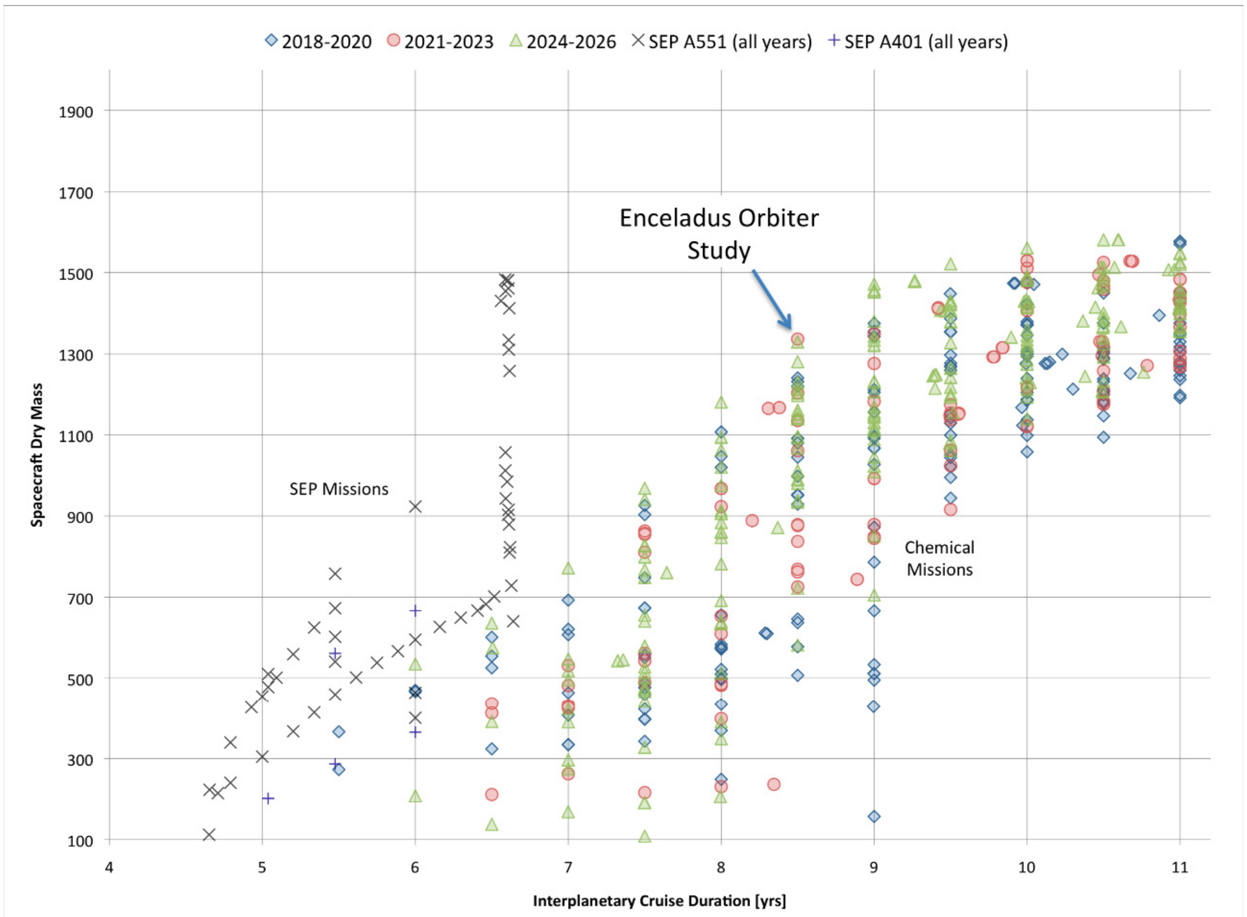

Figure 3-4. A comparison of many interplanetary cruise options for missions to Saturn: The x-axis shows the expected duration between launch and Saturn orbit insertion; the y-axis shows the expected dry mass of the spacecraft, including all maneuvers while at Saturn.

Saturn Moon Tour

The mission would arrive at Saturn and insert into a 6-month orbit. At its first apoapse, the spacecraft would perform a 550-m/s maneuver to raise its periapse. This raise would be needed in order to encounter Titan at the spacecraft's first periapse passage. At that point the spacecraft would have entered its Saturn moon tour phase.

The Saturn moon tour described here is well documented in [5]. One might imagine an even more optimal tour of Saturn's moons, but it is a very good representation of the type of tour that could be available for this mission. Additional research may uncover dramatic improvements in the tour, i.e., ways to get to Enceladus faster, with fewer flybys, or with more scientifically valuable flybys.

The purpose of the moon tour would be to reduce the spacecraft's energy in its orbit about Saturn. If the spacecraft immediately inserted into orbit about Enceladus, it would require an orbit insertion maneuver of approximately 3940 m/s. Although it would require about three years and a total of 52 flybys, the flybys of the moons would reduce the required orbit insertion maneuver to only 152 m/s. The moon tour would require approximately 619 m/s; hence, the tour would save the mission over 3000 m/s of propellant.

The tour would remain very close to the equatorial plane of the Saturnian system at all times so that it could make the best use of the gravity of each moon. Each flyby would passes across the equator of the moon to leverage as much energy as possible.

The tour would begin with three Titan flybys; Titan is large enough that it would dramatically reduce the size of the orbit in only a few flybys. The spacecraft would fly by Titan at altitudes between 2280 and 15,000 km. The final Titan flyby would put the spacecraft on a transfer to Rhea. Aside from Titan, all of the natural satellites of Saturn are very small bodies and would not have the gravity to make large changes to the orbit. Hence, it would require many flybys to fulfill the energy change requirement of the tour. The general requirement would be that no flyby could be lower than 50 km, and the first flyby of a moon would need to be at least 100 km if not higher. In addition, no two flybys could be closer together than about 10 days for navigation purposes. The Cassini guideline was to have no less than 12 days between moon flybys, while Galileo had a minimum of 7 days between flybys. The 10-day interval for the EO mission was chosen as a reasonable average for the nominal trajectory. These requirements could change depending on specific spacecraft operations planning.

The tour would continue with 15 Rhea flybys, which would lead the spacecraft to an encounter with Dione. After 10 Dione flybys, the spacecraft would encounter Tethys. The spacecraft would pass by Tethys 12 times before encountering Enceladus for the first time. A total of 12 Enceladus flybys would be required to reduce the spacecraft's orbit and bring the insertion maneuver down to 152 m/s. Additional scientific passes could be scheduled as needed; a total of 10 scientific passes are currently planned. These passes would allow the spacecraft to fly through the geyser plumes. The engineering passes would need to be more equatorial in nature, but the scientific passes could be at nearly any inclination.

The tour would include approximately 25 deterministic maneuvers and numerous statistical maneuvers. The spacecraft would be tracked immediately following a flyby and then a small maneuver could be performed to clean up the errors involved with the flyby. This would make it possible to fly accurately to the next encounter. This statistical maneuver could be combined with a preexisting deterministic maneuver, if one were planned. As the spacecraft would be on its approach to the next flyby, the spacecraft could perform a final targeting maneuver to adjust its flyby trajectory. [5] provides all of the details about each flyby and the leveraging maneuvers that would be planned for the tour.

Enceladus Science Orbit

After performing the 152 m/s orbit insertion, the spacecraft would become captured into its mapping orbit about Enceladus. Enceladus is very close to Saturn; thus any polar orbit would likely be unstable. [6] describes the science orbit that has been used for the proposed design. This orbit is among the lowest-altitude, highly inclined orbits that have been found to date. The orbital parameters would vary substantially from orbit to orbit, but this variation would repeat over the course of ~1.37 days, namely, the orbital period of Enceladus about Saturn. For the first six months in Enceladus orbit, the spacecraft would be in a 6-hour orbit with an average inclination of about 62 degrees and an average altitude of approximately 184 km. The orbit would be elliptical and would oscillate quite a bit: The lowest expected altitude would be approximately 101 km, and the highest expected altitude would be approximately 267 km. Since Enceladus orbits Saturn in only 1.37 days, the orbit would precess a full 360 degrees in 1.37 days, making it very easy to map the entire planet. Because Option 2 includes only 6 months in Enceladus orbit, the spacecraft would then deorbit. In Option 1, the spacecraft would transfer into a lower-altitude orbit after about 6 months. This orbit would have an orbital period of about 4 hours. Its mean altitude would be approximately 81 km, and it would also vary in altitude, though it is not clear by how much. The spacecraft's altitude would probably dip down to about 30 km.

Finally, with the spacecraft's science mission complete, the spacecraft would perform a deorbit maneuver and impact the surface of Enceladus. It is currently assumed that it would be acceptable, from a planetary protection (PP) perspective, to strike in a geologically ancient area and avoid impacting any area where there would be potential to contaminate liquid water.

Delta-V Budget

The interplanetary cruise delta-V budget is shown in Table 3-12.Table 3-13 shows the delta-V budget for all phases of the mission. The fuel budget would accommodate all of the following:

- Deterministic maneuvers, namely, those maneuvers that would need to be performed in order to change the orbit appropriately to reach the next destination
- Statistical maneuvers, namely, TCMs that would be required to correct the trajectory in the presence of maneuver execution errors, modeling errors, etc.
- PP biasing maneuvers, namely, maneuvers that would purposefully adjust the trajectory to avoid inadvertently impacting a protected body
- Contingency: fuel that is budgeted in the current mission design but which could be removed if the spacecraft were constructed to be less "noisy"; that is, if the spacecraft were expected to execute its maneuvers with near-perfect accuracy then some statistical fuel budget could be removed.
- Margin: fuel that could be needed for unknown adjustments or extensions

The statistical delta-V for orbit maintenance for one year of Enceladus orbit is estimated to be 50 m/s. Two analyses of the delta-V requirements for orbit maintenance produced estimates significantly lower than 50 m/s, but this number was chosen as a conservative estimate.

The delta-V budget includes enough delta-V to ensure that the spacecraft would be able to afford all TCMs and orbit maintenance maneuvers to achieve its mission objectives. Indeed, it is designed to be a conservative budget. For instance, there would be 66 flybys in the nominal trajectory that would need to be targeted well. The budget would include 5 m/s of delta-V for each flyby. Although some of the flybys could require the full 5 m/s (or more), it is very likely that the spacecraft would not use all of the budgeted propellant during the moon tour.

