COMPASS Final Report: Enceladus Solar Electric Propulsion Stage

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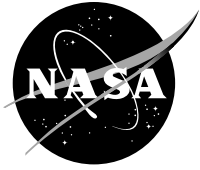
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Acknowledgments

The COMPASS team wishes to thank the support of the Engineering Directorate, and specifically branch Chief Maria Babula and mission analysts of the Mission Design and Analysis Branch for their unending support and high caliber of engineer support.

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1.0   Executive Summary

The results of the NASA Glenn Research Center (GRC) COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) internal Solar Electric Propulsion (SEP) stage design are documented in this report (Figure 1.1). The SEP Stage was designed to deliver a science probe to Saturn (the probe design was performed separately by the NASA Goddard Space Flight Center's (GSFC) Integrated Mission Design Center (IMDC)). The SEP Stage delivers the 2444 kg probe on a Saturn trajectory with a hyperbolic arrival velocity of 5.4 km/s. The design carried 30 percent mass, 10 percent power, and 6 percent propellant margins. The SEP Stage relies on the probe for substantial guidance, navigation and control (GN&C), command and data handling (C&DH), and Communications functions. The stage is configured to carry the probe and to minimize the packaging interference between the probe and the stage. The propulsion system consisted of a 1+1 (one active, one spare) configuration of gimbaled 7 kW NASA Evolutionary Xenon Thruster (NEXT) ion propulsion thrusters with a throughput of 309 kg Xe propellant. Two 9350 W GaAs triple junction (at 1 Astronomical Unit (AU), includes 10 percent margin) ultra-flex solar arrays provided power to the stage, with Li-ion batteries for launch and contingency operations power. The base structure was an Al-Li hexagonal skin-stringer frame built to withstand launch loads. A passive thermal control system consisted of heat pipes to 'north and south' radiator panels, multilayer insulation (MLI) and heaters for the Xe tank. All systems except tanks and solar arrays were designed to be single fault tolerant.

Figure 1.1.—Mission overview.
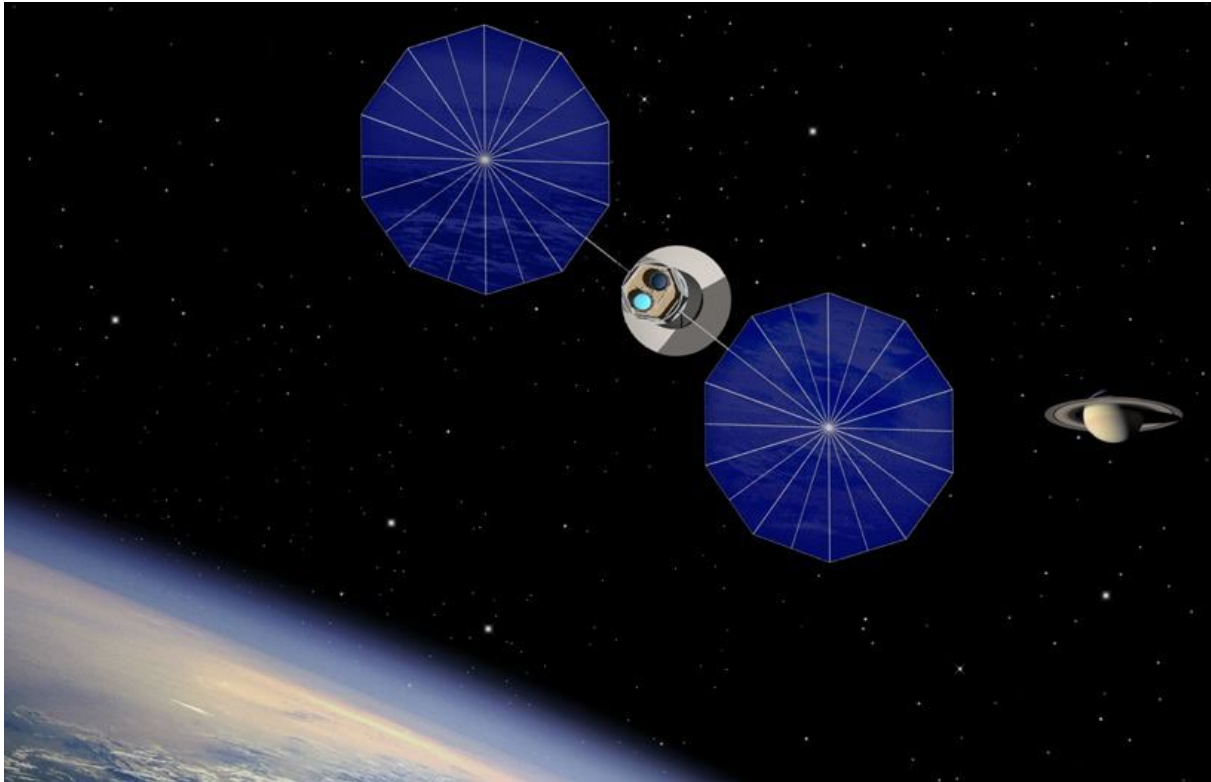
Figure 1.2.—Conceptual SEP Stage to deliver science payload package to Saturn.

Table 1.1 lists the major components and descriptions of each major subsystems that make up the SEP transfer stage.

TABLE 1.1.—MISSION AND S/C SUMMARY

Subsystem area	Details	Total mass with growth
Top level SEP Stage system	SEP Stage to provide probe mass of ~2500 kg to a Saturn transfer trajectory Figures of merit (FOMs): Probe mass, stage cost, min probe capture velocity	1026 kg
Mission, Operations, GN&C	1+1 SEP Stage, 1- to 3-AU orbit range, <3 yr SEP Stage ops, 8 yr for Probe to reach Saturn, 3% additional Xe for nav/trajectory and other errors, 1000 km Earth flyby, SEP Stage thrust duty cycle 90%, ion thrusters can act as pointing control during thrusting for yaw and pitch, ion thrusters may be able to off-set solar perturbations (during thrusting) and/or dump momentum wheels	
Attitude Control System (ACS)	Science Probe provides ACS for all coasting, commands ion thruster gimbal, solar array pointing, and probe provides roll control during all phases. control and actuators sized to accommodate stage mass properties	1.2 kg
Launch	Atlas 551, $C_3 = 27$ km^2/s^2, 20 day launch window, Adapter: 65 in, D1666, 10% launch margin taken from performance. Launch Loads: ~5g (See Section 2.9)	3864 kg
Science	Science probe to be designed in IMDC subsequent session. Volume and mass to be determined. Probe allowable mass output from SEP Stage performance analysis	2444 kg
Power	18.5 kW power (with 200 W housekeeping), each array oversized by 1.5 kW to address degradation (cell, string, diode losses), two Ultraflex arrays design based on ST-8, CEV Li ion batteries for pre-array deployment and contingency operations	321 kg
Propulsion	One active 7 kW NEXT Ion engine with one cold spare, two Power Processing Units (PPU) (one is spare), 1% Xe unusable, <2 in. below adapter at launch, 66 cm thruster to thruster center separation, 99% of ion plume in 15° half angle cone, off-the-shelf (OTS) tanks	168 kg

TABLE 1.1.—MISSION AND S/C SUMMARY

Subsystem area	Details	Total mass with growth
Structures and Mechanisms	Primary: Hexagonal, 63 in. diameter, Truss, Al-Li, Carry ~2500 kg probe and major stage components, Secondary: first order design based on component masses Two thruster gimbals, two-axis range of motion: ±19°, ±17°, Single axis SA drives, solar array/boom deployment, louvers on radiators	98 kg
Communications	Probe provides all communications support for Stage, Electric Propulsion (EP) thrust vector constrains communications link design	none
C&DH	Use Digital Control Interface Units (DCIU) to run all probe functions; rely on probe for C&DH	12 kg
Thermal	Passive heat-pipe system, body mounted radiators (north and south SADA faces) (main loads 500 W from PPU, 500 W from power distribution units (PDU), 100 layers MLI between thruster platform and S/C to prevent heat leak back, active tank heaters to prevent two-phase Xe, louvers on radiators Radiation level, micrometeoroid environment	39 kg

The deployed SEP Stage developed in this study is shown in Figure 1.3. The two Ultraflex solar arrays are shown deployed and span 18.9 m at their widest point across. The arm placement of the solar arrays onto the stage is such that the 49° half angle thruster plume cone does not interfere with the arrays. A 3 m diameter dish antenna is placed on top of the, as yet, not designed science probe. Note that the science probe is shown in this design simply as a conical shape in order to allow for Expendable Launch Vehicle (ELV) packaging and stage layout during the design study.
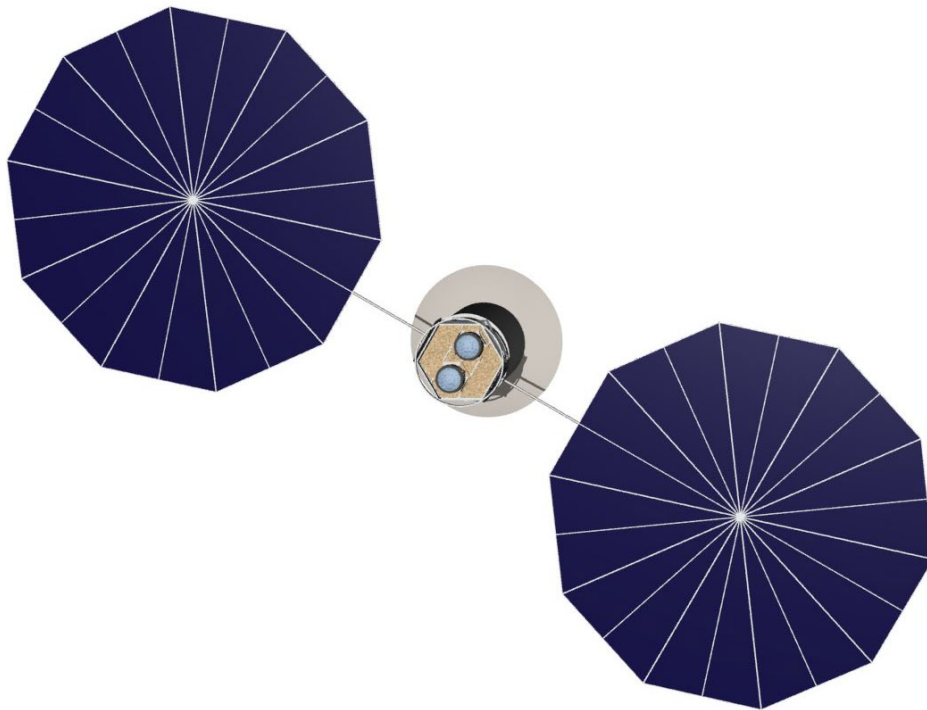
Figure 1.3.—Conceptual SEP Stage deployed configuration.

2.0   Study Background and Assumptions

2.1   Introduction

Cassini performed several close flybys of Enceladus in 2005, revealing the moon's surface and environment in greater detail. In particular, the probe discovered a water-rich plume venting from the moon's South Polar Region (Figure 2.1). This discovery, along with the presence of escaping internal heat and very few (if any) impact craters in the South Polar Region, shows that Enceladus is geologically active today.

NASA is considering several of Saturn's moon for the next Flagship mission, and in light of Enceladus' increased importance in terms of the interest of the scientific community, GSFCs IMDC was tasked with designing a science probe mission to Saturn's moons of Titan and Enceladus. To help with this effort, the COMPASS team was tasked to design a SEP interplanetary stage to be used to send a "to be designed" IMDC scientific probe to the Saturn system. The NEXT ion propulsion system is well suited to future missions to the Saturn system. NEXT is used within the inner solar system, in combination with a Venus or Earth gravity assist, to establish a fast transfer to the Saturn system. The NEXT system elements are accommodated in a separable SEP module, or are integrated into the main S/C bus, depending on the mission architecture and performance requirements.
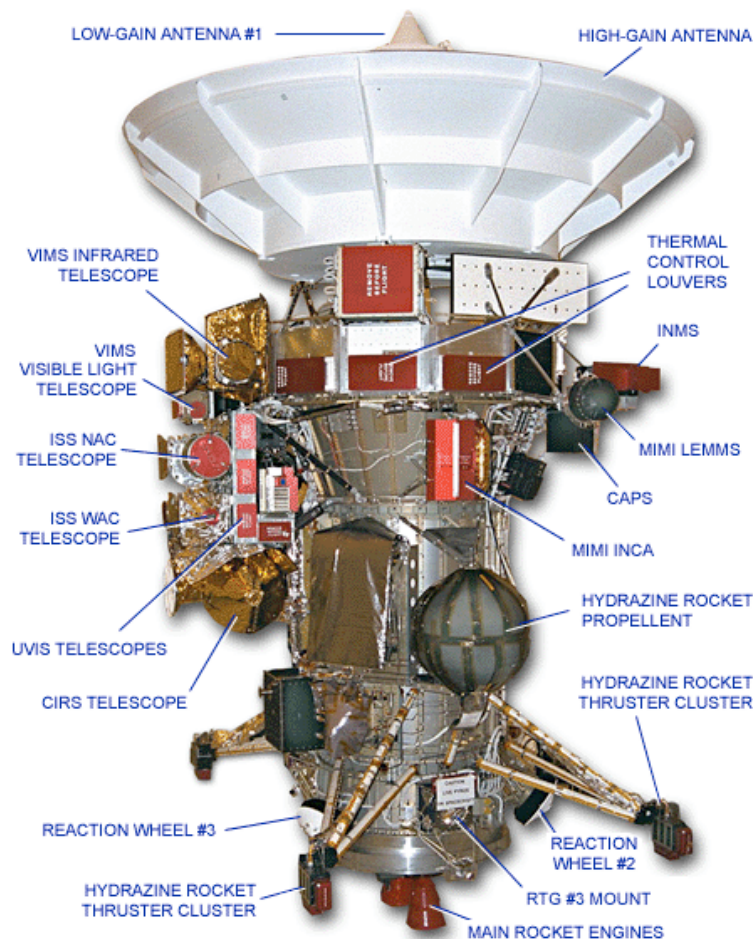
Figure 2.1.—Cassini S/C (probe example).

2.2   Assumptions

The SEP Stage will be designed to deliver an, as yet not designed, science probe to Saturn. The assumptions for the probe to be delivered to Saturn are as follows

- Large antenna
- Chemical propulsion
- Radioisotope powered

The SEP Stage provides structural interface with the Atlas launch vehicle. In the stowed configuration, the SEP Stage solar arrays remain below the probe separation plane. For this study session the SEP Stage concept design assumes the following basis of design

- Estimate probe needs of mass and size based on past Saturn probes
- Design SEP Stage based on past SEP S/C (Deep Space 1 (DS-1) and Dawn)
- Utilize OTS products and components where possible to minimize costs
- Single fault tolerant
- New Frontiers Class (Flagship class in subsequent studies)
- Mass growth (contingency) based on ANSI/AIAA R-020A-1999, additional growth carried at system level to total 30 percent for the stage

2.3   Growth, Contingency and Margin Policy

Mass Growth: At the time of this study, the COMPASS team used the ANSI/AIAA R-020A-1999, *Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles* for mass growth allowance calculations and allocations. Table 2.1 shows the Percent Mass Growth Allowance separated into a matrix specified by level of design maturity and specific subsystem. The percent growth factors are applied to each subsystem, after which the total system growth at the vehicle level is calculated. The COMPASS team desired total growth to be 30 percent. Therefore additional growth is carried at the system level in order to achieve a total system growth of a 30 percent limit on the dry mass of the system. Note that for designs requiring propellant, growth in propellant is either book kept in the propellant itself or in the ΔV used to calculate the propellant necessary to fly a mission.

The COMPASS team uses the Discover Announcement of Opportunity (AO) definitions of Contingency and mass Margin.

From the Discovery AO: Definitions of Contingency and Mass Contingency (or Reserve), when added to a resource, results in the maximum expected value for that resource. Percent contingency is the value of the contingency divided by the value of the resource less the contingency.

Margin is the difference between the maximum possible value of a resource (the physical limit or the agree-to limit) and the maximum expected value for a resource. Percent margin for a resource is the available margin divided by its maximum expected value.

Power Growth: The COMPASS team uses a 30 percent margin on the bottoms up power requirements in modeling the power system. See Section 3.2.2 for the power system assumptions.

TABLE 2.1—PERCENT MASS GROWTH ALLOWANCE

Code	Design maturity (basis for mass determination)	Percent mass growth allowance									
		Electrical/electronic components			Structure	Thermal control	Propulsion	Batteries	Wire harnesses	Mechanisms	Instrumentation
		0 to 5 kg	5 to 15 kg	>15 kg							
E	Estimated (preliminary sketches)	30	20	15	18	18	18	20	50	18	50
L	Layout (or major modification of existing hardware)	25	20	15	12	12	12	15	30	12	30
P	Pre-Release Drawings (or minor modification of existing hardware)	20	15	10	8	8	8	10	25	8	25
C	Released Drawings (calculated values)	10	5	5	4	4	4	5	5	4	5
X	Existing Hardware (actual mass from another program)	3	3	3	2	2	2	3	3	2	3
A	Actual Mass (measured flight hardware)	0	0	0	0	0	0	0	0	0	0
CFE	Customer Furnished Equipment	0	0	0	0	0	0	0	0	0	0

2.4   Mission Description

The baseline mission used an Atlas 551 to launch the SEP Stage with payload to a C_3 of 27 km^2/s^2. The following baseline assumptions for the technology of the SEP Stage as related to trajectory modeling were used in the baseline mission modeling.