Other Mission Design Parameters

Table 3-14 gives other key parameters of the EO mission design.

Table 3-13. Delta-V Budget for the Enceladus Orbiter

Event	# Flybys	Delta-V (m/s)	Statistical (m/s)	Comments
Launch			50	Injection clean-up and Venus targeting
Venus flyby	1		5	Target Earth
Earth flyby	1		5	Target Venus
Venus flyby	1		10	Target Earth, biasing.
Earth flyby	1	350	5	Target Saturn
Saturn orbit insertion		960	28.8	Gravity losses included, 3% statistical for clean-up
Periapse raise		534	16	3% statistical for clean-up
Titan tour	3	29	15	5 m/s statistical per flyby
Rhea tour	15	146	75	5 m/s statistical per flyby
Dione tour	10	26	50	5 m/s statistical per flyby
Tethys tour	12	12	60	5 m/s statistical per flyby
Enceladus tour	22	146	110	Includes planetary protection biasing and 5 m/s statistical per flyby
Enceladus orbit insertion		148.3	3.9	Includes 15% gravity losses (deterministic) and 3% clean-up (statistical)
Science orbit transfer		20	0.6	This applies only to Option 2
Orbit maintenance			50	Option 2 requires only 25 m/s for the 6-month mission.
Disposal		25		
Total	**64**	**2396.3**	**484.3**	**2880.6 m/s Total**

Table 3-14. Mission Design

Parameter	Value		Units
Science orbit inclination	62		deg
Science orbit average altitude	184		Km
Science orbit period	6		hours
Mission lifetime	156	150	mos
Maximum eclipse period	<60		min
Launch site	KSC		
Total flight system mass **with** contingency (includes instruments)	1154	1150	kg
Propellant mass **without** contingency	2199	2144	kg
Propellant contingency	225	266	%
Propellant mass **with** contingency	2434	2410	kg
Launch adapter mass **with** contingency	N/A		kg
Total launch mass	3589	3560	kg
Launch vehicle	Atlas V 521		Type
Launch vehicle lift capability	3640 for 11.7 C3		kg
Launch vehicle mass margin	51.5	79.8	kg
Launch vehicle mass margin (%)	1	2	%

Ground System

During the 3.5-year Saturn moon tour, the ground system would support one 4-hour DSN pass per day. After Enceladus orbit insertion, the spacecraft would have one 8-hour DSN pass per day, which would enable 5 hours of data downlink. This ground system design, summarized in Table 3-15, would provide sufficient science data downlink as well as two-way Doppler measurements for radio science experiments. The 96-GB flash memory card on the spacecraft would provide significant storage margins and enable large amounts of data to be captured in a short period of time.

Table 3-15. Mission Operations and Ground Data Systems

<table>
<tr><th></th><th>Mission Phase 1: Inner Planet Flybys</th><th>Mission Phase 2: Inner Solar System Cruise</th><th>Mission Phase 3: Quiet Cruise</th><th>Mission Phase 4: Saturn System Pumpdown & Flybys</th><th>Mission Phase 5: Orbital Science Phase</th></tr>
<tr><td colspan="6">Downlink Information</td></tr>
<tr><td>Number of contacts per week</td><td>7</td><td>1</td><td>1</td><td>7</td><td>7</td></tr>
<tr><td>Number of weeks for mission phase, weeks</td><td>15</td><td>223</td><td>176</td><td>177</td><td>54</td></tr>
<tr><td>Downlink frequency band, GHz</td><td colspan="5">Ka-band, 31.8–32.3, X-band, 8.40-8.45</td></tr>
<tr><td>Telemetry data rate(s), kbps</td><td colspan="5">65 for Ka-band, 10 for X-band</td></tr>
<tr><td>Transmitting antenna type(s) & gain(s), DBi</td><td colspan="5">3-m HGA</td></tr>
<tr><td>Transmitter peak power, Watts</td><td>76</td><td>76</td><td>76</td><td>120</td><td>135</td></tr>
<tr><td>Downlink receiving antenna gain, DBi</td><td colspan="5">34-m beam waveguide (BWG) (two arrayed)</td></tr>
<tr><td>Transmitting power amplifier output, W</td><td colspan="5">25 for X-band, 50 for Ka-band</td></tr>
<tr><td>Total daily data volume, (MB/day)</td><td>41Mb</td><td>41Mb</td><td>41Mb</td><td>936Mb</td><td>1170Mb</td></tr>
<tr><td colspan="6">Uplink Information</td></tr>
<tr><td>Number of uplinks per day</td><td>1/day</td><td>1/week</td><td>1/week</td><td>1/day</td><td>1/day</td></tr>
<tr><td>Uplink frequency band, GHz</td><td colspan="5">Ka-band, X-band, 34.2 -34.7, X-band, 7.145 -7.19</td></tr>
<tr><td>Telecommand data rate, kbps</td><td colspan="5">2</td></tr>
<tr><td>Receiving antenna type(s) & gain(s), DBi</td><td colspan="5">3-m HGA</td></tr>
</table>

Science Data Observation/Acquisition Phases

A straw set of data acquisition activities suitable for fulfilling the science objectives was defined to allow assessment of onboard data storage and data-return issues. The telecommunications downlink assessment (Table 3-16) assumes a 3-m HGA at Ka-band (single polarization), 50-W radiated power, and a DSN receiving array of two 34-m dishes. The downlink data rate is ~65 kbps (before accounting for overhead or compression) to two arrayed DSN 34-m antennas. The assessment demonstrated that all architectures would have sufficient telemetry margins to meet the science objectives. Instruments that produce the largest data volumes are present in all architectures, so the assessment is relevant across all architectures.

Table 3-16. Mission Data Assessment

Mission data plan (orbital)
Compression: 2
Track hours/day: 8

	Resolution (meters/pix)	#Bands	#Observations	Fractional Coverage	Uncompressed Bits	Return Time (days)
Global panchromatic map	20	1	1	1.00	3.19E+10	8.10
Global stereo map	50	2	1	1.00	1.02E+10	2.59
Global color map	100	4	1	1.00	5.11E+09	1.30
Global thermal map	500	6	2	1.00	6.13E+08	0.16
Global phase function map	1000	4	5	1.00	2.56E+08	0.06
Subtotal	—	—	—	—	**4.81E+10**	**12.2**
S. pole panchromatic map	5	1	2	1.00	5.12E+10	12.98
S. pole stereo map	10	2	1	1.00	1.28E+10	3.25
S. pole color map	50	4	2	1.00	2.05E+09	0.52
S. pole thermal map	20	6	10	1.00	9.60E+10	24.34
S. pole phase function map	100	4	4	1.00	1.02E+09	0.26
Hi-res S. pole panchromatic samples	0.5	1	1	0.05	1.28E+11	32.45
Subtotal	—	—	—	—	**2.91E+11**	**73.80**
	years					
Fields and particles	1	—	—	—	3.15E+11	79.95
Mission total					**6.54E+11**	**165.96**

Enhanced Opportunities beyond the Science Baseline

There are some enhanced opportunities beyond the EO science baseline. For instance, it is possible to return a 1-m/pixel map of the entire moon in less than a year. Images at this resolution on Mars have increased knowledge of the geology and current activity dramatically and would likely do the same for Enceladus.

Planetary Protection

In accordance with NASA's "Planetary Protection Provisions for Robotic Extraterrestrial Missions" (NPR 8020.12C) [7], the EO mission would be expected to be PP Category III. Accordingly, the mission would need to meet the Category III PP requirements (Appendix A, Section A.2 of [7]). The PP category of the mission would be formally established by the NASA Planetary Protection Officer (PPO) in response to a written request from the EO Project Manager, submitted by the end of Phase A.

The EO project would prepare all PP documents and hold all reviews as required by the NASA PPO. The EO project would plan to demonstrate compliance with the nonimpact requirements for Mars and Titan by a combination of trajectory biasing and analyses performed by the navigation team. Compliance with the probability of biological contamination of Enceladus requirement would be demonstrated by analysis. The probability of accidental impact of Enceladus and an impact analysis, in the event of accidental impact, would be performed, similar to the analyses performed for Juno. The EO team would also calculate the probability of creating a transient "special region"[1] in the event of an uncontrolled entry. Heat-flow analyses of the perennial heat source(s) would be performed for both the nominal end of mission and the uncontrolled entry case, to determine if it would be possible for any spacecraft hardware containing a perennial heat source to melt water ice at its final resting point. In addition, the spacecraft microbial burden at launch would be estimated by performing bioassay sampling of all spacecraft hardware. If the probability of contamination were to exceed the requirement, then the spacecraft would be cleaned or microbially reduced as needed to meet the requirement (note: this is not included in the cost estimate). The results of all of these analyses would be documented in the PP Prelaunch Report. The navigation team would also identify the location of the impacting point on Enceladus. That location would be reported in the PP End-of-Mission Report.

The current assumption is that if it could be shown that the orbiter would be able to meet the required low probability of impacting a special region (assumed to be 55–90 degrees latitude in the south-polar region), the orbiter would not need to be sterilized. If the probability of interaction with a liquid water environment requirement could not be met, or if the requirement were to change, PP would require sterilization of the orbiter at the cost of approximately $100M and the possible addition of a 300-kg biobarrier to the spacecraft to prevent site contamination. The additional mass of the biobarrier could necessitate the use of a larger launch vehicle than currently assumed in the design, at significant added cost. If the mission were to need to avoid the currently planned impact with Enceladus at end of mission, an added 300 kg of propellant would need to be included in the design for either biasing the spacecraft, or landing in an acceptable region of Enceladus.

Risk List

The EO study identified three medium and nine low risks as significant at the system level. Many of the risks would be common for any outer-planets mission with a long cruise stage and the use of nuclear power. Figure 3-5 provides a summary of identified EO risks on a 5 x 5 matrix. Table 3-17 defines the risk levels used in the matrix. Table 3-18 describes the moderate risks and a selected set of the identified low risks in some detail. All identified risks would apply to both options, since the only substantive difference between the two is the length of Phase E. Option 1 has a Phase E duration of 155 months while Option 2 has a Phase E duration of 149 months. Because of the early stage of the design and limited study time, risk mitigations have not yet been identified.