- 20 kW Class SEP Stage and Earth flyby to accelerate the SEP + probe
    – SEP Stage operates 1 AU ~ 3 AU then is disposed
    – Earth flyby safety considerations (probe plutonium powered) are to be taken into account during final mission modeling
    – Two NEXT engines (1 +1), ~300 kg Xe propellant
    – Two solar arrays for beginning of life (BOL) of 17 kW (only 7 kW bus used—extra solar array used at >1 AU)
- 20 day launch window chosen, worst case payload and Xe propellant load for window chosen as design basis

2.4.1   Mission Analysis Assumptions

The following mission assumptions were used in the trajectory modeling of the low thrust SEP trajectory to Saturn.

Overall Assumptions for Mission Analysis
- The initial departure epoch is in the 2015 to 2016 time period. Epochs between 2015 and 2025 also are to be achievable, but not necessarily with the same propellant loading.
- Use a SEP system consisting of solar arrays, power distribution system and an EPS consisting of some number of thrusters, power processing units, propellant tanks and a distribution system for the propellant.

- Maximize the injected mass in the vicinity of Saturn while minimizing the hyperbolic excess velocity (V_{hp}) at Saturn arrival.
- Use a launch vehicle from the United States' existing fleet of ELVs. The ELV puts the S/C on an escape hyperbola.
- The mission would use a Venus or Earth gravity assist(s) to increase the injected S/C mass.

Power Assumptions for Mission Analysis

- The SEP system model includes a generic solar array whose power output is proportional to the inverse of the square of the distance from the S/C to the sun. This assumption provides a degree of conservatism since most solar arrays can provide slightly more power than inverse square because of low incidence and low temperature affects. In addition, array degradation is incorporated into the power model.
- Modeling power degradation with an inverse-square ratio without any specific array coefficients
- Sweep of power level (P_0)
    - P_0 = 10-, 15-, 16- and 17-kW cases investigated
- S/C power (SAP) = $1 + 2/R + 3/R^2$ (this permits modeling of increased S/C demand with distance (R), i.e., turning on heaters) where R = the distance to the Sun
- 2 percent array degradation per year

Propulsion Assumptions for Mission Analysis

The nomenclature used for the propulsion system is $N+1$, where N represents the number of thruster strings required for the optimized mission, and the "+1" represents the spare string.

- 1+1 Configuration (one thruster with redundant thruster available)
- SEP system operations are assumed to occur with a 90 percent propulsive duty cycle during thrusting phases. The 10 percent downtime allows the S/C to perform navigation and communications functions on a regular basis.

The NEXT thruster can be operated across a wide throttle range. This capability has been modeled with curve fits of the extremes of the throttle table, representing either a high specific impulse (I_{sp}) mode or high thrust mode. Based upon the results of a previous study, the high I_{sp} model for NEXT is assumed to provide a higher final mass than the high thrust setting and was thus used for these analyses

- NEXT coefficients for high I_{sp} end of the performance envelope used as input to SEPTOP and MALTO for polynomial approximation of thrust and mass flow rate of the NEXT thruster
    - ct = 3.646709187×10^{-2}, 3.774558883×10^{-3}, 6.882669254×10^{-3}, –6.815530390×10^{-4}, 2.334052793×10^{-5}
    - cm = 3.630027151×10^{-6}, –1.726624312×10^{-6}, 6.466718696×10^{-7}, –7.184580335×10^{-8}, 2.891651950×10^{-9}
    - Operating power ranges on the y axis (Figure 2.2)
        - p_{min} = 1.252159689
        - p_{max} = 7.252

2.4.2   Mission Analysis Analytic Methods

The method was to find a low thrust trajectory to Saturn modeled using SEPTOP (2003 edition) exclusively. SEPTOP is a more precise version of VARITOP in that it offers the user a much more accurate simulation of a "real" EP thruster. It shares much with VARITOP including solutions techniques and the robustness of problems it can solve. It generally is used after a preliminary solution is in hand from VARITOP. The performance estimates SEPTOP generates are much more accurate. SEPTOP is a product of the NASA Jet Propulsion Laboratory (JPL) and is only available to employees of NASA centers. Non-NASA personnel should contact JPL directly for SEPTOP.
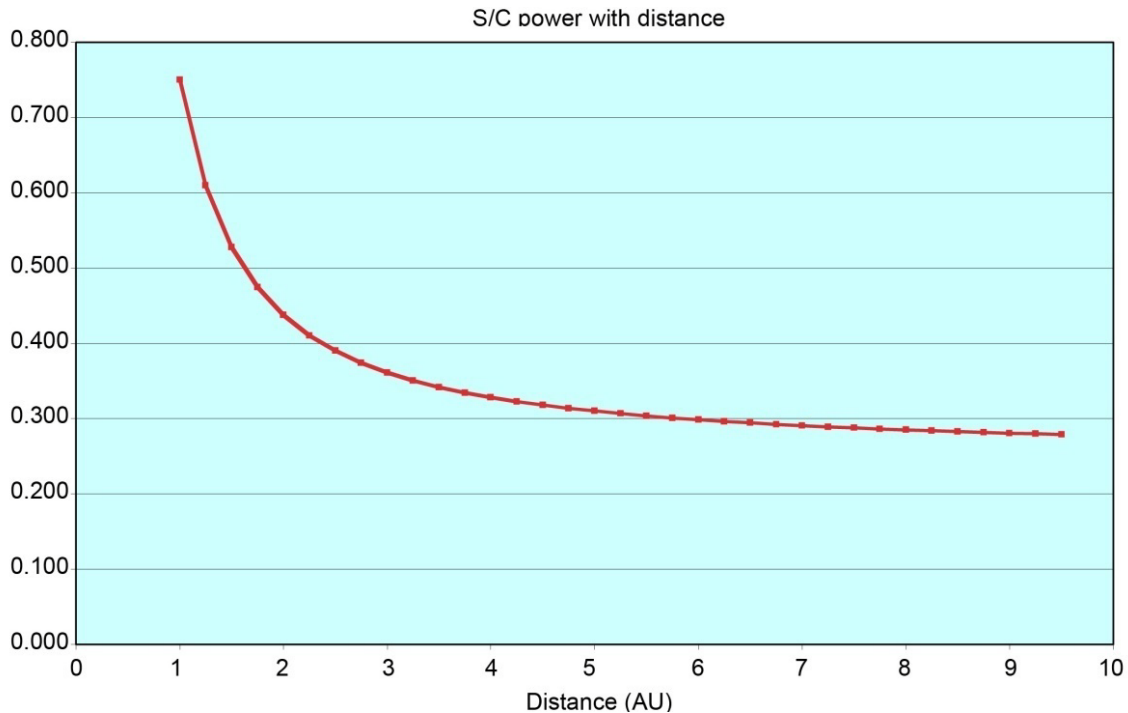
Figure 2.2.—Available S/C power from the solar arrays versus distance from the Sun.

The optimization objective was to maximize payload and minimize arrival hyperbolic arrival velocity. Maximum injected mass is achieved by trajectory optimization. The V_{hp} is minimized by parametrically varying the transfer time from Earth departure to Saturn arrival. Although it may not be obvious, for a low thrust trajectory V_{hp} decreases with increased trip time until it attains a minimum value and then slowly increases. For this study, the computer program SEPTOP provided the optimization while the utility programs Vartable and Newpost (two VARITOP and SEPTOP utilities) used with Microsoft Excel permitted quick visualization of the optimization results. SEPTOP, as provided by Carl Sauer of JPL, obtains optimal trajectories using algorithms based upon classical optimal control theory. It has the capability of determining minimal propellant consumption while satisfying the transversality conditions associated the constraints on the mission. It provides the capability of selecting the optimal number of operating thrusters in multiple thruster configurations and of determining the correct throttle setting when presented with coefficients of the thruster and mass flow as a function of thrust system input power.

A SEP rendezvous with Saturn is virtually impossible without a final thrusting period near Saturn. Because solar power drops below a minimum level necessary to power the thrusters around 3.5 AU, no EP thrusting is possible near Saturn. Hence, the SEP system essentially puts the S/C on a Saturn flyby trajectory. For the Saturn Flyby with Probes mission, this is sufficient, with SEP performing the majority of the mission velocity change required. For S/C capture into orbit around Titan or Saturn (for the Enceladus mission), a separate high thrust propulsion system or aero capture vehicle provides the necessary velocity change.

In order to determine the optimal trajectory, the following gravity assist configurations were examined.

- Solar Electric Earth Gravity Assist (SEEGA)—chosen as optimal
- Solar Electric Venus Gravity Assist (SEVGA)—much less performance
- Solar Electric Jupiter Gravity Assist (SEJGA)—not very affective
- Solar Electric Earth Jupiter Gravity Assist (SEEJGA)—not available

The next step in analysis would be a full verification with a detailed NEXT thruster model.

2.4.3 Mission Trajectory Details

It was determined that a single Earth gravity assist (EGA) was superior to either a Venus or a Jupiter fly-by. The less massive Venus imparts a smaller velocity increase even though the thruster system operates at a high efficiency level for a considerable duration in preparation for the fly-by and thereby also gains energy. Both Earth and Venus fly-bys are available across the 2015 to 2025 period. The JGA was not available throughout the entire 2015 to 2025 period because Jupiter moved away from an advantageous position for the Saturn mission.

Figure 2.3 shows the trajectory from Earth to Saturn, calling out the areas of S/C thrusting and spacecraft (S/C) coasting along the arc. This particular solution does not have any coast phases during the inner solar system section of the trajectory. The Earth gravity assist occurs at approximately 670 days into the mission, after which the power, and resulting thrust tails off as the S/C travels towards Saturn. The SEP Stage and probe are launched on an Atlas 551 to a C_3 of 27 km^2/s^2. The 20 kW Class SEP Stage goes through one Earth flyby to pick up acceleration. The SEP Stage thrusters then perform maneuvers and operate from 1 AU ~ 3 AU. At 3 AU, the SEP Stage is then disposed. Earth flyby safety considerations are taken into account in the choice of Earth flyby distance (note the Probe is plutonium powered). A 20-day launch window was chosen, worst-case payload and Xe propellant load for window chosen as design basis.
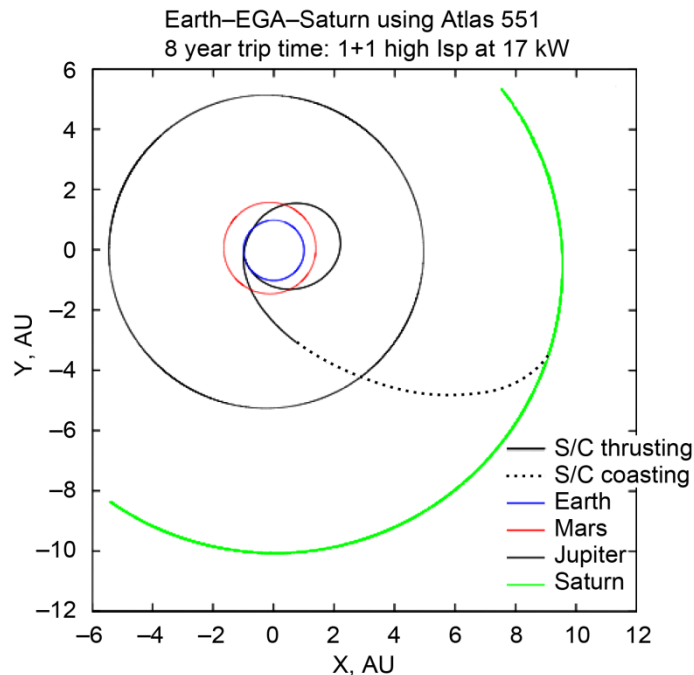
Figure 2.3.—EGA to Saturn trajectory.

SEP Thruster and Power Details

- Two NEXT engines (1 +1), ~300 kg Xe propellant
- Two solar arrays for BOL of 17 kW nominal (only maximum of 7 kW used—extra solar array area allows more thrust at >1 AU)
  - Note that the final design resulted in an 18.5 kW once the additional 30 percent power margin was built into the design.

2.4.4 Launch Window Sensitivity

Launch flexibility, including launch year capability and launch opportunities and windows of significant duration, is of primary interest to planetary science mission customers. Optimization of the nominal launch capability across launch dates spanning 10 yr revealed that the baseline configuration was capable of delivering roughly similar usable S/C masses annually with small variations in arrival V_{hp}. (Figure 2.4). Furthermore, the required SEP system propellant mass changed very little, while the mass delivered to the SEP starting orbit by the launch vehicle did vary. These variations arose from the annual variations in the relative Earth-Saturn geometry that required differing amounts of the Atlas 551 launch capability. Net mass penalties from years 2015 to 2018 can be mitigated by adjusting mission parameters, such as trip time and gravity assist altitude, for each launch year. Launch opportunity capability was addressed in these analyses, but is not presented here. Launch opportunity analysis using SEPTOP requires careful manipulation of constraints and optimization parameters. Results were achieved through several trial cases, with minimal impact to the primary mission capability, but further launch strategy development and analysis is warranted. Detailed launch window analyses have not yet been initiated.

2.5 Mission Operations

2.5.1 Duty Cycle Budget and Operations Phases

The trajectory was run with a 90 percent duty cycle assumed. This duty cycle assumption was used to account for the following unmodeled activities accounting for the other 10 percent of thrusting time:

- 0.1 percent reserved for solar torque offset
- 1 percent used for timing events (e.g., checkout, flyby targeting)
- 1.9 percent saved for margin
- 7 percent remaining for communications uplink/navigation update
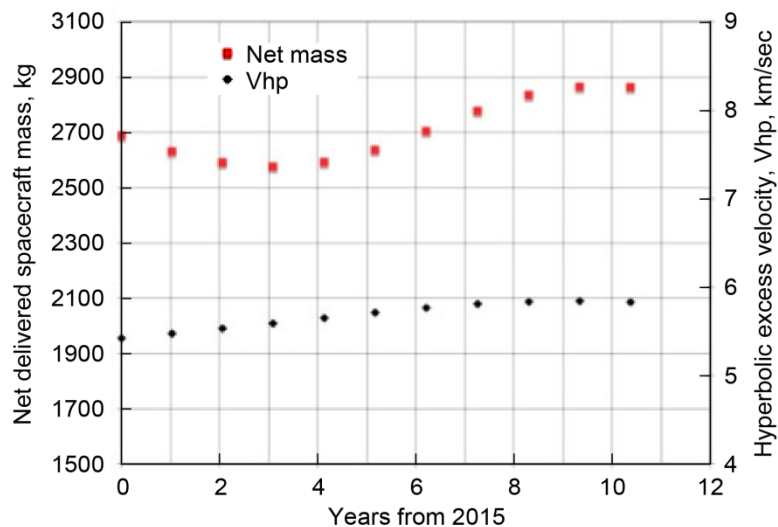
Figure 2.4.—Net delivered S/C mass for given launch date.

The remaining 7 percent duty cycle translates into roughly 12 hr/week for comm. and nav. update and associated slews. Table 2.2 lists the Autonav commands used in the DS-1 mission. These commands and their time length were used to estimate the 7 percent remaining coasting period necessary in the SEP trajectory design.

TABLE 2.2.—MISSION PHASE AND OPERATING MODES

Command name	Description	Argument	Usage	Time required
Nav_Do_OD	Perform orbit determination	None	1/wk	10 to 100 min
Nav_Do_TCM	Execute a TCM	Duration	1/wk	1.5 to 24 hr
Nav_IPS_Off_Mes*	Notify Nav of forced "engine off"	None	1/wk	1 s
Nav_Man_Plan	Perform maneuver planning	None	1/wk	10 to 200 min
Nav_Photo_Op	Perform a nav picture taking processing session, edit and store data	Duration	1/wk	1.5 to 8 hr
Nav_Reset*	Stop all Navexec state machines	None	Seldom*	1 s
Nav_Set_IPS	Start a mission burn	None	1/wk	5 min
Nav_Start_Encntr	Start an encounter sequence	Seq. ID	4/encounter	1 min
Nav_Update_IPS	Update the thrust vector during a mission burn	None	2/day	1 min
Nav_Change_Mode	Change an AutoNav operating mode	Data vectors	2/mon	5 s
Nav_Data_Downlink	Downlink a Nav file	File ID	2/mon	20 s
Nav_Data_Update	Update a Navigation file	File ID	2/mon	20 s
Nav_IPS_Press	Pressurize the main engine	None	1/wk	1 to 30 min
Nav_ACM_Infoturn	Optional desired pointing of the S/C after Nav event	"turnspec"	1/wk	5 s
Nav_BBC_Deadband	Optional desired deadband of the S/C after Nav event	Deadband	1/wk	5 s

*Contingency or emergency back-up command

Table 2.3 lists the mission phases modeled in this analysis and the assumed operating modes of both the science probe and SEP Stage. This operational mode identification combined with the Power Equipment List (PEL) was used by the power system lead in determining the power load requirements.

TABLE 2.3.—MISSION PHASE AND OPERATING MODES

Phase	Max. phase duration (hr)	Probe mode	SEP Stage mode
Launch pad	480	Standby	Standby
Launch from Earth	1	Standby	Standby
Loiter before escape injection	1	Standby	Standby
Escape Injection	0.5	Standby	Standby
Separation from Centaur, cancel tip-off rates	0.5	ACS	Standby
Checkout	24	ACS	Coast
Phasing orbit before Earth flyby	16029	---------	---------
Thrusting	---------	Standby	Thrust
Coasting	---------	ACS	Coast
Communication Attitude	---------	ACS	Comm.
Earth flyby targeting phase	168	ACS	Coast
Earth flyby	24	ACS	Coast
Interplanetary cruise to ~2 AU	6168	---------	---------
Thrusting	---------	Standby/ACS	Thrust
Coasting	---------	ACS	Coast
Communication attitude	---------	ACS	Comm.
SEP Stage jettison and decommissioning	2	ACS	Coast
Probe-only mission	47710	Main	Standby

2.5.2   S/C Trajectory and Pointing

The S/C attitude is near-Earth centered inertial (ECI) during the Earth flyby phasing orbit portion of the trajectory. The S/C attitude provides for tangential thrusting after the Earth flyby as shown in Figure 2.5. Arrows plotted along this phase indicate the direction of the thrust vector required to achieve the optimized trajectory.

The trajectory requires the thrust vector to point between +20° and –50° out-of-plane throughout the trajectory to achieve the 2.5° inclination change necessary to enter into orbit about Saturn. Saturn's orbit is inclined 2.49° to the ecliptic plane. Figure 2.6 shows the Local Vertical Local Horizontal (LVLH) pitch and yaw angles for the S/C over the entire course of the powered trajectory.

The assumptions that went into the modeling of the thruster power profile throughout the trajectory are as follows:

- Up to 10 kWe of excess power at ~1 AU
- 2 percent per year degradation of arrays included in the modeling
- Between 350 and 750 We were available for non-propulsion S/C functions throughout the SEP portion of the trajectory
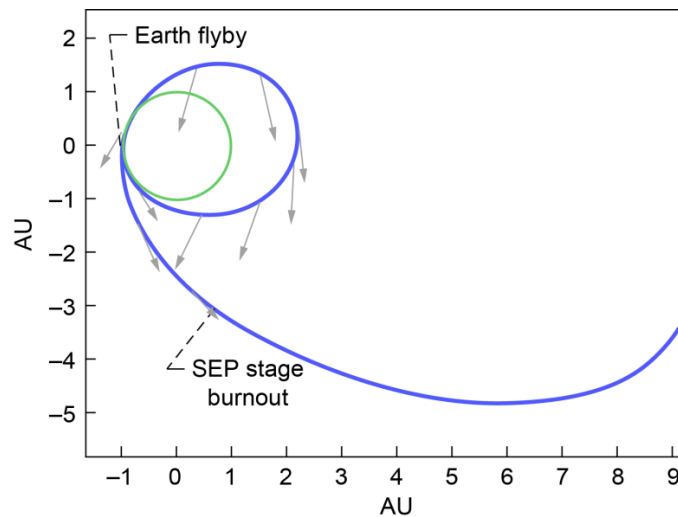
Figure 2.5.—Tangential thrusting during SEP operation.
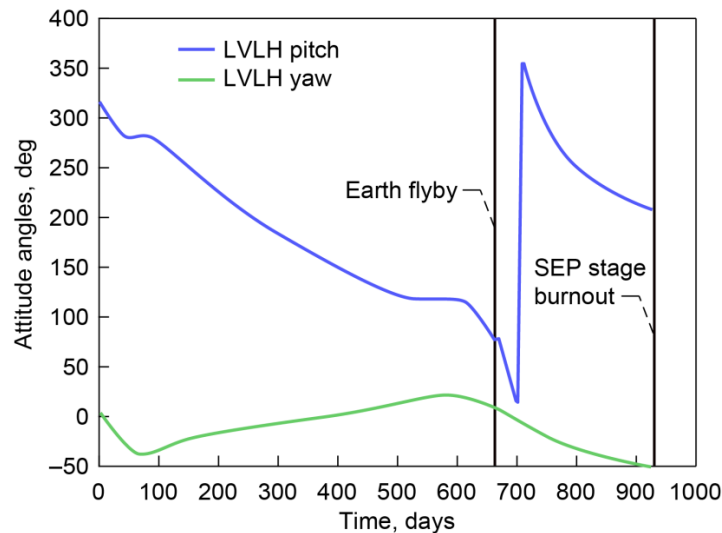
Figure 2.6.—Attitude angle over mission time.

Figure 2.7 shows the power profile and distance from the Sun as a function of days into the mission based on the above assumptions. Figure 2.8 shows gimbal angle over mission time.
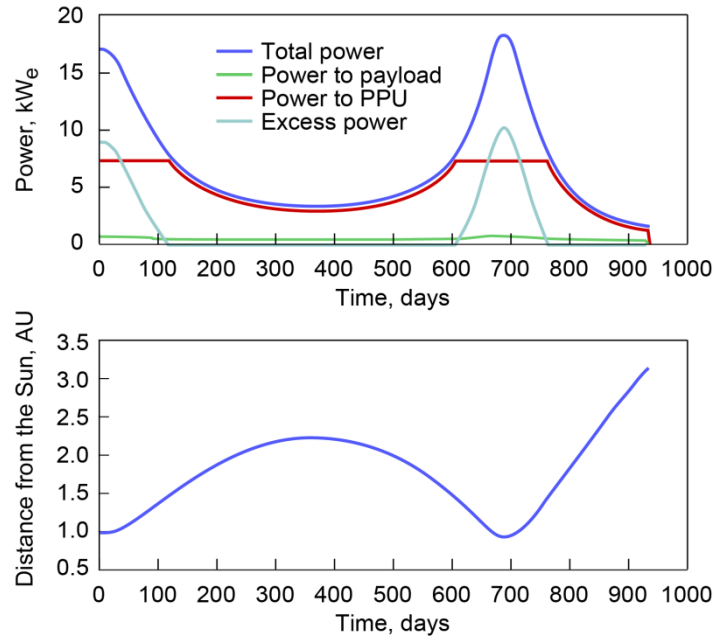
Figure 2.7.—Power profile and distance to the Sun over mission time.
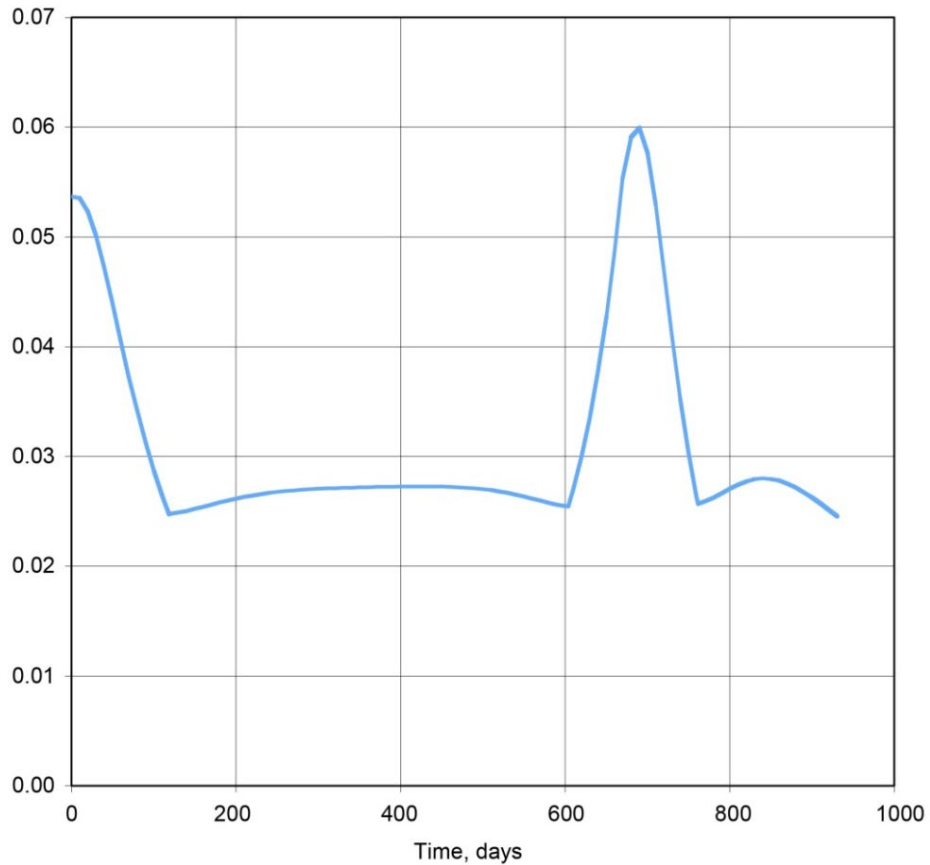
Figure 2.8.—Gimbal angle over mission time.

2.6   Solar Radiation Pressure

Solar radiation pressure is a minor disturbance since the entire trajectory was outside of 1 AU, but was analyzed to ensure the EPS can provide sufficient control to the S/C. Solar torques can easily be counteracted by slightly angling the thruster off of the nominal profile. Alternatively, the solar radiation pressure torque could also be offset with momentum wheels on the probe that are despun with the ion thrusters. The total penalty is less than 0.1 percent equivalent efficiency decrease of the thruster.

2.7   Slew Performance

Slew performance was estimated for a 180° slew in the pitch axis for main antenna pointing to Earth during coast periods.

Assumptions:

- ±7° deadbands
- 0.4 N thrusters in opposing pairs with 200 s I_{SP}
- Simple Proportional Integral Derivative (PID) control
- Inertia matrix from solid model with assumed probe mass properties
- Moment arms attained from solid model

Given the above assumptions on deadbands, and the section of the trajectory during which antenna pointing is required, 133 slew maneuvers are required. Each maneuver involves a slew in pitch angle and then back to the original position. Assuming that the 180° pitch slew is worst-case for all slews, approximately 0.1 kg of propellant per slew maneuver is needed. Each slew maneuver takes approximately 600 s to complete. This yields a propellant mass total of 26.6 kg for 133 slew maneuvers to 180° and 133 maneuvers back to 0° using the RCS thrusters.

$$M\_slewpropellant = (0.1 \text{ kg/slew} * 133 \text{ slews}) * 2$$

2.8   Launch Vehicle Details: Atlas 551

The launch vehicle used in the baseline mission was the Atlas 551 ELV, see Figure 2.9 for an Atlas 551 ELV chosen as the baseline launch vehicle for this study.

Figure 2.9.—Atlas 551 ELV.

PAYLOAD FAIRING (PLF)

Features	5-m Short	5-m Medium	5-m Long
Diameter:	5.4 m	5.4 m	5.4 m
Length:	20.7 m	23.4 m	26.5 m
Mass:	3,540 kg	4,019 kg	4,394 kg

Subsystems
Fairing: Bisector, Sandwich Construction with Graphite Epoxy Face Sheets and an Aluminum Honeycomb
Boattail: Fixed, Composite Sandwich Construction
Separation: Vertical Separation by a Linear Piston and Cylinder Activated by a Pyrotechnic Cord; Horizontal Separation by an Expanding Tube Shearing a Notched Frame, Activated by a Pyrotechnic Cord

CENTAUR

Features
Size: 3.05-m Dia x 12.68-m Length with Extended Nozzle
Inert Mass: 2,138 kg
Propellant: 20,830-kg LH_2 & LO_2
Guidance: Inertial

Subsystems
Structure: Pressure Stabilized Stainless Steel Tanks Separated by Common Ellipsoidal Bulkhead
Propulsion: One or Two Pratt & Whitney Restartable Engine(s)
— Model: RL10A-4-2
— Thrust: 99.2 kN (SEC)   198.4 kN (DEC)
— I_{sp}: 450.5 s
(SEC) One Electromechanically Actuated 51-cm Columbium Fixed Nozzle
Four 27-N Hydrazine Thrusters
Eight 40-N Lateral Hydrazine Thrusters
(DEC) Two Hydraulically Actuated 51-cm Columbium Extendible Nozzles
Eight 40-N Hydrazine Thrusters
Four 27-N Hydrazine Thrusters
Pneumatics: Common with Atlas V 400 Series
Avionics: Common with Atlas V 400 Series
Insulation: Polyvinyl Chloride Foam (1.6-cm Thick), Modified Adhesive Bonding with Optional Radiation Shields

SOLID ROCKET BOOSTERS (SRB)
Zero-to-Three Ground-Lit
Size: 155-cm Dia x 19.5-m Length
Mass: 46,559 kg (Each Fueled)
Thrust: 1,361 kN (Each)
I_{sp}:   275 s
Nozzle Cant: 3 deg

CENTAUR INTERSTAGE ADAPTER (C-ISA LARGE)

Features
Size: 3.81-m Dia x 4.46-m Length
Mass: 2,292 kg (Includes ISA, Aft Stub Adapter and Boattail)

Subsystems
Structure: Composite Sandwich (Aluminum Core/ Graphite Epoxy Face Sheets)

CCB CYLINDRICAL INTERSTAGE ADAPTER

Features
Size: 3.81-m Dia x 0.32-m Length
Mass: 282 kg
Structure: Aluminum Machined Rolled-Ring Forging

COMMON CORE BOOSTER™ (CCB)

Features   Common with Atlas V 400 Series
Size: 3.81-m Dia x 32.46-m Length
Inert Mass: 21,336 kg for 55Z Configuration
Propellant: 284,089-kg LO_2 & RP-1
Guidance: From Upper Stage

Subsystems
Structure: Structurally Stable Aluminum Isogrid Tanks; Integrally Machined Aft Transition Structure; Composite Heat Shield
Separation: 8 Retro Rockets
Propulsion: Pratt & Whitney/NPO Energomash RD-180 Booster Engine (2 Chambers)
SL 100% Thrust = 3,827 kN, I_{sp} = 311.3 s
Vac 100% Thrust = 4,152 kN, I_{sp} = 338.4 s
Pneumatics: Helium for Tank Pressurization, Computer-Controlled Pressurization System
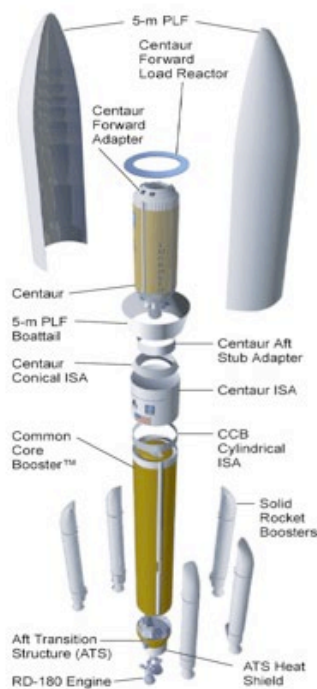Hydraulics: Fluid — Integral with Engine Provides Gimbal Control
Avionics: Flight Control, Flight Termination, Telemetry, Redundant Rate Gyros, Electrical Power

Diagram labels:
5-m PLF; Centaur Forward Load Reactor; Centaur Forward Adapter; Centaur; 5-m PLF Boattail; Centaur Conical ISA; Common Core Booster™; Aft Transition Structure (ATS); RD-180 Engine; Centaur Aft Stub Adapter; Centaur ISA; CCB Cylindrical ISA; Solid Rocket Boosters; ATS Heat Shield

Figure 2.10.—Atlas V ELV.

Figure 2.10 shows the details of the Atlas V from the Lockheed Martin Atlas V fact sheet. These details were used by the configuration designer in packaging the stage into the payload fairing and by the structures engineer for load and launch variables used in structures calculations.

2.9   ELV Performance Calculations

The C_3 range used for the Saturn mission and applied in the low thrust trajectory analysis was in the range of 18.6 to 67.3 km^2/s^2. The mission analysis used a 10 percent ELV contingency factor without an adaptor mass fraction or explicit mass quotation. Figure 2.11 is the ELV performance for the Atlas 551 as modeled in the low thrust trajectory code used in this analysis.

The 5-m short fairing was chosen to maximize payload delivered to Saturn for the given trajectory C_3. Figure 2.12 is the performance of the 5 m fairing Atlas 551 for comparison to Figure 2.11. Note that SEP Trajectories require C_3 of 20 to 30 km^2/s^2.

A D1666/C22 adapter configuration was chosen to provide ample diameter for the SEP Stage. The Atlas V 551 performance to a C_3 of 27 km^2/s^2 is 3864 kg. The following calculations were used in determining the performance of the ELV to Saturn, and in backing out the remaining mass available for the science probe once the SEP Stage had been sized.

- Subtract out ELV margin of 10 percent = 386 kg
- ELV performance after margin (i.e., that which is available to the SEP Stage and Science probe) = 3478 kg
- ELV S/C adaptor = 73 kg taken out of the already subtracted ELV margin.
- ELV margin after adaptor = 314 kg (this reduces ELV margin to 8 percent)
- Subtract off the SEP Stage wet mass including the subsystem and system level growth (contingency) = 1034 kg (see Table 2.4 for details of the wet mass)
- Remaining mass available for the Probe = 2444 kg
- Chose lowest performance over 20 day launch window

Comparison of Atlas 551 Performance Curvefits
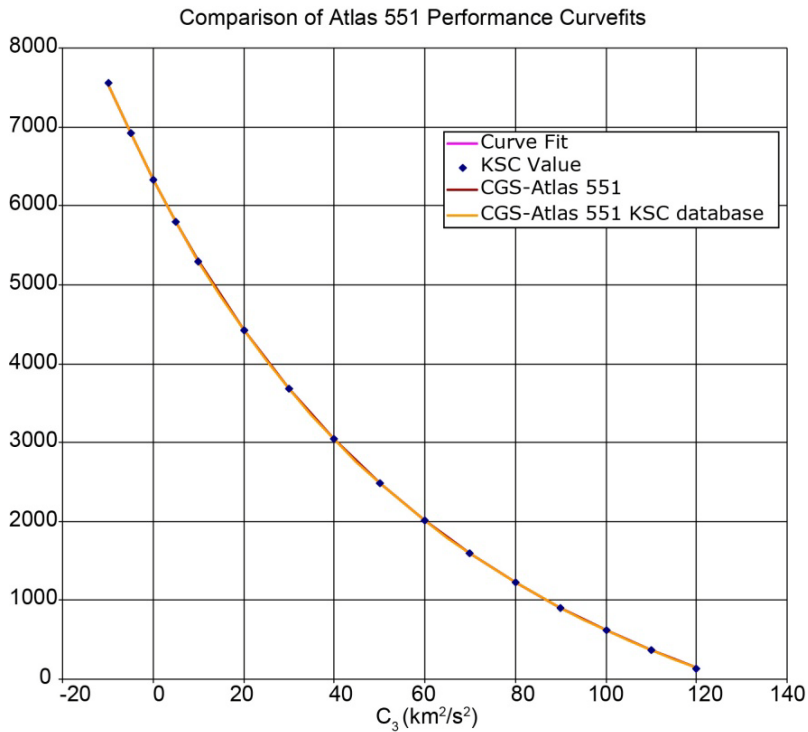
Figure 2.11.—Atlas 551 performance for C_3-internal trajectory modeling.