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[1] A "special region" is a localized environment where conditions (temperature, water activity) might occur that are conducive to replication of any terrestrial organisms carried on the spacecraft.

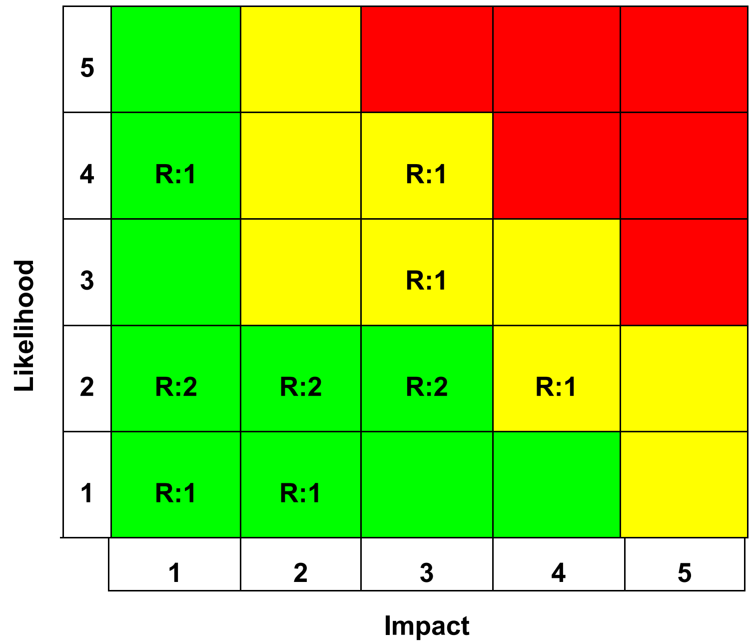

***Figure 3-5. Risk Chart*[2]**

Table 3-17. Risk Level Definitions

Levels	Mission Risk		Implementation Risk	
	Impact	**Likelihood of Occurrence**	**Impact**	**Likelihood of Occurrence**
5	Mission failure	Very high, >25%	Consequence or occurrence is not repairable without engineering (would require >100% of margin)	Very high, ~70%
4	Significant reduction in mission return (~25% of mission return still available)	High, ~25%	All engineering resources would be consumed (100% of margin consumed)	High, ~50%
3	Moderate reduction in mission return (~50% of mission return still available)	Moderate, ~10%	Significant consumption of engineering resources (~50% of margin consumed)	Moderate, ~30%
2	Small reduction in mission return (~80% of mission return still available)	Low, ~5%	Small consumption of engineering resources (~10% of margin consumed)	Low, ~10%
1	Minimal (or no) impact to mission (~95% of mission return still available)	Very low, ~1%	Minimal consumption of engineering resources (~1% of margin consumed)	Very low, ~1%

[2] The number of risks scored in each square of the risk matrix is indicated by the numbers in the Risk Chart. The three moderate and nine low risks that were identified in the study are shown in the matrix.

Table 3-18. Detailed Risk Description and Mitigation Strategy–Either Mission Option

R:#	Risk	Level	Description	Impact	Likelihood	Mitigation
1	Planetary protection estimates based on analysis	M	The PP cost estimates were based on the assumption that analysis would provide sufficient data to show that the mission could meet PP requirements. However, if it were determined that a biobarrier would be required for protection against launch vehicle contamination, or if impacting Enceladus were unacceptable, then a significant increase in mass and cost could result.	3	4	
2	Spacecraft component reliability issues due to long mission lifetime	M	Since most spacecraft components would not be qualified for the long mission duration required for this mission, there would be some likelihood of mission failure resulting from failure of critical components.	3	3	
3	Inability to execute mission as designed due to unavailability of plutonium	M	If the supply of plutonium were to continue to be limited, then the Enceladus mission could be rendered impossible for architectures dependent on radioisotope power sources such as ASRGs.	4	2	
4	Spacecraft damaged by plume or E-ring Large Particles	L	If large particles were to impact critical components of the spacecraft during plume or E-ring passes, there could be significant damage, causing loss of science. The design would include a main engine cover that would reduce the likelihood of damage to the main engine, and thus would lower the likelihood of this risk.	3	2	
5	Plumes may be inactive during the science mission	L	If Enceladus's plumes were inactive during the mission, then the mission's science return could be reduced.	2	2	
6	Uncertainty in ASRG launch approval costs resulting from near-earth flyby	L	Architectures that carry ASRGs and include Earth flybys in the trajectory could be affected by increased costs to mitigate risks of contaminating Earth with nuclear material. A change to Earth-avoidance requirements could increase the cost of the mission (as happened with Cassini).	2	2	
7	ASRG single converter failure resulting in large undamped vibrations	L	If one converter out of the two in an ASRG unit were to fails, then there could be large undamped vibrations that could interfere with spacecraft instruments.	1	2	

4. Development Schedule and Schedule Constraints

High-Level Mission Schedule

Figure 4-1 provides a feasible schedule for both options of the EO mission. The mission complexity would fall between that of a typical Discovery-class mission and that of a large, directed mission. The reference schedules used for this study were derived from the JPL mission schedule database, which extends back to the Voyager mission.

No major schedule drivers or long-lead items would need to be addressed beyond the proposed schedule. Table 4-1 provides key phase durations for the project. Since this mission is being treated as a New Frontiers mission, all instruments and the flight system would be planned to be delivered at the beginning of system-level integration and test.

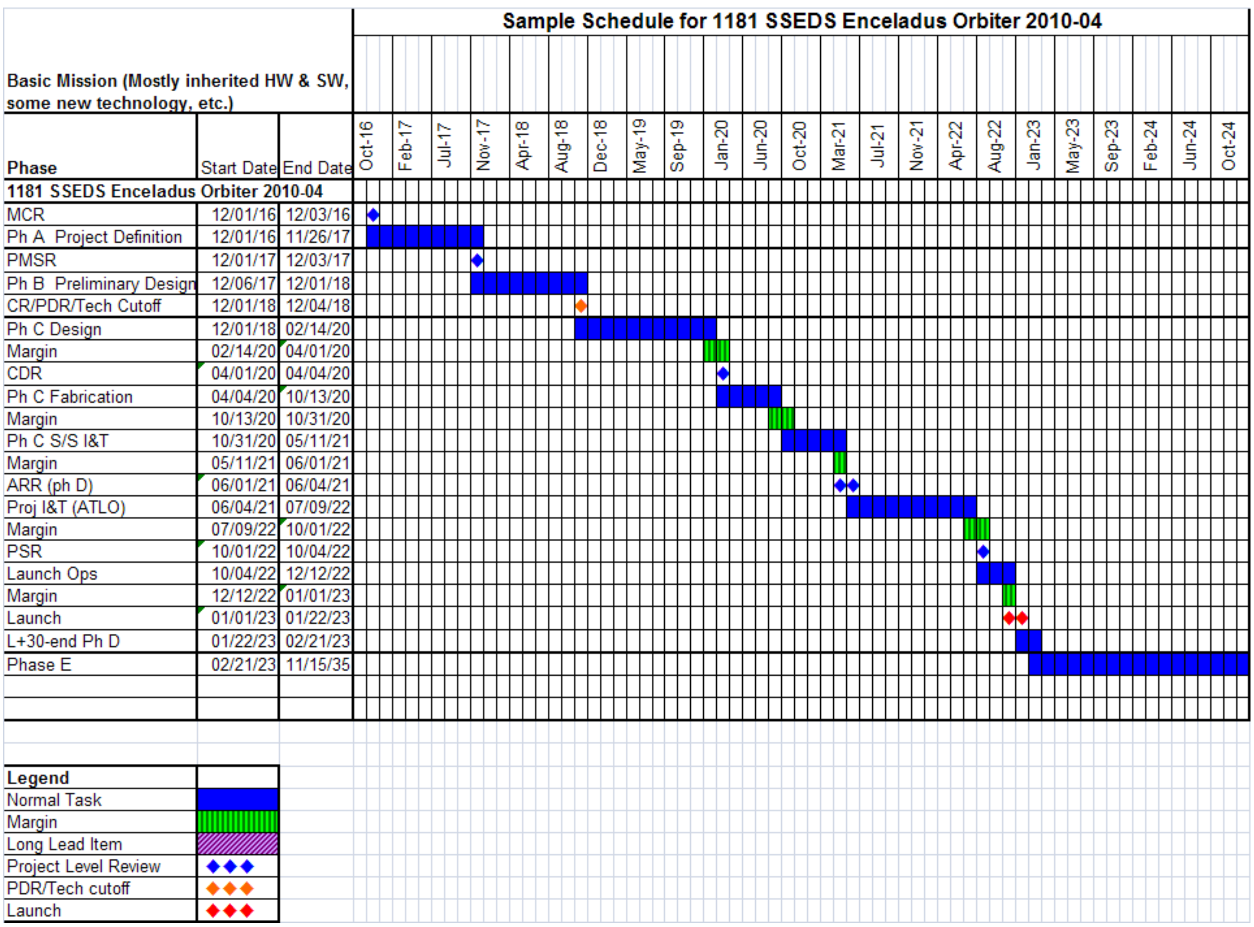

Figure 4-1. Mission Schedule

Table 4-1. Key Phase Duration

Project Phase	Duration (Months)
Phase A—Conceptual Design	12
Phase B—Preliminary Design	12
Phase C—Detailed Design	30
Phase D—Integration & Test	20
Phase E—Primary Mission Operations	155 (Option 1) 149 (Option 2)
Phase F—Extended Mission Operations	6
Start of Phase B to PDR	12
Start of Phase B to CDR	16
Start of Phase B to delivery of instruments	22*
Start of Phase B to delivery of flight system	22
System-level integration & test	7
Project total funded schedule reserve	7
Total development time Phase B–D	62

*Note: Indicated instrument development duration is notional.

Technology Development Plan

The only new technology for the EO mission would be the ASRG used in both options. It has a separate technology development project that is already separately funded and is currently undergoing lifetime testing. Any identified risks that are associated with the ASRGs are discussed in Section 3, Risk.

Although not considered a development risk because it is not required for the telecommunications subsystem, the UST being developed for the 2016 ExoMars mission would be included in the nominal EO design. If this new UST were not available in time for EO, existing SDSTs could be modified to provide the same functionality.

Development Schedule and Constraints

Because EO has been designed as a New Frontiers–class mission, no major schedule issues have been identified.