Launch vehicle: Atlas V 551
Payload fairing: 5-m short
Launch site: CCAFS LC-41
Centaur: 2 burns
PO perigee attitude: 185 km circular
PO coast: 1 hr
To perigee altitude: ≥ 185km
Confidence level: 3-σ GSC
LV includes C22 & B1194 adapters
for Star 48V OIS

551 with Star 48V OIS
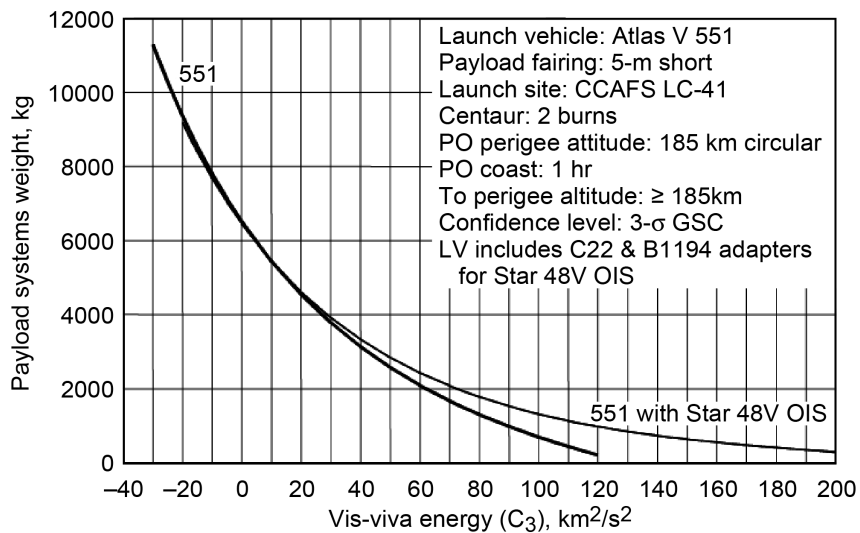
Figure 2.12.—Atlas 551 5 m fairing performance versus C_3 curve.

The performance available for the science probe is calculated at 2444 kg. This is the ELV performance minus the margin, minus the adaptor mass, minus the total wet mass of the SEP Stage. What remains is that which is allocated to the "to be designed" science probe.

TABLE 2.4.—SEP STAGE MASS CALCULATION

Launch masses	Value
Launch vehicle...Atlas V (551)	
ELV performance with margin ....3864.8 kg	
ELV margin (10%) ....386.5 kg	
ELV performance after margin...........................3478.3 kg	
Payload adaptor ....72.6 kg	
ELV margin less adaptor ....313.9 kg	
New net ELV margin...8.1%	
SEP Stage wet mass...1034 kg	
SEP performance to Saturn...................................2444 kg	

2.10 SEP Stage Launch Envelope

The solar arrays and SEP Stage configuration fits within the 5 m Atlas payload fairing envelope as shown in Figure 2.13.

The SEP Stage is designed to keep out of the "keep out zone" in the Atlas 5 m fairing adaptor as shown in Figure 2.14. Figure 2.15 shows the dimensions of the SEP Stage in launch configuration. The probe, as yet not designed, is not shown.
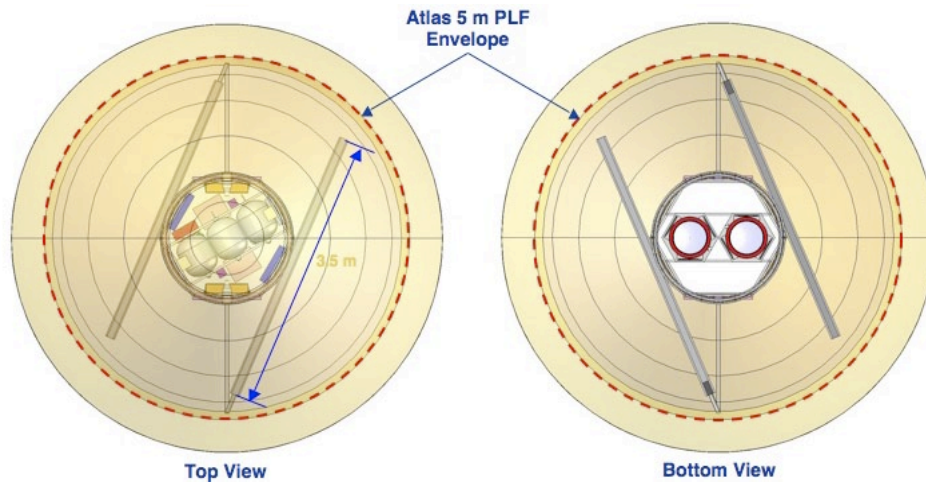
Figure 2.13.—Atlas 551 5 m fairing envelope.
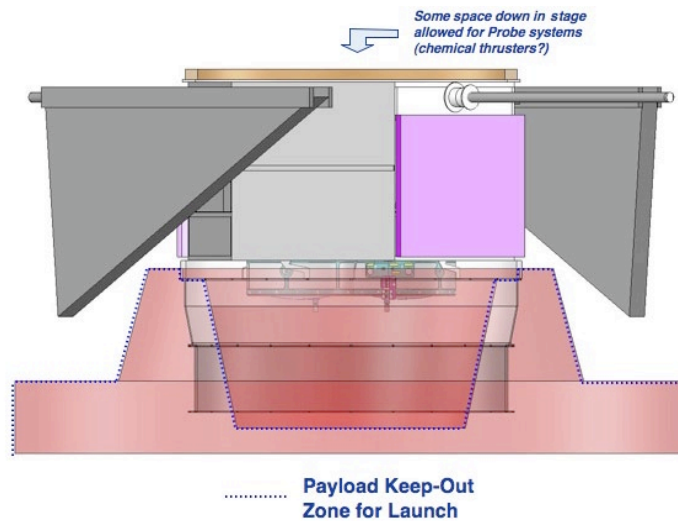
Figure 2.14.—Atlas 551 5 m fairing payload keep out zone.
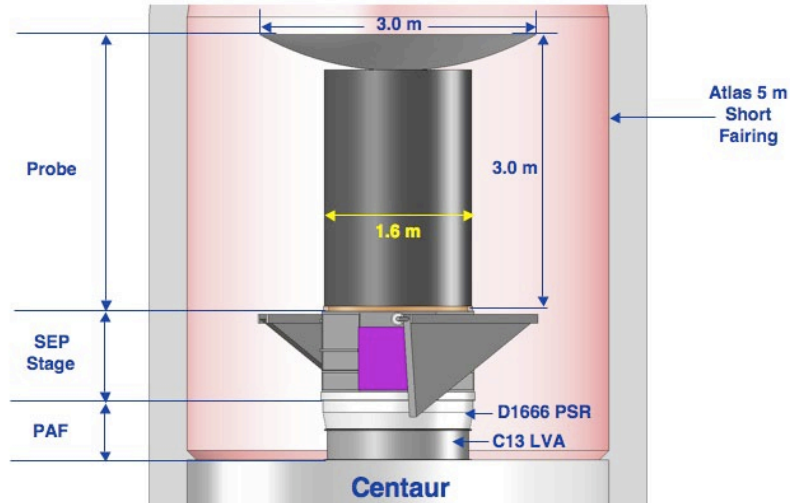
Figure 2.15.—Dimensions of SEP and probe in Atlas V fairing.

2.11   System Design Trade Space

The trade space examined in these designed focused on the number of NEXT thrusters used in the EPS to perform the mission. Cases 2 and 3 were a larger, Flagship class, mission, and were launched on a Delta IV H rather than the Atlas 551. Details on these two trades can be found in Section 7.0 near the end of this report.

Case 1: EP thruster configuration 1+1 (Baseline)

- In this configuration, there is one active thruster and one spare to perform the delta V of the mission.

Case 2: EP thruster configuration 2+1

- In this configuration, there are two active thrusters and one spare to perform the delta V of the mission.

Case 3: EP thruster configuration 3+1

- In this configuration, there are three active thrusters and one spare to perform the delta V of the mission.

Case 3 will be documented in a future COMPASS report of the Saturn Flagship SEP Stage Summary, COMPASS document number CD–2007–09.

2.12   Internal COMPASS Details

COMPASS is a multidisciplinary collaborative engineering team whose primary purpose is to perform integrated vehicle systems analysis and provide trades and designs for both Exploration and Space Science Missions.

2.12.1 GLIDE Study Share

GLobal Integrated Design Environment (GLIDE) is a data collaboration tool that enables secure transfer of data between a virtually unlimited number of sites from anywhere in the world. GLIDE is the primary tool used by the COMPASS design team to pass data real-time between subsystem leads.

2.12.2 GLIDE Architecture

The architecture set up to hold the following design studies is:

Architecture: Saturn_SEP_Stage

2.12.3 GLIDE Study Container(s)

Study name	Summary	GLIDE study container
Study 1	EP thruster configuration 1+1	SaturnSEP_1plus1
Study 2	EP thruster configuration 2+1	SaturnSEP_2plus1
Study 3	EP thruster configuration 3+1	SaturnSEP_3plus1

3.0   Baseline Design

3.1   Baseline System Design

The deployed SEP module (Figure 3.1) and representative S/C are dominated by the large single axis, solar arrays to collect the power to operate a single NEXT thruster throughout the trajectory. Two thrusters are flown with one active and one a cold spare. Gimbals on the thruster provide thrust vector pointing and yaw and pitch control. Radiators are placed on the SEP module face just below the solar array drive assemblies so that they can view deep space during NEXT thruster operation. The science probe was designed as a cylinder probe of unknown design and notional antenna dish in the figure attached to the detailed SEP Stage and solar arrays.

This SEP Stage is capable of delivering a 2444 kg probe on a Saturn trajectory with a hyperbolic arrival velocity of 5.4 km/s. The system assumed a growth of 30 percent mass, 10 percent power, and 6 percent on propellant margins. The SEP Stage relies on the probe for GN&C, C&DH, and Communications. The structure and packaging design of the SEP Stage is configured to carry the science probe and minimize packaging interference with Probe in shroud. All subsystems (except tanks and solar arrays) used a single fault tolerant design assumption approach.
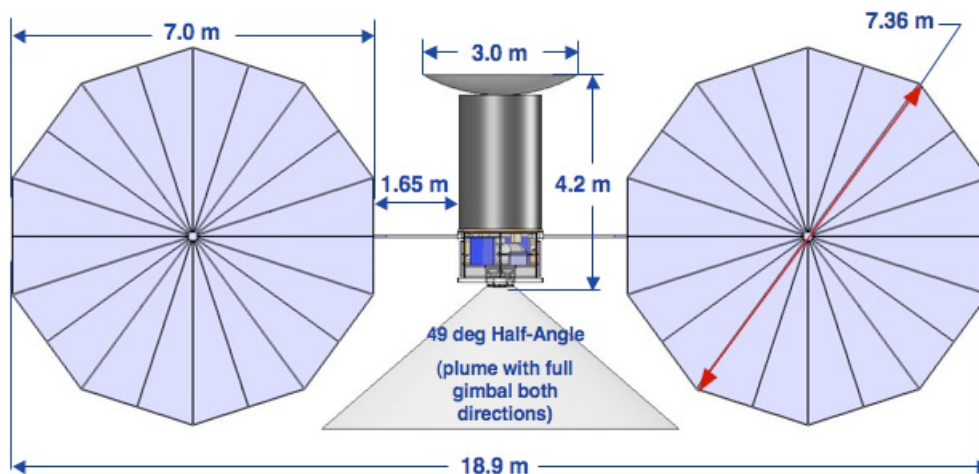
Figure 3.1.—Conceptual SEP Stage deployed configuration.

3.2   Top Level Design (MEL and PEL)

3.2.1   Master Equipment List (MEL)

Where the MEL (Table 3.1) captures the bottoms-up estimation of CBE and growth percentage line item by item from the subsystem designer. Table 3.2, in Section 3.2.2 wraps up those total masses, CBE and total mass after applied growth percentage. In order to meet the total of 30 percent at the system level, an allocation is necessary for system level growth.

TABLE 3.1.—MASTER EQUIPMENT LIST (MEL)

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission				867.5	10	82.9	950.3
01	Saturn SEP S/C	-	-	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0	0.0	0	0.0	0.0
01.2	SEP Stage	-	-	867.5	10	82.9	950.3
01.2.1	Avionics and communications	-	-	10.2	20	2.0	12.2
01.2.2	GN&C	-	-	0.8	50	0.4	1.2
01.2.3	EP subsystem	-	-	267.6	16	44.2	320.8
01.2.4	Thermal control (non-propellant)	-	-	32.7	18	5.9	38.6
01.2.5	Structures and mechanisms	-	-	82.6	18	14.9	97.5
01.2.6	Propulsion	-	-	152.9	10	15.5	168.4
01.2.7	Propellant	-	-	311.7	0	0.0	311.7

3.2.2   Power Equipment List (PEL)

The power requirements for nominal loads of the SEP Stage are shown in Table 3.2. The power growth carried on the analysis was 15 percent. To provide for margin each array sized to 9350 W. The 17000 W total solar arrays (plus margin taking the total to approximately 18.7 kW) account for degradation during the mission (cell, string, diode losses) as well as being oversized to provide sufficient propulsion power for the final thrusting leg up to ~5 AU. The power system's batteries are sized based on the standby power collected from the subsystem requirements in the PEL.

TABLE 3.2.—POWER EQUIPMENT LIST (PEL)

Subsystem	Nominal power (W)	Standby power (W)
C&DH	50	10
GN&C	10	0
Electrical Power	20	5
Thermal Control	0	3
Structures and Mechanisms	10	0
Propulsion	7080	0
Total SEP Stage	7170	18

3.3   System Level Summary

The Total Mass Growth for this study is set to 30 percent (subsystem growth 14.9 percent, system growth 15.1 percent). Table 3.3 is the system integration table that captures the bottoms-up mass estimates for subsystems CBE (current best estimate) masses of their system level items. For clarification in this study, the term Contingency/Growth was mass that was "flown" to the destination and required propellant to perform the ΔV along the mission. This is performance (or mass) delivered on the ELV that cannot be allocated to the science Probe. Subsystem leads applied growth (contingency) percent per subsystem item. Total dry mass growth (contingency) calculated for the system was 14.9 percent. The desired total growth (contingency) at the

system level was 30 percent as a ground rule of this analysis. Reaching the 30 percent total required an additional 83.9 kg shown in Table 3.3 as "System level growth (contingency)" in the last line of the table. The total SEP Stage mass (with subsystem level growth and system contingency reserves) was 1034.2 kg as shown for the line item "SEP Stage total mass with system growth".

TABLE 3.3.—SYSTEM INTEGRATION SUMMARY

S/C system details Master equipment list—Mass	CBE (kg)	Growth (kg)	Total Mass (kg)	Percent of dry mass
Science Probe	N/A	N/A	2444	N/A
SEP Stage			-------	N/A
Avionics and Communications	10.2	2.0	12.2	1.9
GN&C	0.8	0.4	1.2	0.2
Electrical Power	276.6	44.2	320.8	50.2
Thermal Control	32.7	5.9	38.6	6.0
Structures	82.6	14.9	97.5	15.3
Propulsion	152.9	15.5	168.4	26.4
Propellant	311.7	0.0	311.7	N/A
SEP Stage total mass without system growth	867.5	82.9	950.3	
SEP Stage dry mass	555.8	82.9	638.7	
SEP Stage inert mass	558.9	N/A	641.8	
SEP Stage total mass with system growth		83.9	1034.2	
System level growth (contingency) tracking				
Dry mass growth (contingency)	14.9%			
Desired total growth (contingency)	30%			
Desired total growth (contingency)	166.7 kg			
Mass system level growth (contingency)	83.9 kg			
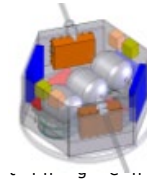
3.4   Design Concept Drawing and Description

Figure 3.2 shows the SEP Stage housing with the major components called out. The thrusters are on the bottom face and not shown in this drawing. The solar arrays are also not shown in this drawing but would be attached at the points labeled "Solar Array Drive".