5. Mission Life-Cycle Cost

Costing Methodology and Basis of Estimate

The Decadal Survey guidelines for this study [2] were to provide independent design and costing analysis for each mission concept. The cost estimates summarized in this document were generated as part of a Pre-Phase-A preliminary concept study, are model-based, were prepared without consideration of potential industry participation, and do not constitute an implementation-cost commitment on the part of JPL or Caltech. Table 5-1 lists the guidelines and assumptions used to develop the cost estimate. The accuracy of the cost estimate is commensurate with the level of understanding of the mission concept, and should be viewed as indicative rather than predictive.

The cost estimation process begins with the customer providing the base information for the cost estimating models and defining the mission characteristics, such as

- Mission architecture
- Payload description
- Master equipment list (MEL) with heritage assumptions
- Functional block diagrams
- Spacecraft/payload resources (mass [kg], power [W], etc.)
- Phase A–F schedule
- Programmatic requirements
- Model-specific inputs

JPL has created 33 subsystem cost models, each owned, developed, and operated by the responsible line organization. These models are customized and calibrated using actual experience from completed JPL planetary missions. The models are under configuration management control and are used in an integrated and concurrent environment, so that the design and cost parameters are linked.

Table 5-1. Cost Guidelines and Assumptions

Decadal Survey Ground Rules	Value	Comments
Fixed-year dollars	2015	
Real-year dollars		
Reserves for Phases A–D	50%	
Reserves for Phase E	25%	
Schedule		
Phase A Start	2016	
Phase A–D duration	74 months	
Phase E duration for Option 1	155 months	
Phase E duration for Option 2	149 months	
Project end	2035	
General		
Total mission cost funding profile assumes the mission is totally funded by NASA and all significant work is performed in the US.		
Mission is costed using JPL's institutional cost models within the Team X environment.		

Decadal Survey Ground Rules	Value	Comments
All options are costed by WBS.		JPL WBS
Instrument costs are generated using NASA's Institutional Cost Model (NICM).		
Launch vehicle is costed using the fifth LV in the NASA Planetary Decadal Survey.	5-meter fairing; launch vehicle cost $257M	

Cost Estimates

Table 5-2 and Table 5-3 show workforce by phase for all science activities for the mission for Options 1 and 2, respectively.

Table 5-2. Science Team Workforce by Phase, Option 1

	A W-M	B W-M	C W-M	D W-M	E W-M	F W-M	Total W-M	Total W-Y
Science	**12.9**	**62.0**	**449.8**	**157.1**	**1279.8**	**81.2**	**2042.9**	**170.2**
Science Management	3.9	19.3	54.0	36.0	103.0	13.5	229.6	19.1
Science Office	3.9	19.3	54.0	36.0	103.0	13.5	229.6	19.1
Science Implementation	9.0	42.7	395.9	121.1	1176.8	67.7	1813.2	151.1
Participating Scientists	3.6	3.6	34.7	24.2	214.2	23.0	303.2	25.3
Teams Summary	5.4	39.1	361.2	96.9	962.6	44.8	1510.0	125.8

Table 5-3. Science Team Workforce by Phase, Option 2

	A W-M	B W-M	C W-M	D W-M	E W-M	F W-M	Total W-M	Total W-Y
Science	**12.9**	**62.0**	**449.8**	**157.1**	**1165.1**	**81.2**	**1928.1**	**160.7**
Science Management	3.9	19.3	54.0	36.0	85.0	13.5	211.7	17.6
Science Office	3.9	19.3	54.0	36.0	85.0	13.5	211.7	17.6
Science Implementation	9.0	42.7	395.9	121.1	1080.1	67.7	1716.5	143.0
Participating Scientists	3.6	3.6	34.7	24.2	183.6	23.0	272.6	22.7
Teams Summary	5.4	39.1	361.2	96.9	896.5	44.8	1443.8	120.3

Table 5-4 and Table 5-5 provide the cost for each of the two options in detail. The total cost is given by WBS element, based on fiscal year (FY) 2015 funds and the cost per real year (RY) funds for each of the years from project start (FY2017) to completion of Phase F (FY2033). It is assumed that the mission would be totally funded by NASA and that all significant work would be performed in the US.

Table 5-4. Total Mission Cost Funding Profile: Option 1

(FY costs[1] in Real Year Dollars, Totals in Real Year and 2015 Dollars)

Item	(all A) FY2017	FY2018	FY2019	FY2020	FY2021	FY2022	FY2023	FY2024	FY2025	FY2026	FY2027	FY2028	FY2029	FY2030	FY2031	FY2032	FY2033	FY2034	FY2035	FY2036	Total (Real Yr.)	Total (FY2015)
Cost	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M
Phase A concept study (included below)	9.7	0	0	0	0	0	0															
Technology development																						
	Phase A - D																					
Mission PM/SE/MA	1.0	9.2	19.6	21.0	26.0	33.2	8.6														118.6	102.1
Pre-launch science	0.2	1.7	3.7	3.9	4.9	6.2	1.6														22.2	19.1
Instrument PM/SE	0.1	0.9	1.9	2.0	2.5	3.2	0.8														11.4	9.9
Imager/MAC	0.1	1.0	2.2	2.4	2.9	3.7	1.0														13.4	11.5
Thermal mapper	0.1	1.1	2.3	2.5	3.1	3.9	1.0														14.1	12.1
Mass spectrometer	0.6	5.2	11.0	11.8	14.6	18.6	4.8														66.5	57.3
Dust analyzer	0.2	1.8	3.9	4.2	5.2	6.6	1.7														23.5	20.2
Magnetometer	0.1	0.7	1.5	1.7	2.0	2.6	0.7														9.3	8.0
Flight Element PM/SE	0.5	4.6	9.7	10.4	12.9	16.5	4.3														59.0	50.8
Flight Element (Orbiter)	2.8	26.6	56.5	60.6	75.1	95.9	24.8														342.3	294.7
MSI&T 2	0.3	2.5	5.4	5.8	13.4	24.5	6.4														58.2	49.4
Ground data system dev	0.2	2.1	4.5	4.8	5.9	7.6	2.0														27.1	23.3
Navigation & mission design	0.3	2.5	5.3	5.7	7.0	9.0	2.3														32.0	27.6
Total dev. w/o reserves	6.4	59.9	127.3	136.7	175.7	231.7	59.9														797.7	686.0
Development reserves	3.3	30.9	65.8	70.6	87.5	111.7	28.9														398.7	343.3
Total A–D development cost	9.7	90.8	193.1	207.3	263.2	343.4	88.8														1196.3	1029.3
Launch services			55.3	62.2	77.0	98.3	25.4														318.3	272.0
							Phase E															
Phase E science							3.1	4.3	4.4	4.6	4.7	4.8	4.9	5.1	5.2	5.3	5.5	5.6	5.8	1.8	65.1	44.3
Other Phase E cost							14.3	19.7	20.2	20.7	21.3	21.9	22.5	23.1	23.7	24.3	25.0	25.7	26.4	8.1	296.8	201.9
Phase E reserves							4.0	5.5	5.6	5.8	5.9	6.1	6.2	6.4	6.6	6.8	7.0	7.1	7.3	2.3	82.5	56.1
Total Phase E							21.4	29.4	30.2	31.1	31.9	32.8	33.6	34.6	35.5	36.4	37.4	38.4	39.5	12.2	444.5	302.3
Education/outreach	0.02	0.2	0.4	0.5	0.6	0.8	0.2	0.7	0.7	0.7	0.7	0.8	0.8	0.8	0.8	0.8	0.9	0.9	0.9	0.3	12.5	9.3
Other (specify)																					0.0	0
Total Cost	$ 9.8	$ 91.0	$ 248.9	$ 269.9	$ 340.8	$ 442.5	$ 135.9	$ 30.1	$ 30.9	$ 31.8	$ 32.6	$ 33.5	$ 34.4	$ 35.4	$ 36.3	$ 37.3	$ 38.3	$ 39.3	$ 40.4	$ 12.4	$ 1,972	$ 1,613
																					Total Mission Cost	$ 1,613

[1] Costs include all costs including any fee.

[2] MSI&T—Mission System Integration and Test and preparation for operations

Table 5-5. Total Mission Cost Funding Profile: Option 2

(FY costs[1] in Real Year Dollars, Totals in Real Year and 2015 Dollars)

Item	(all A) FY2017	FY2018	FY2019	FY2020	FY2021	FY2022	FY2023	FY2024	FY2025	FY2026	FY2027	FY2028	FY2029	FY2030	FY2031	FY2032	FY2033	FY2034	FY2035	FY2036	Total (Real Yr.)	Total (FY2015)
Cost	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	$M	SM	$M	$M	SM	$M	$M	$M
Phase A concept stucy (included below)	9.7	0	0	0	0	0	0															
Technology development																						
	Phase A - D																					
Mission PM/SE/MA	1.0	9.2	19.5	21.0	26.0	33.2	8.6														118.4	101.9
Pre-launch science	0.2	1.7	3.7	3.9	4.9	6.2	1.6														22.2	19.1
Instrument PM/SE	0.1	0.9	1.9	2.0	2.5	3.2	0.8														11.4	9.9
Imager/MAC	0.1	1.0	2.2	2.4	2.9	3.7	1.0														13.4	11.5
Thermal mapper	0.1	1.1	2.3	2.5	3.1	3.9	1.0														14.1	12.1
Mass spectrometer	0.6	5.2	11.0	11.8	14.6	18.6	4.8														66.5	57.3
Dust analyzer	0.2	1.8	3.9	4.2	5.1	6.6	1.7														23.5	20.2
Magnetometer	0.1	0.7	1.5	1.6	2.0	2.6	0.7														9.3	8.0
Flight Element PM/SE	0.5	4.6	9.7	10.4	12.9	16.5	4.3														59.0	50.8
Flight Element (Orbiter)	2.8	26.6	56.5	60.6	75.2	95.9	24.8														342.5	294.9
MSI&T 2	0.3	2.5	5.3	5.7	13.3	24.4	6.3														57.6	48.9
Ground data system dev	0.2	2.1	4.5	4.8	5.9	7.6	2.0														27.0	23.2
Navigation & mission design	0.2	2.1	4.5	4.8	6.0	7.7	2.0														27.3	23.5
Total dev. w/o reserves	6.4	59.5	126.4	135.7	174.5	230.2	59.5														792.2	681.3
Development reserves	3.3	30.7	65.3	70.1	86.9	110.9	28.7														395.9	340.9
Total A–D development cost	9.7	90.2	191.7	205.8	261.4	341.1	88.2														1188.1	1022.2
Launch services			55.3	62.2	77.0	98.3	25.4														318.3	272.0
							Phase E															
Phase E science							3.0	4.2	4.3	4.4	4.5	4.7	4.8	4.9	5.0	5.2	5.3	5.5	4.2	0.2	60.2	41.3
Other Phase E cost							14.4	19.8	20.3	20.8	21.4	22.0	22.6	23.2	23.8	24.4	25.1	25.8	19.9	1.2	284.5	194.9
Phase E reserves							4.0	5.4	5.6	5.7	5.9	6.0	6.2	6.4	6.5	6.7	6.9	7.1	5.5	0.3	78.3	53.6
Total Phase E							21.4	29.4	30.2	31.0	31.8	32.7	33.5	34.5	35.4	36.3	37.3	38.3	29.6	1.7	423.0	289.8
Education/outreach	0.02	0.2	0.4	0.5	0.6	0.7	0.2	0.7	0.7	0.7	0.8	0.8	0.8	0.8	0.8	0.9	0.9	0.9	0.7	0.04	12.2	9.2
Other (specify)																					0.0	0
Total Cost	$ 9.7	$ 90.4	$ 247.5	$ 268.5	$ 339.0	$ 440.1	$ 135.2	$ 30.1	$ 30.9	$ 31.7	$ 32.6	$ 33.4	$ 34.3	$ 35.3	$ 36.2	$ 37.2	$ 38.2	$ 39.2	$ 30.3	$ 1.8	$ 1,942	$ 1,593
																					Total Mission Cost	$ 1,593

[1] Costs include all costs including any fee.

[2] MSI&T—Mission System Integration and Test and preparation for operations

Appendix A. Acronyms

ACS	attitude control subsystem
ASRG	Advanced Stirling Radioisotope Generator
BOL	beginning of life
BWG	beam waveguide
C&DH	command and data handling
CBE	current best estimate
CDA	cosmic dust analyzer (Cassini)
CIRS	composite infrared spectrometer (Cassini)
CML	concept maturity level
CPHP	capillary-pumped heat pipe
CRCC	critical relay control card
Diviner	lunar radiometer (LRO)
DLA	declination of launch asymptote
DSN	Deep Space Network
EO	Enceladus Orbiter
EOI	Enceladus orbit insertion
EOL	end of life
FOV	field of view
FY	fiscal year
HGA	high-gain antenna
IMU	inertial measurement unit
INMS	ion and neutral mass spectrometer (Cassini)
ISS	imaging science subsystem (Cassini)
JPL	Jet Propulsion Laboratory
LA	laser altimeter
LGA	low-gain antenna
LRO	Lunar Reconnaissance Orbiter
LV	launch vehicle
MAC	medium-angle camera
MAG	magnetometer (Galileo)
MCIC	MSAP motor control and interface card
MEL	master equipment list
MEV	maximum expected value
MGA	medium-gain antenna
MLI	multilayered insulation
MMRTG	multimission radioisotope thermoelectric generator
MREU	MSAP remote engineering unit
MS	mass spectrometer
MSAP	Multi-Mission System Architectural Platform
MSIA	MSAP serial interface assembly
MTIF	MSAP telecom interface card
MVIC	multispectral visible imaging camera (New Horizons)
NICM	NASA Institutional Cost Model
NPR	NASA Procedural Requirements
NRC	National Research Council
NTO	nitrogen tetroxide
PMD	propellant management device
PP	planetary protection
PPO	Planetary Protection Officer
PSDS	Planetary Science Decadal Survey
Ralph	multispectral imager (New Horizons)
RHU	radioisotope heater unit
RMA	Rapid Mission Architecture
RPS	radioisotope power system
RPWI	radio and plasma wave instrument
SDST	small deep space transponder
STM	science traceability matrix
TCM	trajectory correction maneuver
TID	total ionizing dose
TRL	technology readiness level
TWTA	travelling-wave tube amplifier
UST	universal space transponder
WBS	work breakdown structure
VIMS	visual and infrared mapping spectrometer (Cassini)

Appendix B. References

[1] Jet Propulsion Laboratory, “Design, Verification/Validation & Ops Principles for Flight Systems (Design Principles),” Rev. 3. JPL Rules! DocID 43913, Dec. 11, 2006.

[2] National Aeronautics and Space Administration, “Ground Rules for Mission Concept Studies in Support of Planetary Decadal Survey,” Rev. 2, Nov. 10, 2009.

[3] Lockheed Martin Energy Systems, “ASRG Functional Description,” Contract No. DE-AC07-00SF22191, Document 912IC002144, October 2009.

[4] Jet Propulsion Laboratory, “Enceladus Study Final Report,” Planetary Science Decadal Survey, JPL Rapid Mission Architecture (RMA), April 2010.

[5] S. Campagnola, N. J. Strange, and R. P. Russell, “A fast tour design method using non-tangent V-infinity leveraging transfers,” presented at the AAS/AIAA Space Flight Mechanics Meeting, Paper AAS 10-164, San Diego, California, February 2010.

[6] R. P. Russell and M. Lara, “On the design of an Enceladus science orbit,” presented at the AIAA/AAS Astrodynamics Specialist Conference and Exhibit, Paper AIAA 2008-7072, Honolulu, Hawaii, Aug. 18–21, 2008.[7] National Aeronautics and Space Administration, “Planetary Protection Provisions for Robotic Extraterrestrial Missions,” NASA Program Requirements (NPR) 8020.12C, April 27, 2005.

[7] National Aeronautics and Space Administration, “Planetary Protection Provisions for Robotic Extraterrestrial Missions,” NPR 8020.12C, April 27, 2005.

Appendix C. DSN Tracking Schedule

Table C-1 is a DSN tracking schedule for Option 1; Table C-2 is a DSN tracking schedule for Option 2.

Table C-1. DSN Tracking Schedule for Option 1

Support Period		Antenna Size (m)	Service Year	Hrs per Track	Tracks per Wk	Weeks Required
(#)	Name (Description)					
1	Launch and operations	34 BWG	2023	8	21.0	2.0
2	Launch and operations	34 BWG	2023	8	14.0	2.0
3	Inner solar system cruise (w/ flybys)—cruise	34 BWG	2023	8	1.0	223.0
4	Inner solar system cruise (w/ flybys)—TCMs	34 BWG	2023	8	7.0	12.0
5	Inner solar system cruise (w/ flybys)—annual health checks	34 BWG	2023	8	7.0	3.0
6	Cruise to Saturn—cruise	34 BWG	2023	8	1.0	176.0
7	Cruise to Saturn—TCMs	34 BWG	2023	8	7.0	3.0
8	Cruise to Saturn—annual health checks	34 BWG	2023	8	7.0	2.0
9	Encounter prep and Saturn orbit insertion—Cruise	34 BWG	2023	8	2.0	20.0
10	Encounter prep and Saturn orbit insertion—approach hvy	34 BWG	2023	8	21.0	3.0
10	DDOR	34 BWG	2023	1	4.0	3.0
11	Encounter prep and Saturn orbit insertion—approach lt	34 BWG	2023	8	14.0	3.0
11	DDOR	34 BWG	2023	1	3.0	3.0
12	Orbit pumpdown/flybys—Tour	34 BWG	2023	4	7.0	115.0
13	Orbit pumpdown/flybys—TCMs/Flybys	34 BWG	2023	8	7.0	62.0
14	Enceladus orbit insertion and science—orbit	34 BWG	2023	8	7.0	54.0

Table C-2. DSN Tracking Schedule for Option 2

Support Period		Antenna Size (m)	Service Year	Hrs per Track	Tracks per Wk	Weeks Required
(#)	Name (Description)					
1	Launch and Operations	34 BWG	2023	8	21.0	2.0
2	Launch and Operations	34 BWG	2023	8	14.0	2.0
3	Inner Solar System Cruise (w/ Flybys)—Cruise	34 BWG	2023	8	1.0	223.0
4	Inner Solar System Cruise (w/ Flybys)—TCMs	34 BWG	2023	8	7.0	12.0
5	Inner Solar System Cruise (w/ Flybys)—annual health checks	34 BWG	2023	8	7.0	3.0
6	Cruise to Saturn—Cruise	34 BWG	2023	8	1.0	176.0
7	Cruise to Saturn—TCMs	34 BWG	2023	8	7.0	3.0
8	Cruise to Saturn—annual health checks	34 BWG	2023	8	7.0	2.0
9	Encounter prep and Saturn orbit insertion—cruise	34 BWG	2023	8	2.0	20.0
10	Encounter prep and Saturn orbit insertion—approach hvy	34 BWG	2023	8	21.0	3.0
10	DDOR	34 BWG	2023	1	4.0	3.0
11	Encounter prep and Saturn orbit insertion—approach lt	34 BWG	2023	8	14.0	3.0
11	DDOR	34 BWG	2023	1	3.0	3.0
12	Orbit pumpdown/flybys—tour	34 BWG	2023	4	7.0	115.0
13	Orbit pumpdown/flybys—TCMs/flybys	34 BWG	2023	8	7.0	62.0
14	Enceladus orbit insertion and science—orbit	34 BWG	2023	8	7.0	26.0

Appendix D. Master Equipment List

This appendix provides the Master Equipment Lists (MELs) for Options 1 and 2.