C&DH
- Digital Control Interface Unit (DCIU) controls power and
- Propulsion system operation commands from probe

Power
- Two Ultra-Flex GaAs Triple Junction Solar Arrays
- Total power at 1 AU 17 kW nominal (18.5 kW with contingency) but run at a maximum power of 7 kW (8.5 kW with contingency), for single string redundancy at 1 AU
- Single axis array drives
- Li-Ion batteries for launch and contingency ops

Propulsion
- 1+1 gimbaled 7 kW NEXT ion propulsion system with
- ~300 kg Xe propellant
- Single thruster operation
- Gimbal provides yaw/pitch control and ability to offset perturbations

Structure
- Al-Li hexagonal, skin-stringer structure supports probe during launch
- At end of SEP Stage operations, a LightBand separation system allows the probe to jettison the stage
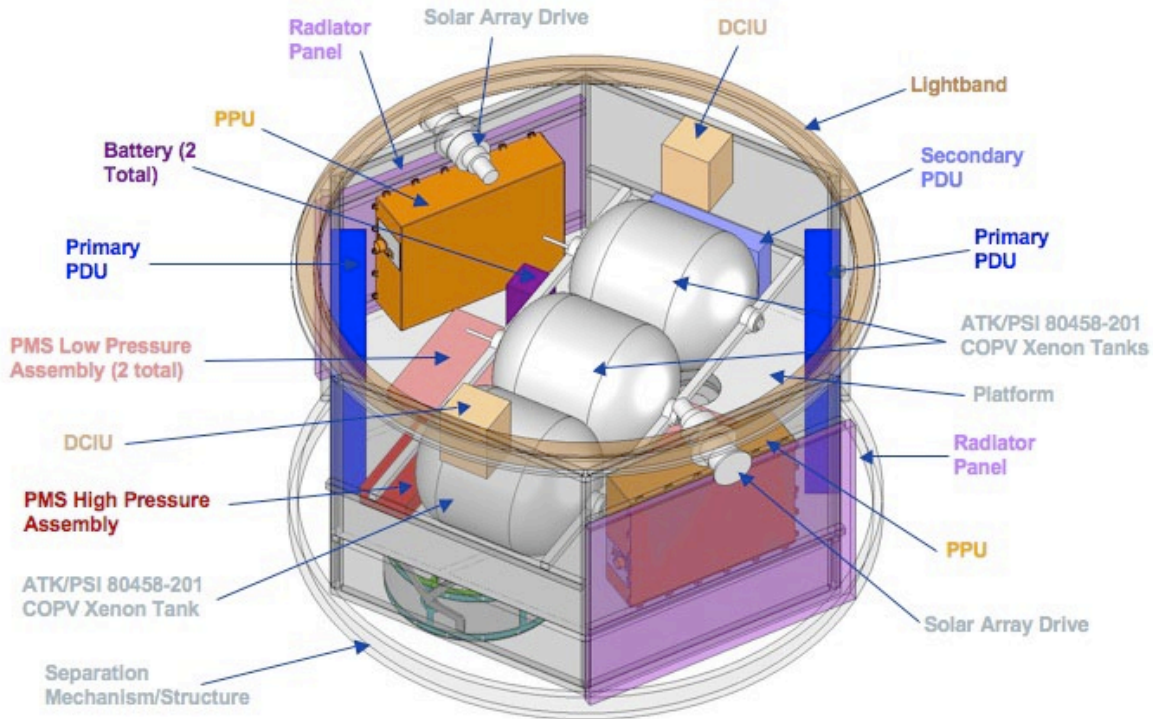
Figure 3.2.—Conceptual SEP Stage Main Components.

Thermal

- Passive heat pipes to 'north and south' radiator panels
- MLI and heaters on Xe tank

4.0   Subsystem Breakdown

4.1   Communications

4.1.1   Communications Requirements

Most of the communications functions will be performed by the systems on the science probe. Therefore, only minimal communications infrastructure is carried on the SEP Stage.

4.1.2   Communications Assumptions

Most of the Communications system was assumed to be part of the science probe. A 3 m dish antenna is shown in the S/C configuration based on assumptions of data link rates and data transfer requirements for the science probe.

4.1.3   Communications Design and MEL

For minimal communications on the stage, two omni-directional antennas were placed on back of propulsion stage. Antennas were included in the design to allow for communications with Earth while thrusters are running. Antennas are assumed to be on the rear or the propulsion unit and interface with the communications hardware in the payload. The SEP Stage was not responsible for probe communicants. The signal is passed to the payload for processing.

4.1.4 Communications Trades

No trades were run on the communications system components since only a minimal set of components were included.

TABLE 4.1.—COMMUNICATIONS SYSTEM MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	---	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	---	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	---	867.5	10	82.9	950.3
01.2.1	Avionics and communications	-	---	10.2	20	2.0	12.2
01.2.1.a	C&DH	-	---	9.5	20	1.9	11.4
01.2.1.b	Communications	-	---	0.7	20	0.1	0.8
01.2.1.b.a	Transponder	0	0.0	0.0	0	0.0	0.0
01.2.1.b.b	RF assembly	0	0.0	0.0	0	0.0	0.0
01.2.1.b.c	Processing module	0	0.0	0.0	0	0.0	0.0
01.2.1.b.d	Antenna	2	0.3	0.6	20	0.1	0.7
01.2.1.b.e	Coaxial cable	2	0.0	0.1	20	0.0	0.1
01.2.1.b.f	Installation—mounting and circuitry	0	0.0	0.0	0	0.0	0.0
01.2.1.b.g	Miscellaneous no. 1	0	0.0	0.0	0	0.0	0.0
01.2.1.b.h	Miscellaneous no. 2	0	0.0	0.0	0	0.0	0.0
01.2.1.b.i	Miscellaneous no. 3	0	0.0	0.0	0	0.0	0.0
01.2.2	GN&C	-	---	0.8	50	0.4	1.2
01.2.3	Electrical Power	-	---	276.6	16	44.2	320.8
01.2.4	Thermal Control	-	---	32.7	18	5.9	38.6
01.2.5	Structures	-	---	82.6	18	14.9	97.5
01.2.6	Propulsion	-	---	152.9	10	15.5	168.4
01.2.7	Propellant	-	---	311.7	0	0.0	311.7

4.1.5 Communications Analytical Methods

Design was based upon hardware currently available from BAE, Maxwell and Aitech which greatly exceeds the capabilities of hardware used in DS-1. Changes to the model are manually made based upon feedback from other subsystem leads.

4.1.6 Communications Risk Inputs

Communications I/O not well defined.

4.1.7 Communications Recommendation

Further details are required from the science probe in order to better understand the requirements on the communications system on the SEP Stage.

4.2 GN&C

4.2.1 GN&C Requirements

The GN&C system must be able to make up for slew maneuvers during the coasting phase of the SEP Stage's trajectory to the Saturn system. All other GN&C systems will reside on the science probe since they will be required after the SEP Stage has been jettisoned.

4.2.2 GN&C Assumptions

- The thruster configuration is for a 1+1 SEP Stage
- The SEP operates for a 1 AU to 3 AU orbit range
- The SEP Stage operates for < 3 yr
- It will take a total of 8 yr for the Probe to reach Saturn
- 3 percent additional Xe is carried for navigation/trajectory and other errors
- 1000 km Earth flyby
- SEP Stage thrust duty cycle 90 percent

4.2.3 GN&C Design and MEL

Ion thrusters can act as pointing control during thrusting for yaw and pitch. Ion thrusters may be able to off-set solar perturbations (during thrusting) and/or dump momentum wheels. Probe provides ACS for all roll control, coasting, and solar array pointing. The probe also commands the ion thruster gimbals. Sun sensors are the only additional ACS hardware component included in the SEP Stage and are used in conjunction with the solar arrays for pointing during maneuvers.

TABLE 4.2.—GUIDANCE, NAVIGATION AND CONTROL SYSTEM MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	---	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	---	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	---	867.5	10	82.9	950.3
01.2.1	Avionics and Communications	-	---	10.2	20	2.0	12.2
01.2.2	GN&C	-	---	0.8	50	0.4	1.2
01.2.2.a	IMU	0	0.0	0.0	0	0.0	0.0
01.2.2.b	Star sensor	0	0.0	0.0	0	0.0	0.0
01.2.2.c	Sun sensor	4	0.2	0.8	50	0.4	1.2
01.2.2.d	ACS	0	0.0	0.0	0	0.0	0.0
01.2.2.e	Flight control electronics	0	0.0	0.0	0	0.0	0.0
01.2.3	Electrical Power	-	---	276.6	16	44.2	320.8
01.2.4	Thermal Control	-	---	32.7	18	5.9	38.6
01.2.5	Structures	-	---	82.6	18	14.9	97.5
01.2.6	Propulsion	-	---	152.9	10	15.5	168.4
01.2.7	Propellant	-	---	311.7	0	0.0	311.7

The slew maneuver RCS propellant is reported in the propulsion system MEL.

4.2.4 GN&C Trades

No trades performed.

4.2.5 GN&C Analytical Methods

GN&C components chosen from off the shelf catalogs. Calculations of the propellant mass required to produce the slew maneuvers is documented in Section 2.6.

4.2.6 GN&C Recommendation

Since the GN&C system has been assumed to reside on the science probe, this push back needs to be iterated with the science probe design team to make sure that the science probe is active during the SEP Stage portion of the trajectory.

4.3 Command and Data Handling (C&DH)

The C&DH system is sometimes referred to as Avionics in documentation. Both terms are equally valid in describing this system.

4.3.1 C&DH Requirements

The SEP Stage C&DH systems were required to provide control interface for all SEP Stage propulsion, power and thermal systems. In this design, the Communications and Navigation functionality were handled by payload.

TABLE 4.3.—C&DH SYSTEM MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	---	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	---	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	---	867.5	10	82.9	950.3
01.2.1	Avionics and communications	-	---	10.2	20	2.0	12.2
01.2.1.a	C&DH	-	---	9.5	20	1.9	11.4
01.2.1.a.a	DCIU	2	4.0	8.0	20	1.6	9.6
01.2.1.a.b	Electrical integration assembly	0	0.0	0.0	0	0.0	0.0
01.2.1.a.c	Data cabling (analog/digital)	1	0.8	0.8	20	0.2	0.9
01.2.1.a.d	Power cabling to DCIU	1	0.8	0.8	20	0.2	0.9
01.2.1.a.e	Remote interface unit	0	0.0	0.0	20	0.0	0.0
01.2.1.a.f	Time generation unit	0	0.0	0.0	0	0.0	0.0
01.2.1.b	Communications	-	---	0.7	20	0.1	0.8
01.2.2	GN&C	-	---	0.8	50	0.4	1.2
01.2.3	Electrical Power	-	---	276.6	16	44.2	320.8
01.2.4	Thermal Control	-	---	32.7	18	5.9	38.6
01.2.5	Structures	-	---	82.6	18	14.9	97.5
01.2.6	Propulsion	-	---	152.9	10	15.5	168.4
01.2.7	Propellant	-	---	311.7	0	0.0	311.7

4.3.2 C&DH Assumptions

Most of the C&DH functions of the combined SEP/Science probe S/C will be performed by the science probe. Only the thruster control will be performed onboard the SEP Stage.

4.3.3 C&DH Design and MEL

The C&DH system design was based on NASA Solar Technology Application Readiness (NSTAR) DS-1 DCIU system, expanded to control power and thermal systems. A DCIU accepts and executes high level commands from the S/C computer and provides propulsion subsystem telemetry to the S/C data system.

The design has been expanded and includes two DCIUs to allow for control of the 1+1 thruster configuration. DCIUs were identical and cross-strapped to each thruster. The DCIUs use the 1553 protocol to communicate to each other and the payload and the RS-422 to communicate with SEP Stage systems. The avionics cabling is assumed to be minimal due to the small number of I/O channels and close proximity of the avionics hardware to their related systems. The block diagram of the C&DH system is shown in Figure 4.1.
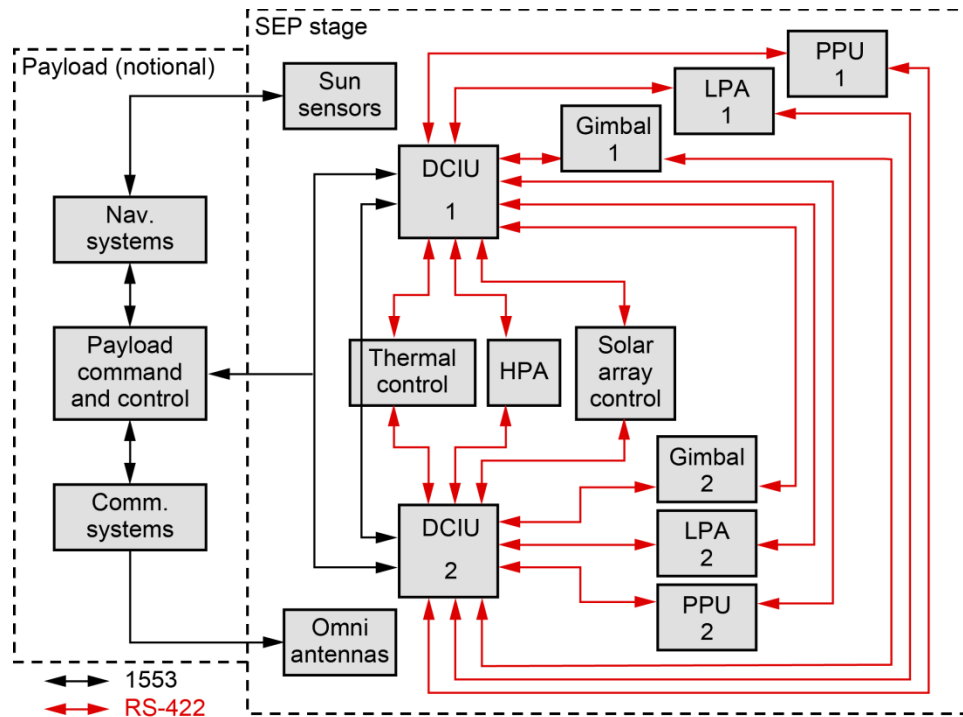
Figure 4.1.—C&DH block diagram.

4.3.4 C&DH Trades

Dedicated propulsion DCIU and remote interface unit versus a single DCIU to run the entire stage.

4.3.5 C&DH Analytical Methods

Design is based upon hardware currently available from BAE, Maxwell, and Aitech which greatly exceeds the capabilities of hardware used in DS-1. Changes to the model are manually made based upon feedback from other subsystems leads. The power estimates based on data from DS-1 avionics and currently available space rated hardware.

4.3.6 C&DH Risk Inputs

- C&DH I/O not well defined.
- C&DH component availability issues.

4.4 Electrical Power System

Figure 4.2 shows the conceptual SEP Stage with arrays deployed.

4.4.1 Power Requirements

Photovoltaic (PV) array power needs to be controllable based on S/C position and EP power demands. PV array must fit inside launch vehicle faring. PV array must not interfere with Saturn Probe (during launch as well as during cruse). PV array must be sized to allow for nominal degradation (.

4.4.2 Power Assumptions

PV array power level requirement for EP portion of S/C was determined by mission analysis. The mission analysis also provides S/C position and attitude information which drives array size and control requirements. Power Management and Distribution (PMAD) system designed for one fault tolerance. Power to EPS stage during launch and array deployment provided by launch vehicle.
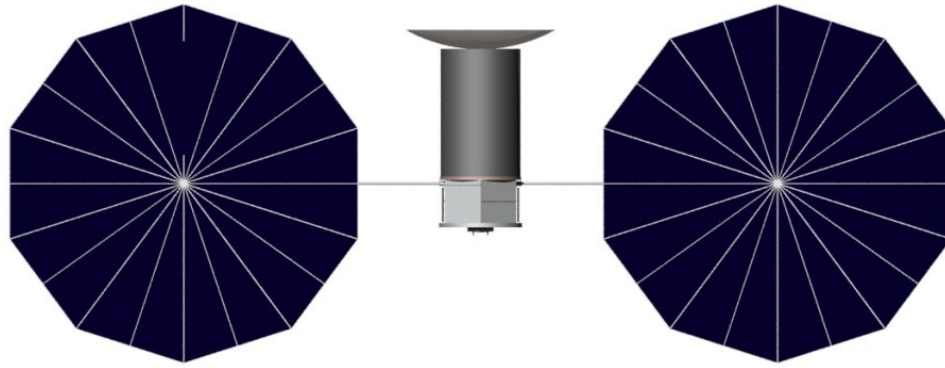
Figure 4.2.—Conceptual SEP Stage with arrays deployed.

4.4.3 Power Design and MEL

In order to provide the required 18.5 kW power (with 200 W housekeeping), each array was oversized by 1.5 kW to address degradation (cell, string, diode losses), two Ultraflex solar arrays design based on ST-8, CEV. Li Ion batteries for pre-array deployment and contingency operations. The power delivered by the solar array was dependant on distance to the sun throughout the mission. Figure 4.5 shows the distance to the sun of the SEP stage over the mission timeline and was used to size the solar arrays.

Figure 4.3 shows an example of an Ultraflex solar array. The schematic of the EPS system designed for this SEP Stage is shown in Figure 4.4.