MEL for Option 1

	CBE Mass Per Unit	# of Units	Current Basic Est.	%-Unc. (% of CBE)	Predicted Basic Est.
Launch Mass			**3241.4 kg**	**11%**	**3588.5 kg**
Launch Vehicle PLA			**0.0 kg**	**30%**	**0.0 kg**
Stack (w/ Wet Element)			**3241.4 kg**	**11%**	**3588.5 kg**
Useable Propellant			**2434.3 kg**	**0%**	**2434.3 kg**
Stack (w/ Dry Element)			**807.2 kg**	**43%**	**1154.3 kg**
Carried Elements			**0.0 kg**	**0%**	**0.0 kg**
Dry Element			**807.2 kg**	**43%**	**1154.3 kg**
Wet Element			**3241.4 kg**	**11%**	**3588.5 kg**
Useable Propellant			**2434.3 kg**	**0%**	**2434.3 kg**
System 1: Biprop			**2434.3 kg**	**0%**	**2434.3 kg**
Propellant			**2359.3 kg**	**0%**	**2359.3 kg**
Pressurant			**11.3 kg**	**0%**	**11.3 kg**
Residuals			**63.7 kg**	**0%**	**63.7 kg**
Dry Element			**807.2 kg**	**43%**	**1154.3 kg**
System Contingency			**129.3 kg**	**16%**	
Subsystem Heritage Contingency			**217.8 kg**	**27%**	
Payload			**56.0 kg**	**30%**	**72.8 kg**
Instruments		**5**	**56.0 kg**	**30%**	**72.8 kg**
Imager/MAC	10.0 kg	1	10.0 kg	30%	13.0 kg
Thermal mapper	12.0 kg	1	12.0 kg	30%	15.6 kg
Mass spectrometer	25.0 kg	1	25.0 kg	30%	32.5 kg
Dust analyzer	3.0 kg	1	3.0 kg	30%	3.9 kg
Magnetometer	6.0 kg	1	6.0 kg	30%	7.8 kg
Additional Payload		**0**	**0.0 kg**	**30%**	**0.0 kg**
Bus			**751.2 kg**	**27%**	**952.2 kg**
Attitude Control		**18**	**40.0 kg**	**10%**	**44.0 kg**
Sun Sensor	0.0 kg	8.0	0.0 kg	10%	0.0 kg
Star Tracker	1.5 kg	2.0	3.0 kg	10%	3.2 kg
IMU	4.0 kg	2.0	8.0 kg	10%	8.8 kg
Reaction Wheel	7.0 kg	4.0	28.0 kg	10%	30.8 kg
Electronics	1.0 kg	1.0	1.0 kg	10%	1.1 kg
Command & Data		**20**	**20.8 kg**	**15%**	**23.8 kg**
Processor: RAD750	0.6 kg	2	1.1 kg	5%	1.2 kg
Memory: FMC	1.6 kg	2	3.2 kg	5%	3.4 kg
Telecom_I_F: MTIF	0.7 kg	2	1.5 kg	5%	1.5 kg
General_I_F: MSIA	0.7 kg	4	2.8 kg	5%	3.0 kg
Custom_Special_Function_Board: CRC	0.7 kg	2	1.3 kg	17%	1.5 kg
Power: CEPCU	1.2 kg	2	2.3 kg	10%	2.5 kg
Backplane: CPCI backplane (6 slots)	0.6 kg	2	1.2 kg	30%	1.6 kg
Chassis: CDH chassis (6 slot)	2.9 kg	2	5.7 kg	30%	7.4 kg
Analog_I_F: MREU	0.8 kg	2	1.6 kg	6%	1.7 kg

	CBE Mass Per Unit	# of Units	Current Basic Est.	%-Unc. (% of CBE)	Predicted Basic Est.
Power		**24**	**117.6 kg**	**30%**	**152.8 kg**
Li-ION (Secondary Battery)	7.9 kg	4	31.5 kg	30%	41.0 kg
Advanced Stirling (ASRG-850C)	22.0 kg	3	65.9 kg	30%	85.7 kg
Chassis	6.0 kg	1	6.0 kg	30%	7.8 kg
Load Switches Boards	0.8 kg	2	1.6 kg	30%	2.1 kg
Thruster Drivers* Boards	0.8 kg	2	1.6 kg	30%	2.1 kg
Pyro Switches* Boards	0.8 kg	2	1.6 kg	30%	2.1 kg
Houskeeping DC-DC Converters* Boards	1.0 kg	2	2.0 kg	30%	2.6 kg
Power/Shunt Control* Boards	1.0 kg	2	2.0 kg	30%	2.6 kg
Battery Control Boards	0.8 kg	4	3.2 kg	30%	4.2 kg
Diodes* Boards	0.8 kg	1	0.8 kg	30%	1.0 kg
Shielding	1.3 kg	1	1.3 kg	30%	1.7 kg
Propulsion		**95**	**135.9 kg**	**27%**	**172.3 kg**
System 1: Biprop		95	135.9 kg	27%	172.3 kg
Hardware		95	135.9 kg	27%	172.3 kg
Gas Service Valve	0.2 kg	4	0.9 kg	2%	0.9 kg
HP Latch Valve	0.4 kg	2	0.7 kg	2%	0.7 kg
Solenoid Valve	0.4 kg	4	1.4 kg	2%	1.4 kg
HP Transducer	0.3 kg	2	0.5 kg	2%	0.6 kg
Gas Filter	0.1 kg	2	0.2 kg	2%	0.2 kg
NC Pyro Valve	0.2 kg	2	0.4 kg	2%	0.4 kg
Temp. Sensor	0.0 kg	4	0.0 kg	2%	0.0 kg
Liq. Service Valve	0.3 kg	2	0.6 kg	2%	0.6 kg
Test Service Valve	0.2 kg	2	0.5 kg	2%	0.5 kg
LP Transducer	0.3 kg	8	2.2 kg	2%	2.2 kg
Liq. Filter	0.5 kg	2	0.9 kg	2%	0.9 kg
LP Latch Valve	0.4 kg	4	1.4 kg	2%	1.4 kg
NC Pyro Valve	0.2 kg	8	1.4 kg	2%	1.5 kg
Mass Flow Control	0.0 kg	2	0.1 kg	2%	0.1 kg
Temp. Sensor	0.0 kg	20	0.2 kg	2%	0.2 kg
Lines, Fittings, Misc.	10.0 kg	1	10.0 kg	50%	15.0 kg
DM Monoprop Thrusters 1	0.4 kg	4	1.6 kg	2%	1.7 kg
DM Monoprop Thrusters 2	0.3 kg	16	5.3 kg	10%	5.8 kg
Biprop Main Engine	8.1 kg	1	8.1 kg	10%	9.0 kg
Fuel Pressurant Tank	25.8 kg	1	25.8 kg	30%	33.6 kg
Ox Pressurant Tank	8.8 kg	1	8.8 kg	30%	11.5 kg
Fuel Tanks	47.4 kg	1	47.4 kg	30%	61.7 kg
Oxidizer Tanks	17.4 kg	1	17.4 kg	30%	22.6 kg
Mechanical		**9**	**302.3 kg**	**30%**	**392.9 kg**
Struc. & Mech.		7	230.3 kg	30%	299.4 kg
Primary Structure	166.5 kg	1	166.5 kg	30%	216.4 kg
Secondary Structure	12.7 kg	1	12.7 kg	30%	16.4 kg
Support Structure (Dust Analyzer and M	8.0 kg	1	8.0 kg	30%	10.4 kg
Magnetometer Boom, Hinge, and Laun	1.6 kg	1	1.6 kg	30%	2.0 kg
Thruster Cluster Support Booms	10.0 kg	1	10.0 kg	30%	13.0 kg
Main Engine Cover and Actuator	20.0 kg	1	20.0 kg	30%	26.0 kg
Integration Hardware	11.7 kg	1	11.7 kg	30%	15.1 kg
Adapter, Spacecraft side	24.7 kg	1	24.7 kg	30%	32.1 kg
Cabling Harness	47.2 kg	1	47.2 kg	30%	61.4 kg

	CBE Mass Per Unit	# of Units	Current Basic Est.	%-Unc. (% of CBE)	Predicted Basic Est.
Telecom		**58**	**71.5 kg**	**18%**	**84.4 kg**
X/Ka-HGA 3.0m diam Parabolic High Gain A	33.7 kg	1	33.7 kg	20%	40.4 kg
X-MGA (19dB) MER	0.6 kg	1	0.6 kg	10%	0.7 kg
X-LGA (4dB) MER	0.8 kg	2	1.5 kg	10%	1.7 kg
UST (Dual Band)	4.5 kg	2	9.0 kg	20%	10.8 kg
X-band TWTA, RF=25W	3.0 kg	2	6.0 kg	10%	6.6 kg
Ka-band TWTA, RF<100W	2.9 kg	2	5.8 kg	10%	6.4 kg
Hybrid Coupler	0.0 kg	2	0.0 kg	10%	0.0 kg
Coax Transfer Switch (CXS)	0.1 kg	2	0.3 kg	10%	0.3 kg
X-band Diplexer, high isolation	0.8 kg	2	1.6 kg	10%	1.8 kg
Waveguide Transfer Switch (WGTS)	0.4 kg	3	1.1 kg	10%	1.3 kg
Hybrid Coupler	0.0 kg	1	0.0 kg	10%	0.0 kg
Coax Transfer Switch (CXS)	0.1 kg	1	0.1 kg	10%	0.1 kg
Ka-Band Diplexer, high isolation	0.6 kg	2	1.2 kg	10%	1.3 kg
Waveguide Transfer Switch (WGTS)	0.4 kg	1	0.4 kg	10%	0.4 kg
Coax Cable, flex (190)	0.2 kg	10	1.6 kg	25%	2.0 kg
WR-112 WG, rigid (Al)	0.6 kg	11	6.1 kg	25%	7.6 kg
Coax Cable, flex (190)	0.2 kg	8	1.3 kg	25%	1.6 kg
WR-34 WG, rigid (Al)	0.2 kg	5	1.1 kg	25%	1.4 kg
Thermal		**280**	**63.3 kg**	**29%**	**82.0 kg**
Multilayer Insulation (MLI)	0.4 kg	72	26.8 kg	30%	34.9 kg
Thermal Surfaces		40	1.0 kg	30%	1.3 kg
General	0.0 kg	40	1.0 kg	30%	1.3 kg
Thermal Conduction Control		1	1.0 kg	30%	1.2 kg
General	1.0 kg	1	1.0 kg	30%	1.2 kg
Heaters		13	1.1 kg	30%	1.4 kg
Custom	0.1 kg	5	0.3 kg	30%	0.3 kg
Propulsion Line Heaters	0.1 kg	8	0.8 kg	30%	1.0 kg
Temperature Sensors		100	2.0 kg	15%	2.3 kg
Thermistors	0.0 kg	100	2.0 kg	15%	2.3 kg
Thermostats		10	0.2 kg	15%	0.2 kg
Mechanical	0.0 kg	10	0.2 kg	15%	0.2 kg
Thermal Radiator (Area=m2)	7.5 kg	2	11.3 kg	30%	14.6 kg
Heat Pipes		20	15.0 kg	30%	19.5 kg
Loop HP	0.8 kg	20	15.0 kg	30%	19.5 kg
RHU's	0.1 kg	20	2.0 kg	30%	2.6 kg
Other Components		2	3.0 kg	30%	3.9 kg
Shunt Radiator	2.5 kg	1	2.5 kg	30%	3.3 kg
Sun/Venus Shield	0.5 kg	1	0.5 kg	30%	0.7 kg