TABLE 4.4.—POWER SYSTEM MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	-----	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	-----	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	-----	867.5	10	82.9	950.3
01.2.1	Avionics and Communications	-	-----	10.2	20	2.0	12.2
01.2.2	GN&C	-	-----	0.8	50	0.4	1.2
01.2.3	Electrical power subsystem	-	-----	276.6	16	44.2	320.8
01.2.3.a	Solar arrays	-	-----	198.8	15	29.3	228.1
01.2.3.a.a	Solar array mass (cell and structure only)	2	93.7	187.4	15	28.1	215.5
01.2.3.a.b	Solar array gimbals	2	5.7	11.4	10	1.1	12.5
01.2.3.b.	Power management and distribution	-	-----	62.8	16	10.0	72.8
01.2.3.b.a	Power management/control electronics	1	61.2	61.2	15	9.2	70.4
01.2.3.b.b	Power distribution/monitoring wiring harness	2	0.8	1.6	50	0.8	2.4
01.2.3.c	Power cable and harness subsystem	-	-----	7.0	50	3.5	10.6
01.2.3.c.a	S/C bus harness	0	0.0	0.0	0	0.0	0.0
01.2.3.c.b	PMAD harness	0	0.0	0.0	0	0.0	0.0
01.2.3.c.c	EP harness	1	4.0	4.0	50	2.0	6.0
01.2.3.c.d	Solar array to S/C harness	1	3.0	3.0	50	1.5	4.5
01.2.3.c.e	Power cabling	0	0.0	0.0	0	0.0	0.0
01.2.3.d	Battery system	-	-----	8.0	19	1.5	9.5
01.2.3.d.a	Battery assembly-primary	2	3.0	6.0	15	0.9	6.9
01.2.3.d.b	Battery assembly-secondary	2	1.0	2.0	30	0.6	2.6
01.2.4	Thermal Control	-	-----	32.7	18	5.9	38.6
01.2.5	Structures	-	-----	82.6	18	14.9	97.5
01.2.6	Propulsion	-	-----	152.9	10	15.5	168.4
01.2.7	Propellant	-	-----	311.7	0	0.0	311.7

Figure 4.3.—Solar array power versus S/C distance to the sun over the mission timeline.
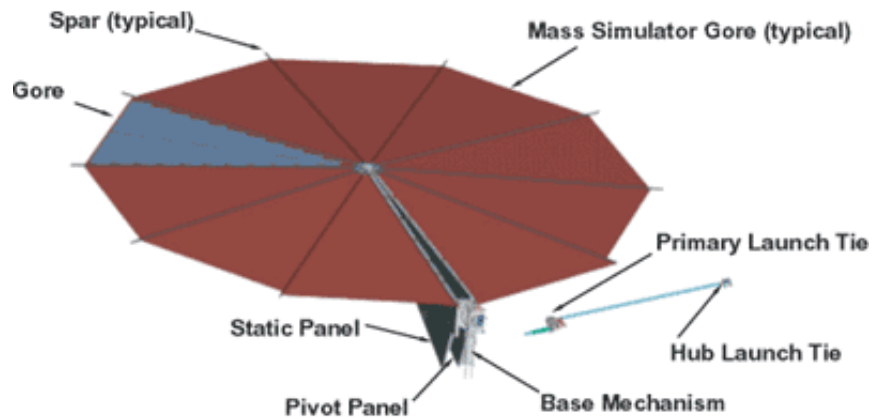
Figure 4.4.—Ultraflex solar array.
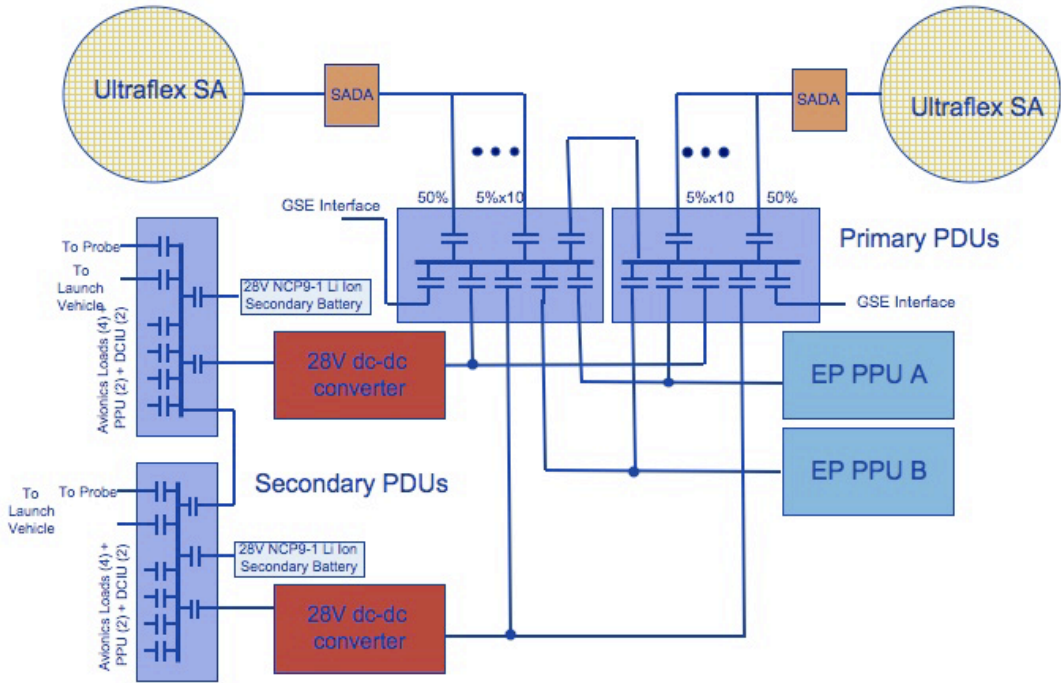
Figure 4.5.—Ultraflex solar array schematic.

Details of the above power system are called out in the bullets below.

Photovoltaic Array

- Two AEC-able Ultraflex arrays at 8500 W (1 AU) each
- Provides 3000 W Contingency at 1 AU (17000 W at 1 AU assumed available for SEP trajectory)
- Each solar array populated with Spectrolab UTJ three-junction solar cells (93.7 kg each)
- PV cells arranged in series strings of 40 cells to provide V_{mp} of 80 V dc at 1 AU (125 V dc at 3 AU)
- Strings arranged in groups to provide 5 to 50 percent of array power at primary PDUs
- Moog SADA (single axis, 5.7 kg each)

PMAD

- Redundant primary PDUs, cross-tied (28.5 kg each)
- Power on buss controlled by string relays. Five percent strings represent 425 W at 1 AU and 47 W at 3 AU
- Redundant 200 W 28 V dc-dc converters feed redundant secondary cross-tied PDUs (5.8 kg total)
- Provisions for 28 V dc feed/receive from launch vehicle and/or Probe

4.4.4 Power Trades

The current baseline design uses probe for pad/contingency power. Solar array power is available to the science probe for transit operations. Option to use solar arrays in lieu of radioisotope power systems.

- Flat plate array versus Ultraflex
  – Ultraflex chosen due to packaging and mass advantages
    ▪ Array power control options (peak power tracker versus sequential shunt vs. sequential switch)
  – Sequential switch chosen for simplicity

4.4.5 Power Comments, Concerns, and Recommendations

Given the cost, availability and safety issues associated with General Purpose Heat Sources (GPHS), only 28 kg available)) and the presence of a large solar array already aboard, does it make sense to examine PV power at Saturn?

4.5 Structures and Mechanisms

4.5.1 Structures and Mechanisms Requirements

The structural design of the SEP Stage must withstand applied loads from launch vehicle and thrusters and provide minimum deflections, sufficient stiffness, and vibration damping. The design will be such that the goal is to minimize weight while meeting the above physical requirements and the structure must fit within the confines of the launch vehicle chosen for this mission.

- Optimized for minimum weight
- Accommodate launch loads (Table 4.5)
  - 5.5g maximum axial load
  - 3.0g maximum lateral load
- Support satellite probe
  - ~2500 kg (Calculated at 2444 kg)
- Support propulsion, navigation, and power systems
- Fit within confines of launch vehicle
- Representative stage/probe separation system
- Solar array and thruster gimbal mechanisms kept in power and propulsion sections, respectively

4.5.2 Structures and Mechanisms Assumptions

The following set of assumptions were applied to the structural design of the SEP Stage.

- Material: Al-Li 2090-T3
- Space frame architecture
  - Riveted and fastened construction
- Satellite probe
  - 2500 kg
  - Center of gravity ~2 m above mounting surface of stage

Atlas 551 loads data was used for structural design. Table 4.5 shows the load limits for the Atlas 551 on the structures.

The main structural will be made of Al and the structural elements will be made of tubular members. Because the mounting area of the science probe to the main structure of the SEP Stage is unknown, assumptions will be made as to the method of connecting the two parts together.

4.5.3 Structures and Mechanisms Design and MEL

The SEP bus space frame architecture was a hex configuration (Figure 4.6 and Figure 4.7). Aluminum sheets were inserted to the design for sheer stiffness and protection of components. Cross members were included to mount components. Rings were incorporated at the top and bottom to match respective separation system.

TABLE 4.5—ATLAS 551 LOAD LIMITS

Load condition	Direction	Atlas V 40Z, 50Z		Atlas V 4YZ, 5YZ	
		Steady state, g	Dynamic, g	Steady state, g	Dynamic, g
Launch	Axial	1.2	±0.5	1.5	±1.5
	Lateral	0.0	±1.0	0.0	±2.0
Flight winds	Axial	1.0 to 2.8	±0.5	1.0 to 2.8	±0.5
	Lateral	±0.4	±1.6	±0.4	±1.6
Strap-on separation	Axial	----	----	3.3	±0.5
	Lateral	----	----	0.0	±0.5
BECO/BETO (Max axial)	Axial	5.5	±0.5	5.5	±0.5
	Lateral	0.0	±0.5	0.0	±0.5
(Max axial)	Axial	3.0 to 0.0	±1.0	3.0 to 0.0	±1.0
	Lateral	0.0	±1.5	0.0	±1.5
SECO	Axial	----	----	----	----
	Lateral	----	----	----	----
MECO/CLE (Max axial)	Axial	4.5 to 0.0*	±1.0	4.5 to 0.0*	±1.0
	Lateral	0.0	±0.3	0.0	±0.3
(Max axial)	Axial	0.0	±2.0	0.0	±2.0
	Lateral	0.0	±0.6	0.0	±0.6

Sign convention
Longitude axis: + (positive) = compression
        – (negative) = tension
        ± may act in either direction					

Lateral and longitudinal loading may act simultaneously during any flight event Loading is applied to the S/C cg					
"Y" in vehicle designator is number of SRBs and ranges from 1 to 3 (400 series) or 1 to 5 (500 series) "Z" in vehicle designator is number of Centaur engines and is 1 or 2					
*Decaying to zero					

The structural design provides support for

- Two ion thrusters and gimbals
  - Two-axis range of motion: ±19°, ±17°
- Three high pressure Xe tanks
- Two low pressure units and one high pressure unit
- Two PPUs and radiators
  - Louvers on the radiators
- Two SADAs
  - Single axis solar array
  - Solar array/boom deployment mechanisms
- Two DCIUs
- Separation unit components
  - The LightBand separation system was used as the initial design point.
- Satellite probe
  - Primary structure, 66.7 kg
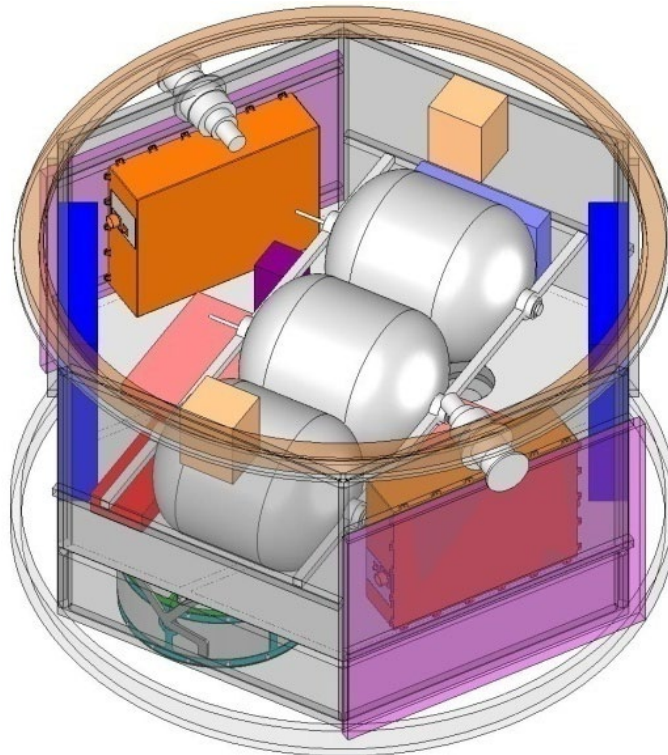 - Secondary structure, total mass 15.9 kg
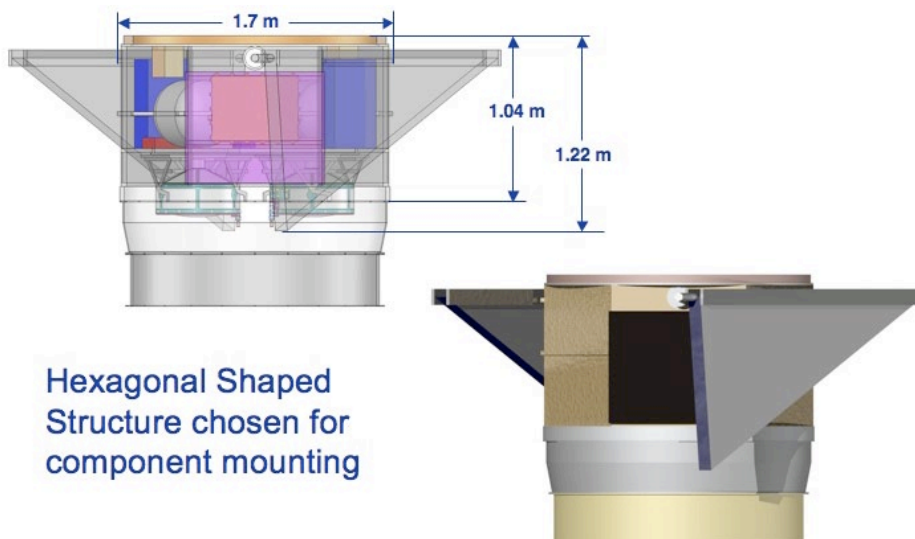
Figure 4.6.—Main S/C bus wire model.

Hexagonal Shaped
Structure chosen for
component mounting

Figure 4.7.—Stage attached to payload fairing adaptor.

TABLE 4.6.—STRUCTURES AND MECHANISMS SYSTEM MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	-----	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	-----	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	-----	867.5	10	82.9	950.3
01.2.1	Avionics and Communications	-	-----	10.2	20	2.0	12.2
01.2.2	GN&C	-	-----	0.8	50	0.4	1.2
01.2.3	Electrical Power Subsystem	-	-----	276.6	16	44.2	320.8
01.2.4	Thermal Control	-	-----	32.7	18	5.9	38.6
01.2.5	Structures and Mechanisms	-	-----	82.6	18	14.9	97.5
01.2.5.a	Structures	-	-----	82.6	18	14.9	97.5
01.2.5.a.a	Primary structures	-	-----	66.7	18	12	78.7
01.2.5.a.a.a	Main bus structure	1	66.7	66.7	18	12	78.7
01.2.5.a.b	Secondary structure (installations)	-	---	15.9	18	2.9	18.8
01.2.5.a.b.a	C&DH installation	2	1.2	2.4	18	0.4	2.8
01.2.5.a.b.b	C&T installation	1	0.0	0.0	0	0.0	0.0
01.2.5.a.b.c	GN&C installation	0	0.0	0.0	0	0.0	0.0
01.2.5.a.b.d	Electric power installation	2	2.6	5.2	18	0.9	6.1
01.2.5.a.b.e	Thermal control installation	0	0.0	0.0	0	0.0	0.0
01.2.5.a.b.f	Electric propulsion installation	1	5.0	5.0	18	0.9	5.9
01.2.5.a.b.g	Tanks support structure	1	3.3	3.3	18	0.6	3.9
01.2.5.b	Mechanisms	-	-----	0.0	0	0.0	0
01.2.6	Propulsion	-	-----	152.9	10	15.5	168.4
01.2.7	Propellant	-	-----	311.7	0	0.0	311.7

4.5.4 Structures and Mechanisms Separation System

An example of the LightBand considered as the separation mechanism between the SEP Stage and probe is shown in Figure 4.7 and Figure 4.8 illustrates the mechanisms of the Standard LightBand (SLB).

Flight Heritage

- December 2006: Three ANDE spheres successfully deployed in LEO on STS-116
- November 2006: Successful deployment of TacSat-2 in LEO
- June 2006: Successful deployment of MITEX in geostationary orbit
- April 2005: Successful deployment of XSS-11 into orbit

LightBand Features and Benefits

- Lightweight: LightBand is typically 1/3rd the mass of an equivalent function V-band/Marman Clamp.
- Flight proven: LightBand has flown on multiple missions and successfully deployed satellites on each.
- Low shock: Non-explosive initiation combined with a low strain energy design produce an order-of-magnitude less shock than pyrotechnics
- Low height: LightBand is half the height of V-band/Marman Clamp devices and 1/8th the cross sectional area allowing larger payload volume.
- Test verified: Every flight LightBand is tested for strength, vibration, thermal vacuum and separation reliability.
- Integrated solution: All of the subsystems of a separation system are completely included in the LightBand.
- Low cost: LightBand typically costs 1/2 as much as an equivalent function V-band/Marman Clamp. LightBand reset cost is typically 1/100th that of V-band/Marman Clamp devices. Typical delivery is 8 months ARO.
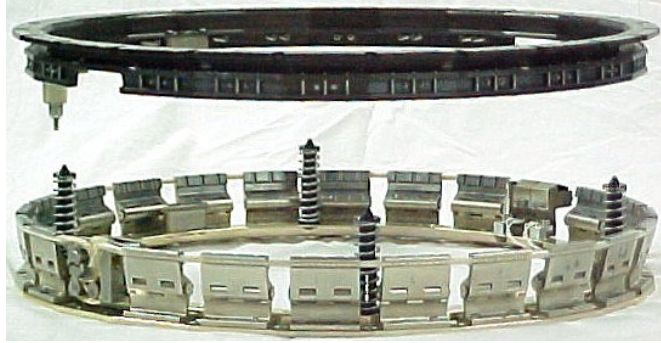
Figure 4.8.—Standard LightBand (SLB).
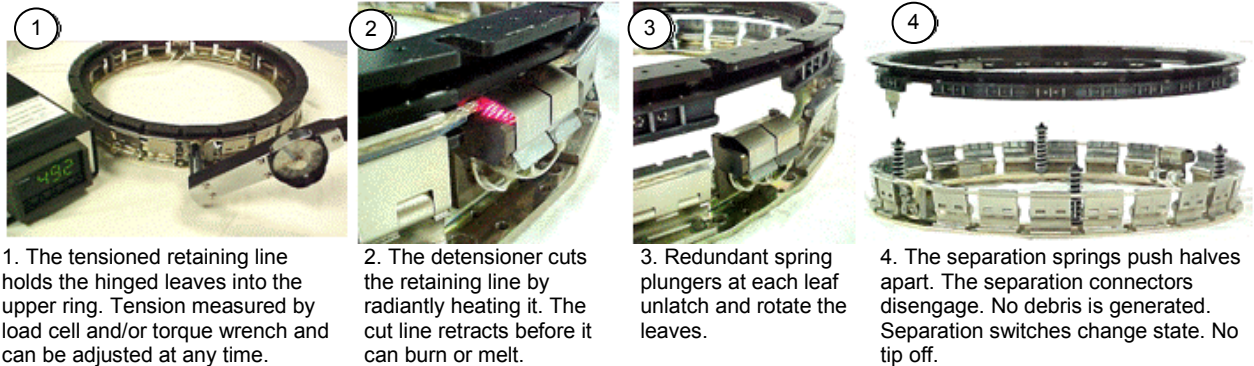
1. The tensioned retaining line holds the hinged leaves into the upper ring. Tension measured by load cell and/or torque wrench and can be adjusted at any time.

2. The detensioner cuts the retaining line by radiantly heating it. The cut line retracts before it can burn or melt.

3. Redundant spring plungers at each leaf unlatch and rotate the leaves.

4. The separation springs push halves apart. The separation connectors disengage. No debris is generated. Separation switches change state. No tip off.

Figure 4.9.—How SLB works.

4.5.5 Structures and Mechanisms Analytical Methods

Preliminary structural analysis with given launch loads.

4.5.6 Structures and Mechanisms Risk Inputs

- Potential impact with foreign object or due to nearby operations.
- Stage interface failures.
- Separation system does not work.
- Potential structural failure
- Solar array deployment failure (mechanisms)

4.5.7 Structures and Mechanisms Recommendation

The following items were suggestions for future cost savings in the structural design.

- Composite sandwich structures may be investigated for further mass reduction but would incur additional costs and may potentially be subject to degradation in space environment

Future detailed analysis is needed to determine

- Detailed mounting methods for components
- Detailed stress analysis
- odal analysis

4.6 Propulsion and Propellant Management

Figure 4.10 shows the propulsion face of the SEP Stage hexagonal core section. This shows the two EP thrusters and their mounting hardware. The SEP Stage in this design operated in a 1+1 configuration, with one operational thruster and one spare for redundancy.
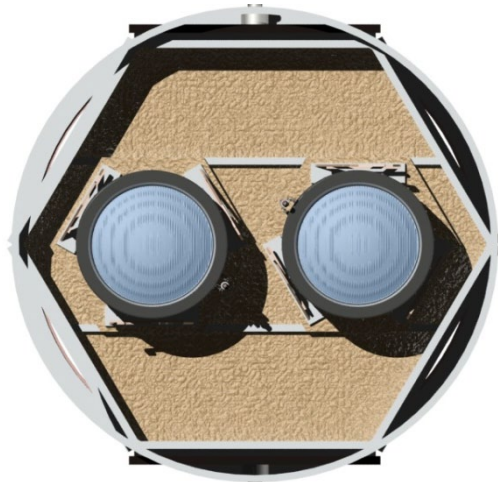
Figure 4.10.—NEXT ion thrusters as shown on bottom of SEP Stage.

4.6.1 Propulsion and Propellant Management Requirements

Because the NEXT Ion thruster was preselected as the thruster for SEP Stage, the primary information required to support the conceptual design of the SEP Stage for this mission consisted of the following:

- Propellant load for mission
- Components masses and sizes
- Power requirements
  - Load; duty cycle
- Configuration requirements
  - Position/proximity

4.6.2 Propulsion and Propellant Management Assumptions

Assumptions driven primarily by need to use mature technologies, including COTS where possible (Table 4.7).

TABLE 4.7.—PROPULSION SYSTEM TRADE SPACE

Assumptions	Action for Study	Response	Options/ Trades	Issues
Subsystem sub-elements technologies preselected	The NEXT thruster, the thruster gimbal, the low and high pressure assemblies that comprise most of the Xe feed system, the PPU for the thruster are baselined for this study	Obtained all mass, sizing, and operational (power) information from in-house experts. This data was compiled into spreadsheet to roll up all of the subsystem characteristic	None for these sub-elements	No sensitivity
Propellant Load determined by mission analysis	Since the required propellant load mass was determined outside of the subsystem sheet, only had to estimate the size of the tanks required to store the Xe	While there is a simple model of sizing high pressure COPV tanks, it was calibrated with PSI tank data for high pressure COPVs. The resulting tanks were very close to the design configuration of an existing tank to ensure maturity	Xe storage density and pressure; No. of tanks	No sensitivity for propellant load; design choices limited by maturity requirement.
Changes in thruster operating points were accounted for in the mission analysis	Because the mission analysis was performed prior to the vehicle study, the thruster performance behavior was not required	The thruster's performance information was limited to throughput which was driver in sizing Xe tanks. Power generation assumed to be sufficient to operate thruster	Thruster performance over mission profile	No sensitivity to thruster performance

4.6.3 Propulsion and Propellant Management Design and MEL

The propulsion subsystem is comprised of (EPS flow diagram is shown in Figure 4.11).

- 1+1 NEXT electrostatic ion thrusters

Figure 4.11.—EPS flow diagram.

- Gimbals on each thruster for thrust vector control
- Three OTS carbon-overwrapped Ti high-pressure storage tanks for the Xe propellant (nominal)
- Xe distribution system to deliver the Xe from the tanks to the thrusters (LPA, HPA, lines)
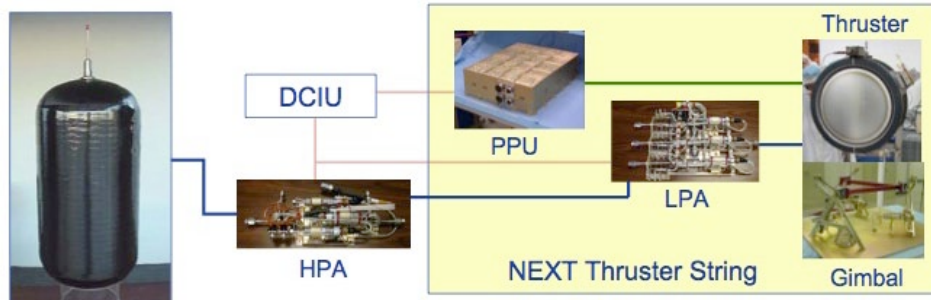- 1+1 PPU for delivering power to each ion thruster

One active 7 kW NEXT Ion engine with one cold spare, two PPUs (one is spare), 1 percent Xe unusable, <2 in. below adapter at launch, 66 cm thruster to thruster center separation, 99 percent of ion plume in 15° half angle cone, OTS tanks.

Table 4.8 lists the MEL for the Propulsion and Propellant management system as well as the propellant for this SEP 1+1 stage design.

TABLE 4.8.—PROPULSION AND PROPELLANT MANAGEMENT SYSTEM AND PROPELLANT MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	-----	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	-----	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	-----	867.5	10	82.9	950.3
01.2.1	Avionics and Communications	-	-----	10.2	20	2.0	12.2
01.2.2	GN&C	-	-----	0.8	50	0.4	1.2
01.2.3	Electrical Power	-	-----	276.6	16	44.2	320.8
01.2.4	Thermal Control	-	-----	32.7	18	5.9	38.6
01.2.5	Structures and Mechanisms	-	-----	82.6	18	14.9	97.5
01.2.6	Propulsion	-	-----	152.9	10	15.5	168.4
01.2.6.a	Primary EP system (thrusters, structure, thermal)	-	-----	42.1	9	3.7	45.8
01.2.6.a.a	Primary EP thrusters	2	12.7	25.4	4	1.0	26.4
01.2.6.a.b	Electric propulsion system structure	-	-----	12.0	18	2.2	14.2
01.2.6.a.b.a	Miscellaneous	1	0.0	0.0	0	0.0	0.0
01.2.6.a.b.b	EP thruster gimbal	2	6.0	12.0	18	2.2	14.2
01.2.6.a.c	EPS thermal control system	-	-----	4.7	12	0.6	5.2
01.2.6.a.c.a	EPS MLI	1	0.6	0.6	12	0.1	0.7
01.2.6.a.c.b	EPS heaters and sensors	1	4.1	4.1	12	0.5	4.5
01.2.6.b	Propellant management	-	-----	35.3	8	2.8	38.1
01.2.6.b.a	Xe propellant tank(s)	3	10.3	30.8	8	2.5	33.2
01.2.6.b.b	Feed system	1	4.5	4.5	8	0.4	4.9
01.2.6.c	PPU	-	-----	75.6	12	8.9	84.5
01.2.6.c.a	PPU mass	2	34.4	68.8	8	5.5	74.3
01.2.6.c.b	Cabling	1	6.8	6.8	50	3.4	10.2
01.2.6.c.c	Miscellaneous	0	0.0	0.0	0	0.0	0.0
01.2.6.d	RCS hardware	-	-----	0.0	0	0.0	0.0
01.2.7	Propellant	-	-----	311.7	0	0.0	311.7
01.2.7.a	Primary EP propellant	-	-----	311.7	0	0.0	311.7
01.2.7.a.a	Primary EP propellant useable	1	308.6	308.6	0	0.0	308.6
01.2.7.a.b	Primary EP propellant residuals (unused)	1	3.1	3.1	0	0.0	3.1
01.2.7.b	RCS propellant (where needed)	-	-----	0.0	0	0.0	0.0
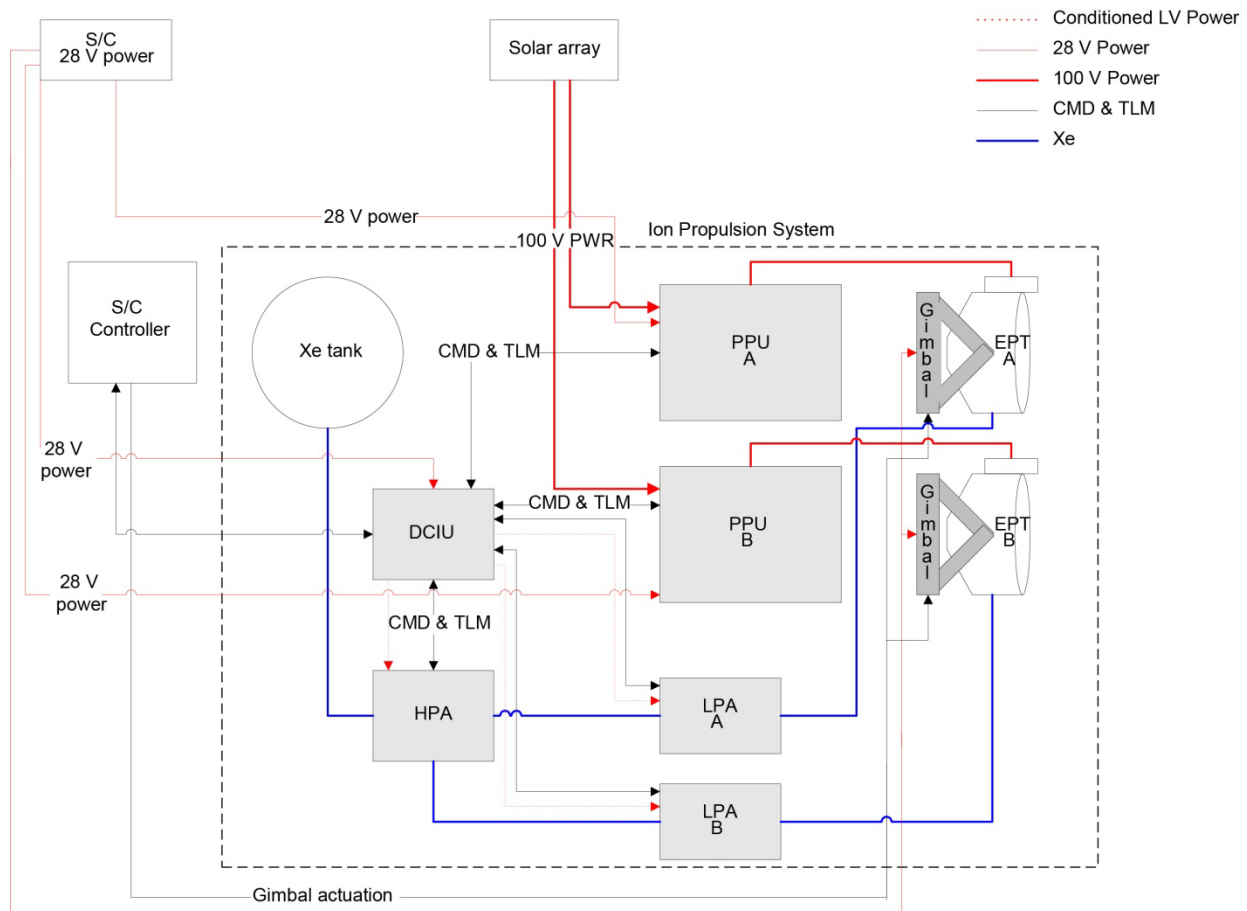
Figure 4.12.—Propellant Management System schematic.

A schematic of the propulsion and propellant management subsystem is shown in Figure 4.12. Derived from the NEXT thruster propulsion system S/C power, controller and DCIU shown for clarity (spare DCIU not shown).

4.6.4 Propulsion and Propellant Management Trades

- OTS versus custom tank design
- Ion engine location (gimbal interference, adapter clearance, contamination from separation system: none considered issues).

4.6.5 Propulsion and Propellant Management Analytical Methods

4.6.5.1 The NEXT Thruster Characteristics

- 0.54 to 6.9 kW thruster input power
- Ring-cusp electron bombardment discharge chamber
- 36 cm beam diameter, two-grid ion optics
- Beam current at 6.9 kW: 3.52 A
- Maximum specific impulse > 4170 s
- Maximum thrust > 236 mN
- Peak efficiency > 70 percent
- Xe throughput > 300 kg, 450 kg qualification level
  – Analysis-based capability >450 kg
- Mass is 12.7 kg (13.5 kg with cable harnesses)
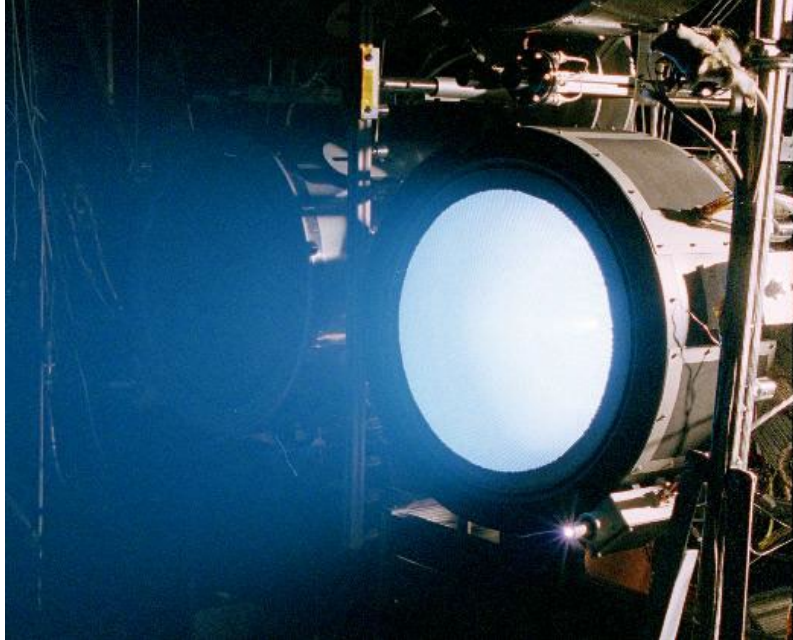
Figure 4.13.—PM 1 Performance Acceptance Test.

Figure 4.13 shows the NEXT thruster in the Prototype Model Thruster (PM1) in Performance Acceptance Test.

4.6.5.2   NEXT Development Status

- PM1 delivered by Aerojet to GRC in January
  – Flight-level design and fabrication processes
- Performance Acceptance Testing successfully completed at GRC
  – Demonstrated throttling range from about 6.9 kW to 540 W, a 12.7:1 range
  – At full power the specific impulse is about 4190 s, producing 236 mN thrust, with an overall thruster efficiency of 71 percent
  – Consistent with performance demonstrated on multiple EM thrusters
- Comprehensive PM1 environmental test sequence completed at JPL
- System integration test planned for FY07

4.6.5.3   Gimbal Overview

Gimbal performance test and CAD illustration are shown in Figure 4.14.

- Breadboard gimbal
  – Designed and fabricated by Swales Aerospace
  – Flight-like design using JPL-approved materials with certifications
- Stepper motors have space-rated option
  – Mass < 6 kg
  – Two-axis range of motion: ±19°, ±17°
- Successful functional testing with PM1 engine
- Gimbal passed two qual-level vibration tests and low-level shock tests with minor issues (fastener backout)
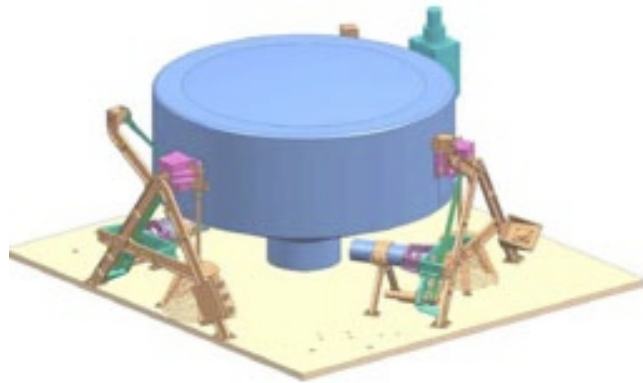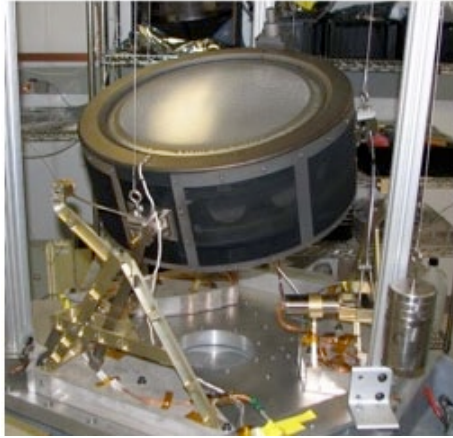
Figure 4.14.—Gimbal performance test and CAD illustration.
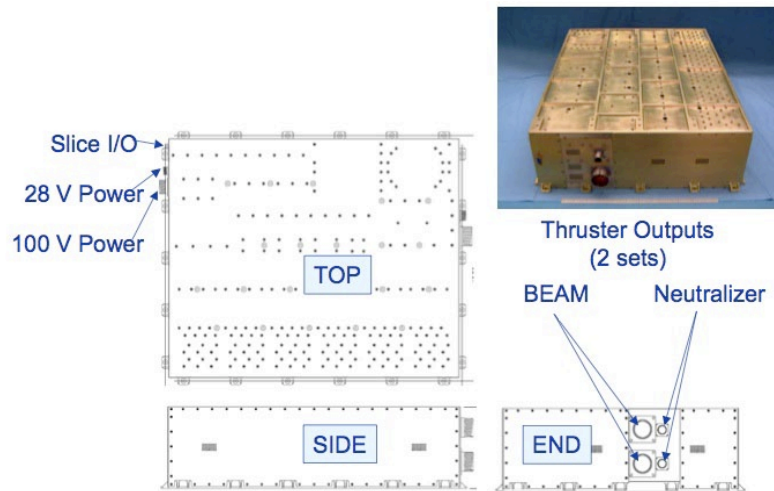
Figure 4.15.—NEXT thruster PPU schematic.