MEL for Option 2

	CBE Mass Per Unit	# of Units	Current Basic Est.	%-Unc. (% of CBE)	Predicted Basic Est.
Launch Mass			**3214.3 kg**	**11%**	**3560.2 kg**
Launch Vehicle PLA			**0.0 kg**	**30%**	**0.0 kg**
Stack (w/ Wet Element)			**3214.3 kg**	**11%**	**3560.2 kg**
Useable Propellant			**2409.8 kg**	**0%**	**2409.8 kg**
Stack (w/ Dry Element)			**804.5 kg**	**43%**	**1150.4 kg**
Carried Elements			**0.0 kg**	**0%**	**0.0 kg**
Dry Element			**804.5 kg**	**43%**	**1150.4 kg**
Wet Element			**3214.3 kg**	**11%**	**3560.2 kg**
Useable Propellant			**2409.8 kg**	**0%**	**2409.8 kg**
System 1: Biprop			**2409.8 kg**	**0%**	**2409.8 kg**
Propellant			**2335.5 kg**	**0%**	**2335.5 kg**
Pressurant			**11.2 kg**	**0%**	**11.2 kg**
Residuals			**63.1 kg**	**0%**	**63.1 kg**
Dry Element			**804.5 kg**	**43%**	**1150.4 kg**
System Contingency			**128.9 kg**	**16%**	
Subsystem Heritage Contingency			**217.0 kg**	**27%**	
Payload			**56.0 kg**	**30%**	**72.8 kg**
Instruments		**5**	**56.0 kg**	**30%**	**72.8 kg**
Imager/MAC	10.0 kg	1	10.0 kg	30%	13.0 kg
Thermal mapper	12.0 kg	1	12.0 kg	30%	15.6 kg
Mass spectrometer	25.0 kg	1	25.0 kg	30%	32.5 kg
Dust analyzer	3.0 kg	1	3.0 kg	30%	3.9 kg
Magnetometer	6.0 kg	1	6.0 kg	30%	7.8 kg
Additional Payload		**0**	**0.0 kg**	**30%**	**0.0 kg**
Bus			**748.5 kg**	**27%**	**948.7 kg**
Attitude Control		**18**	**40.0 kg**	**10%**	**44.0 kg**
Sun Sensor	0.0 kg	8.0	0.0 kg	10%	0.0 kg
Star Tracker	1.5 kg	2.0	3.0 kg	10%	3.2 kg
IMU	4.0 kg	2.0	8.0 kg	10%	8.8 kg
Reaction Wheel	7.0 kg	4.0	28.0 kg	10%	30.8 kg
Electronics	1.0 kg	1.0	1.0 kg	10%	1.1 kg
Command & Data		**20**	**20.8 kg**	**15%**	**23.8 kg**
Processor: RAD750	0.6 kg	2	1.1 kg	5%	1.2 kg
Memory: FMC	1.6 kg	2	3.2 kg	5%	3.4 kg
Telecom_I_F: MTIF	0.7 kg	2	1.5 kg	5%	1.5 kg
General_I_F: MSIA	0.7 kg	4	2.8 kg	5%	3.0 kg
Custom_Special_Function_Board: CRC	0.7 kg	2	1.3 kg	17%	1.5 kg
Power: CEPCU	1.2 kg	2	2.3 kg	10%	2.5 kg
Backplane: CPCI backplane (6 slots)	0.6 kg	2	1.2 kg	30%	1.6 kg
Chassis: CDH chassis (6 slot)	2.9 kg	2	5.7 kg	30%	7.4 kg
Analog_I_F: MREU	0.8 kg	2	1.6 kg	6%	1.7 kg

	CBE Mass Per Unit	# of Units	Current Basic Est.	%-Unc. (% of CBE)	Predicted Basic Est.
Power		**24**	**117.6 kg**	**30%**	**152.8 kg**
Li-ION (Secondary Battery)	7.9 kg	4	31.5 kg	30%	41.0 kg
Advanced Stirling (ASRG-850C)	22.0 kg	3	65.9 kg	30%	85.7 kg
Chassis	6.0 kg	1	6.0 kg	30%	7.8 kg
Load Switches Boards	0.8 kg	2	1.6 kg	30%	2.1 kg
Thruster Drivers* Boards	0.8 kg	2	1.6 kg	30%	2.1 kg
Pyro Switches* Boards	0.8 kg	2	1.6 kg	30%	2.1 kg
Houskeeping DC-DC Converters* Boards	1.0 kg	2	2.0 kg	30%	2.6 kg
Power/Shunt Control* Boards	1.0 kg	2	2.0 kg	30%	2.6 kg
Battery Control Boards	0.8 kg	4	3.2 kg	30%	4.2 kg
Diodes* Boards	0.8 kg	1	0.8 kg	30%	1.0 kg
Shielding	1.3 kg	1	1.3 kg	30%	1.7 kg
Propulsion		**95**	**134.7 kg**	**27%**	**170.8 kg**
System 1: Biprop		95	134.7 kg	27%	170.8 kg
Hardware		95	134.7 kg	27%	170.8 kg
Gas Service Valve	0.2 kg	4	0.9 kg	2%	0.9 kg
HP Latch Valve	0.4 kg	2	0.7 kg	2%	0.7 kg
Solenoid Valve	0.4 kg	4	1.4 kg	2%	1.4 kg
HP Transducer	0.3 kg	2	0.5 kg	2%	0.6 kg
Gas Filter	0.1 kg	2	0.2 kg	2%	0.2 kg
NC Pyro Valve	0.2 kg	2	0.4 kg	2%	0.4 kg
Temp. Sensor	0.0 kg	4	0.0 kg	2%	0.0 kg
Liq. Service Valve	0.3 kg	2	0.6 kg	2%	0.6 kg
Test Service Valve	0.2 kg	2	0.5 kg	2%	0.5 kg
LP Transducer	0.3 kg	8	2.2 kg	2%	2.2 kg
Liq. Filter	0.5 kg	2	0.9 kg	2%	0.9 kg
LP Latch Valve	0.4 kg	4	1.4 kg	2%	1.4 kg
NC Pyro Valve	0.2 kg	8	1.4 kg	2%	1.5 kg
Mass Flow Control	0.0 kg	2	0.1 kg	2%	0.1 kg
Temp. Sensor	0.0 kg	20	0.2 kg	2%	0.2 kg
Lines, Fittings, Misc.	10.0 kg	1	10.0 kg	50%	15.0 kg
DM Monoprop Thrusters 1	0.4 kg	4	1.6 kg	2%	1.7 kg
DM Monoprop Thrusters 2	0.3 kg	16	5.3 kg	10%	5.8 kg
Biprop Main Engine	8.1 kg	1	8.1 kg	10%	9.0 kg
Fuel Pressurant Tank	25.5 kg	1	25.5 kg	30%	33.1 kg
Ox Pressurant Tank	8.8 kg	1	8.8 kg	30%	11.4 kg
Fuel Tanks	46.8 kg	1	46.8 kg	30%	60.9 kg
Oxidizer Tanks	17.3 kg	1	17.3 kg	30%	22.5 kg
Mechanical		**9**	**300.8 kg**	**30%**	**391.1 kg**
Struc. & Mech.		7	229.2 kg	30%	297.9 kg
Primary Structure	165.4 kg	1	165.4 kg	30%	215.0 kg
Secondary Structure	12.6 kg	1	12.6 kg	30%	16.4 kg
Support Structure (Dust Analyzer and M	8.0 kg	1	8.0 kg	30%	10.4 kg
Magnetometer Boom, Hinge, and Laun	1.6 kg	1	1.6 kg	30%	2.0 kg
Thruster Cluster Support Booms	10.0 kg	1	10.0 kg	30%	13.0 kg
Main Engine Cover and Actuator	20.0 kg	1	20.0 kg	30%	26.0 kg
Integration Hardware	11.6 kg	1	11.6 kg	30%	15.0 kg
Adapter, Spacecraft side	24.5 kg	1	24.5 kg	30%	31.8 kg
Cabling Harness	47.2 kg	1	47.2 kg	30%	61.3 kg

	CBE Mass Per Unit	# of Units	Current Basic Est.	%-Unc. (% of CBE)	Predicted Basic Est.
Telecom		**58**	**71.5 kg**	**18%**	**84.4 kg**
X/Ka-HGA 3.0m diam Parabolic High Gain A	33.7 kg	1	33.7 kg	20%	40.4 kg
X-MGA (19dB) MER	0.6 kg	1	0.6 kg	10%	0.7 kg
X-LGA (4dB) MER	0.8 kg	2	1.5 kg	10%	1.7 kg
UST (Dual Band)	4.5 kg	2	9.0 kg	20%	10.8 kg
X-band TWTA, RF=25W	3.0 kg	2	6.0 kg	10%	6.6 kg
Ka-band TWTA, RF<100W	2.9 kg	2	5.8 kg	10%	6.4 kg
Hybrid Coupler	0.0 kg	2	0.0 kg	10%	0.0 kg
Coax Transfer Switch (CXS)	0.1 kg	2	0.3 kg	10%	0.3 kg
X-band Diplexer, high isolation	0.8 kg	2	1.6 kg	10%	1.8 kg
Waveguide Transfer Switch (WGTS)	0.4 kg	3	1.1 kg	10%	1.3 kg
Hybrid Coupler	0.0 kg	1	0.0 kg	10%	0.0 kg
Coax Transfer Switch (CXS)	0.1 kg	1	0.1 kg	10%	0.1 kg
Ka-Band Diplexer, high isolation	0.6 kg	2	1.2 kg	10%	1.3 kg
Waveguide Transfer Switch (WGTS)	0.4 kg	1	0.4 kg	10%	0.4 kg
Coax Cable, flex (190)	0.2 kg	10	1.6 kg	25%	2.0 kg
WR-112 WG, rigid (Al)	0.6 kg	11	6.1 kg	25%	7.6 kg
Coax Cable, flex (190)	0.2 kg	8	1.3 kg	25%	1.6 kg
WR-34 WG, rigid (Al)	0.2 kg	5	1.1 kg	25%	1.4 kg
Thermal		**279**	**63.2 kg**	**29%**	**81.8 kg**
Multilayer Insulation (MLI)	0.4 kg	71	26.7 kg	30%	34.7 kg
Thermal Surfaces		40	1.0 kg	30%	1.3 kg
General	0.0 kg	40	1.0 kg	30%	1.3 kg
Thermal Conduction Control		1	0.9 kg	30%	1.2 kg
General	0.9 kg	1	0.9 kg	30%	1.2 kg
Heaters		13	1.1 kg	30%	1.4 kg
Custom	0.1 kg	5	0.3 kg	30%	0.3 kg
Propulsion Line Heaters	0.1 kg	8	0.8 kg	30%	1.0 kg
Temperature Sensors		100	2.0 kg	15%	2.3 kg
Thermistors	0.0 kg	100	2.0 kg	15%	2.3 kg
Thermostats		10	0.2 kg	15%	0.2 kg
Mechanical	0.0 kg	10	0.2 kg	15%	0.2 kg
Thermal Radiator (Area=m2)	7.5 kg	2	11.3 kg	30%	14.6 kg
Heat Pipes		20	15.0 kg	30%	19.5 kg
Loop HP	0.8 kg	20	15.0 kg	30%	19.5 kg
RHU's	0.1 kg	20	2.0 kg	30%	2.6 kg
Other Components		2	3.0 kg	30%	3.9 kg
Shunt Radiator	2.5 kg	1	2.5 kg	30%	3.3 kg
Sun / Venus Shield	0.5 kg	1	0.5 kg	30%	0.7 kg

Appendix E. Saturn Tour

Figures E-1 through E-5 show the flybys that would be included in the Saturn tour prior to Enceladus orbit insertion.

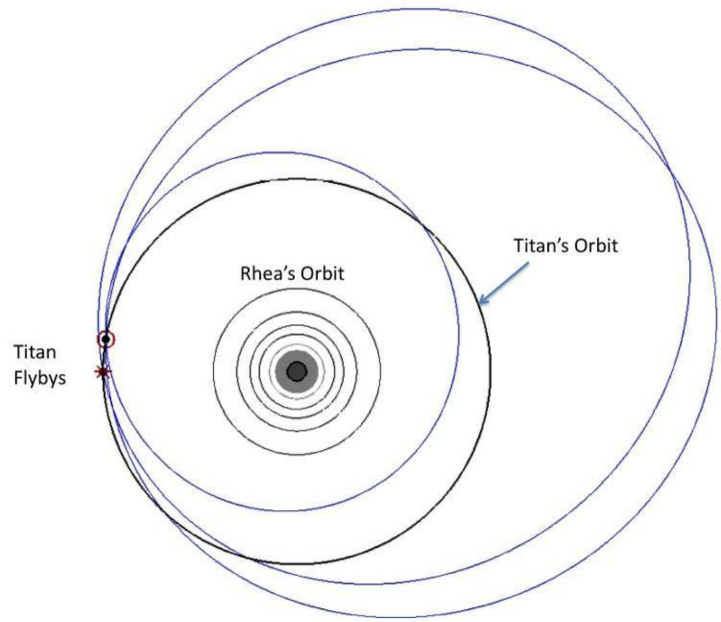

Figure E-1. The Titan tour would include three flybys of Titan and would end with an encounter with Rhea.

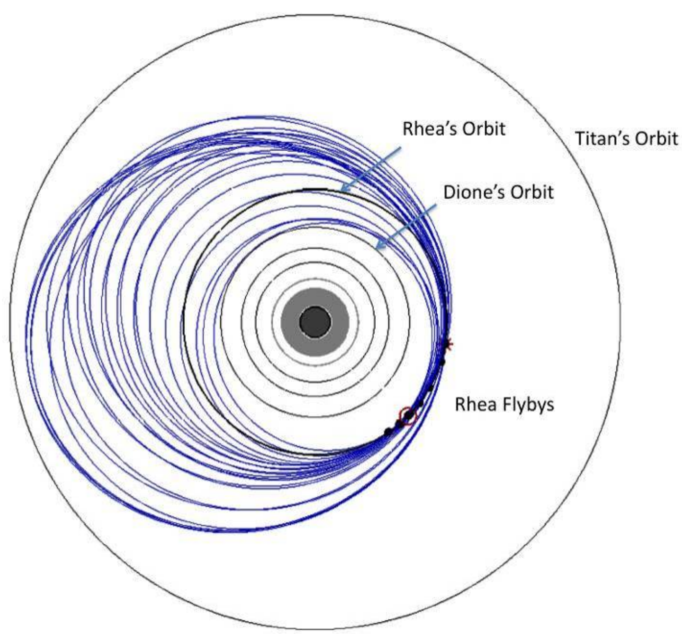

Figure E-2. The Rhea tour would include 15 flybys of Rhea and would end with an encounter with Dione.

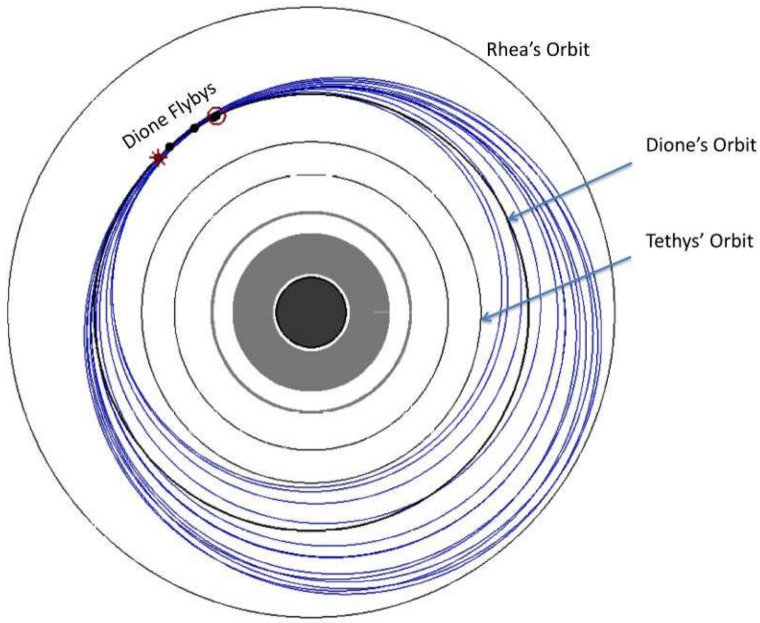

Figure E-3. The Dione tour would include 10 flybys of Dione and would end with an encounter with Tethys.

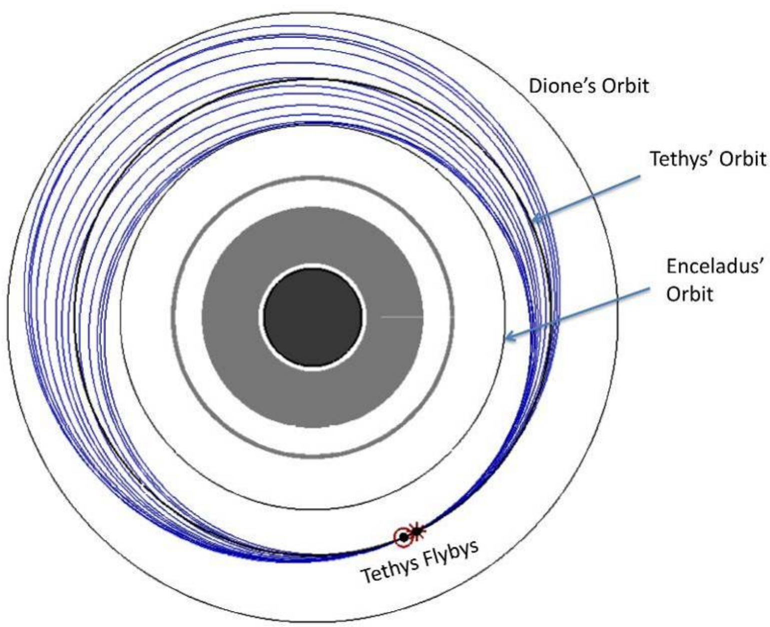

Figure E-4. The Tethys tour would include 12 flybys of Tethys and would end with an encounter with Enceladus.

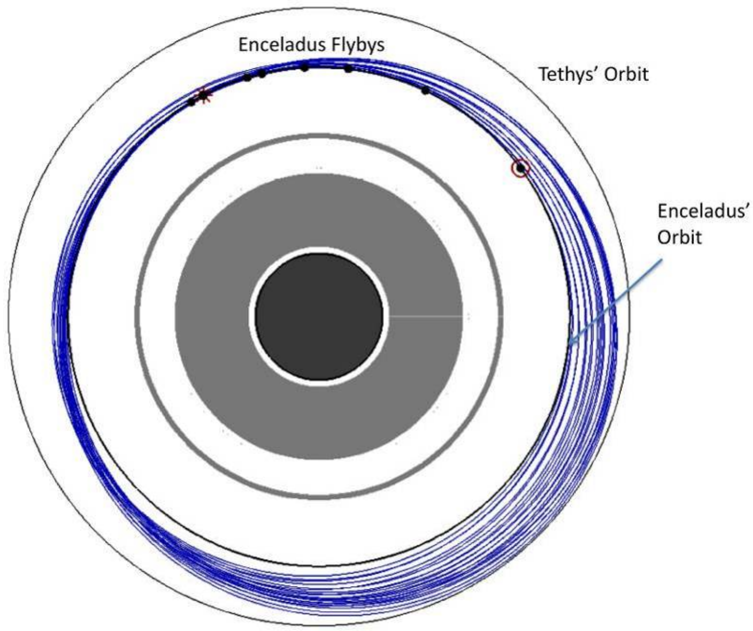

Figure E-5. The Enceladus tour would include 12 engineering flybys and 10 science flybys of Enceladus and would end with Enceladus orbit insertion.