4.6.5.4   NEXT PPU

- EM PPU build is nearing completion at L3 ETI (schematic of NEXT thrust PPU shown in Figure 4.15)
- Modular beam supply and improved packaging provides performance and producability benefits over NSTAR approach
- EM PPU is planned to be delivered to GRC for integration testing, in March 2007—part failure during functional testing and on-going recovery has delayed delivery
- Thorough testing to follow
  - Single-string integration test
    - Functional testing on thruster load
  - Qual-level vibration testing and post-vibe functionals
  - Qual-level thermal/vacuum test
  - EMI/EMC tests
  - Testing planned to be complete in FY2007
- DCIU to be integrated in next development phase
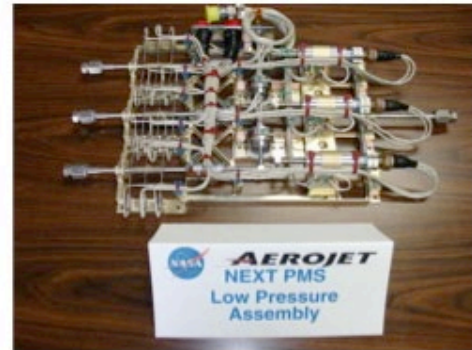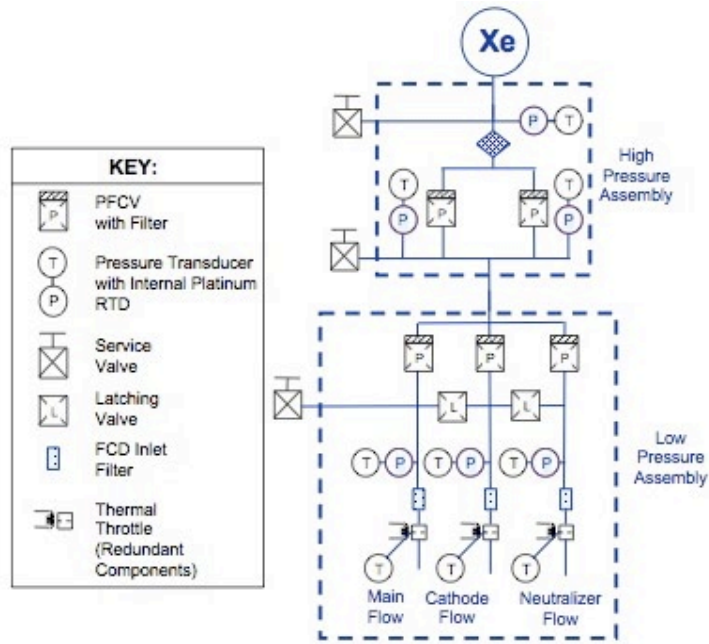
Figure 4.16.—NEXT PMS.

4.6.5.5   NEXT Propellant Management System (PMS)

- All PMS assemblies are complete (Figure 4.16)
    - Two high pressure assemblies (HPA), one flight-like
    - Three low pressure assemblies (LPA), one flight-like
    - Non-flight assemblies are identical except for use of lower cost equivalent parts
- All assemblies have completed functional tests
- Flight-like LPA and HPA successfully completed qual-level vibration testing and post-vibe functionals
- Qual-level thermal/vacuum testing is pending
- PMS will be delivered to GRC this year and tested in single-string integration test

4.6.6   Propulsion and Propellant Management Risk Inputs

- Electric thruster failure to start or operate

4.6.7   Propulsion and Propellant Management Recommendation

Preselection of NEXT thruster system due to technology maturity precludes significant changes in hardware. Changes have been driven by cost instead of mass. Remaining options limited to changes to propellant storage. Reduce number of Xe storage tanks (trade against structural/configuration impacts). Increase Xe storage pressure to reduce storage tank size (trade against stronger tank requirements).

4.7   Thermal Control

4.7.1   Thermal Requirements

To provide spreadsheet based models capable of estimating the mass and power requirements of the various thermal systems. The thermal system must account for heating and cooling issues of the solar arrays, thruster system, and stage electronics during the operating of the SEP Stage from launch at 1 AU from Earth through disposal roughly 3 yr into the trajectory to the Saturn system.

4.7.2   Thermal Assumptions

- Deep Space Operation, worst case at 1 AU

Radiator Sizing

Table 4.9 lists the values of various constants and temperatures used as the radiator sizing assumptions. Note that the Radiator was modeled a thought it always sees deep space with a small (0.05) view factor to the solar array.

TABLE 4.9.—THERMAL SYSTEM
RADIATOR SIZING ASSUMPTIONS

Variable	Value
Radiator solar absorptivity..0.14	
Radiator emissivity 0.84	
Radiator Sun angle ....90°	
Radiator operating temperature ....320 k	
Total radiator dissipation power ....850 W	

Thermal Analysis Propellant Lines and Tanks

The dependence of insulation mass on the number of layers of insulation on the propellant tanks is shown in Figure 4.17. The modeling of the thermal insulation on the tanks and propellant feed lines was done using the assumptions shown in Table 4.10
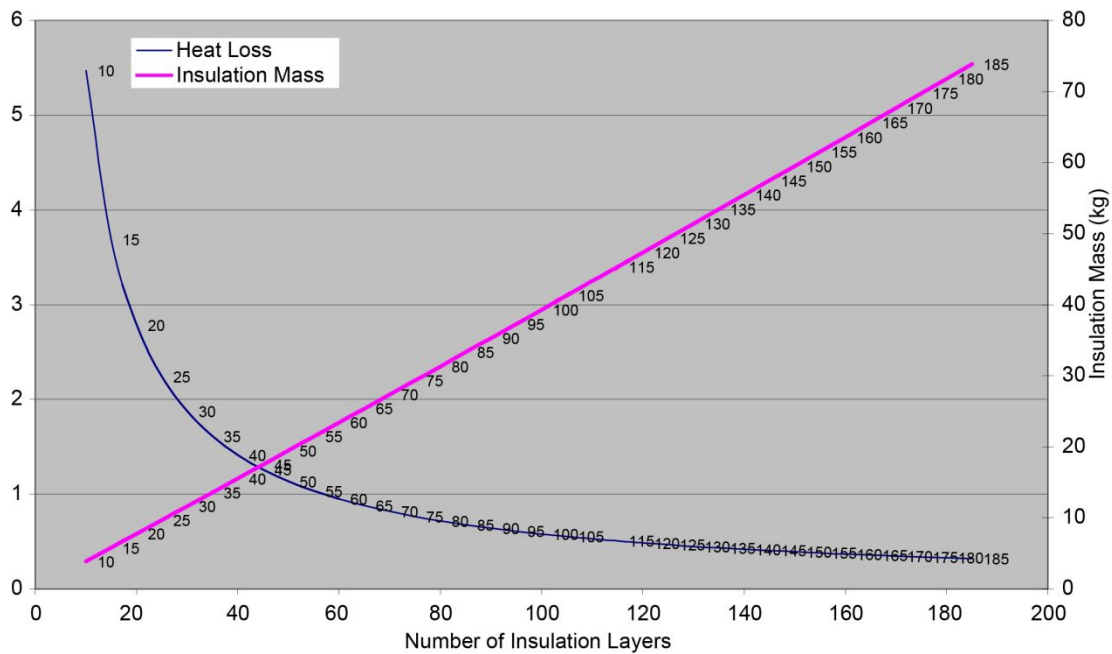
Figure 4.17.—Insulation mass versus number of layers of insulation.

TABLE 4.10.—THERMAL SYSTEM TANK INSULATION SIZING ASSUMPTIONS

Variable	Value
Tank surface emissivity (ε_t)	0.1
MLI emissivity (ε_i)	0.07
MLI material	Al
MLI material density (ρ_i)	2,770 kg/m^3
Internal tank temperature (T_i)	300 K
MLI layer thickness (t_i)	0.025 mm
Number of insulation layers (n_i)	10
MLI layer spacing (d_i)	1.0 mm
Tank immersion heater mass and power level	1.02 kg at up to 1,000 W
S/C inner wall surface emissivity	0.98
S/C outer wall surface emissivity	0.93
Line foam insulation conductivity	0.0027 W/m K
Line foam insulation emissivity	0.07
Propellant line heater specific mass and power	0.143 kg/m at up to 39 W/m
Line foam insulation density	56 kg/m^3

Thermal Analysis—S/C Insulation

The mass of the S/C MLI on the engine bulkhead was modeled to determine the mass of the insulation and the heat loss. The model was based on a first principles analysis of the heat transfer from the S/C through the insulation to space. A near Earth thermal environment was used to size the insulation. Assumptions used are shown in Table 4.11.

TABLE 4.11.—THERMAL SYSTEM TANK
INSULATION SIZING ASSUMPTIONS

Variable	Value
S/C MLI material	Al
S/C MLI material density (ρ_{isc})	2,770 kg/m^3
MLI layer thickness (t_i)	0.025 mm
Number of insulation layers (n_i)	100
MLI layer spacing (d_i)	1.0 mm
S/C radius (rsc)	0.825 m

Thermal Analysis—PMAD Cooling

Thermal control of the electronics and Active Thermal Control System (ATCS) is accomplished through a series of cold plates and heat pipes to transfer the excess heat to the radiators. The model for sizing these components was based on first principles analysis of the area needed to reject the identified heat load to space. From the sizing a series of scaling equations were used to determine the mass of the various system components. Assumptions used are shown in Table 4.12.

TABLE 4.12.—THERMAL SYSTEM PMAD
COOLING SIZING ASSUMPTIONS

Variable	Value
Cooling plate and lines material	Al
Cooling plate and lines material density	2,770 kg/m^3
Number of cooling plates...	2
Cooling plate lengths..	0.5 m
Cooling plate widths...	0.5 m
Cooling plate thickness..	5 mm
Heat pipe specific mass	0.15 kg/m

4.7.3 Thermal Design and MEL

The thermal modeling provides power and mass estimates for the various aspects of the vehicle thermal control system based on a number of inputs related to the vehicle geometry, flight environment and component size.

The thermal control system consists of passive heat-pipes, body mounted radiators (north and south SADA faces) (main loads 500 Wth from PPU, 500 W from PDUs), 100 layers MLI between thruster platform and S/C to prevent heat leakback. Active tank heaters were included in the design to prevent two-phase Xe, louvers on radiators. The radiation level, micrometeoroid environment were other factors considered in the design. See Table 4.13 for the Thermal Protection System MEL and Figure 4.18 for a detailed illustration of the thermal system.

TABLE 4.13.—THERMAL PROTECTION SYSTEM MEL

WBS no.	Description	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
SEP Saturn In-Space Propulsion Mission		-	---	867.5	10	82.9	950.3
01	Saturn SEP S/C	-	---	867.5	10	82.9	950.3
01.1	Saturn Science Probe	0	0.0	0.0	0	0.0	0.0
01.2	SEP Stage	-	---	867.5	10	82.9	950.3
01.2.1	Avionics and Communications	-	---	10.2	20	2.0	12.2
01.2.2	GN&C	-	---	0.8	50	0.4	1.2
01.2.3	Electrical Power	-	---	276.6	16	44.2	320.8
01.2.4	Thermal control (non-propellant)	-	---	32.7	18	5.9	38.6
01.2.4.a	Active thermal control	-	---	1.7	18	0.3	1.9
01.2.4.a.a	Heaters (not tanks)	0	0.0	0.0	18	0.0	0.0
01.2.4.a.b	Thermal control heaters circuit	2	0.2	0.4	18	0.1	0.5
01.2.4.a.c	Data acquisition	1	1.0	1.0	18	0.2	1.2
01.2.4.a.d	Thermocouples	25	0.0	0.3	18	0.0	0.3
01.2.4.b	Passive thermal control	-	---	24.9	18	4.5	29.3
01.2.4.b.a	Heat sinks	2	3.5	6.9	18	1.2	8.2
01.2.4.b.b	Heat pipes	1	0.7	0.7	18	0.1	0.9
01.2.4.b.c	Radiators	1	9.3	9.3	18	1.7	11.0
01.2.4.b.d	Engine bulkhead MLI	1	0.0	0.0	18	0.0	0.0
01.2.4.b.e	Temperatures sensors	50	0.0	0.5	18	0.1	0.6
01.2.4.b.f	Phase change devices	0	0.0	0.0	18	0.0	0.0
01.2.4.b.g	Thermal coatings/paint	1	0.9	0.9	18	0.2	1.1
01.2.4.b.h	Micrometeor shielding	2	0.6	0.9	18	0.2	1.1
01.2.4.b.i	S/C MLI	1	5.5	5.5	18	1.0	6.5
01.2.4.c	Semi-passive thermal control	-	---	6.2	18	1.1	7.3
01.2.4.c.a	Louvers	1	5.4	5.4	18	1.1	6.3
01.2.4.c.b	Thermal switches	4	0.2	0.8	18	0.1	0.9
01.2.5	Structures and Mechanisms	-	---	82.6	18	14.9	97.5
01.2.6	Propulsion	-	---	152.9	10	15.5	168.4
01.2.7	Propellant	-	---	311.7	0	0.0	311.7
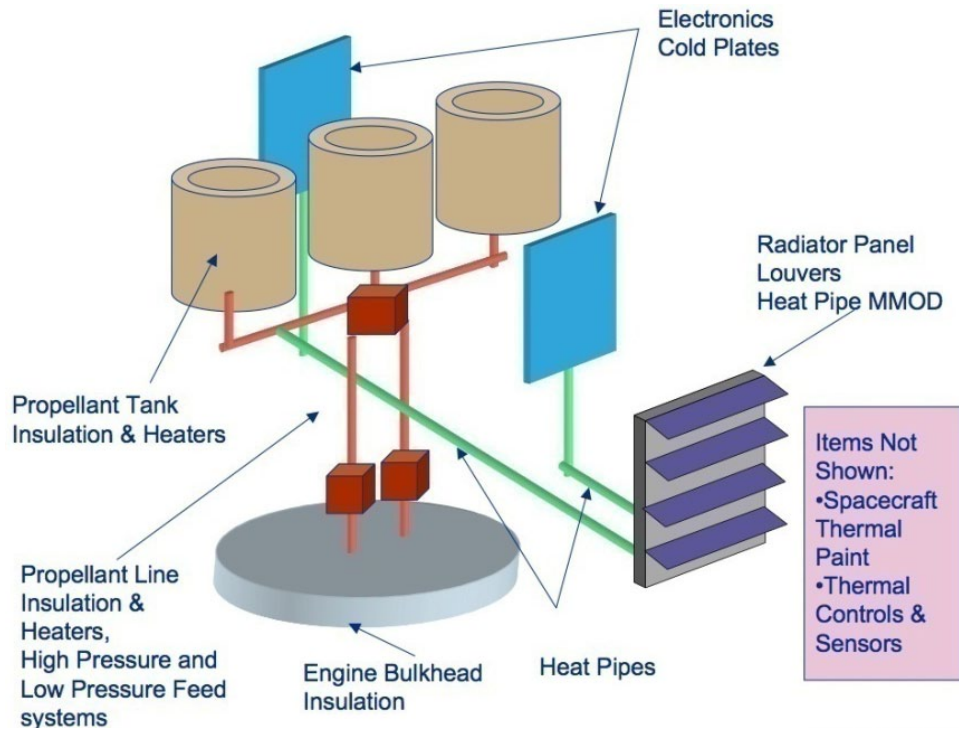
Figure 4.18.—Thermal system.

4.7.4 Thermal Trades

Traded location of radiators and insulation versus heaters.

4.7.5 Thermal Analytical Methods

Table 4.14 details the thermal protection data transfer. Inputs represent data, which the thermal system design required and output represents the data that will be provided to the SEP Stage design from the thermal system.

TABLE 4.14.—THERMAL PROTECTION DATA TRANSFER

Input	Output
S/C dimensions (length, diameter)	Heat pipe length and mass
Power management and electronics dimensions	Cold plate size and mass
Waste heat load to be rejected	Radiator size and mass
Distance from the Sun and S/C orientation	S/C insulation mass and thickness
View factor of the solar arrays and their temperature	Thermal system components mass
Propellant tank dimensions and operating temperature	Propellant tanks insulation mass and heater power level
Propellant line lengths and operating temperature	Propellant line insulation mass and heater power level

Environmental Models

- Solar intensity based on S/C location

Systems Modeled

- Micrometeor shielding on radiator
- Radiator panels
  - The radiator panel area has been model along with a rough estimate of its mass.
  - The model was based on first principles analysis of the area needed to reject the identified heat load to space. From the area a series of scaling equations were used to determine the mass of the radiator
  - 1 AU thermal environment was used to size the radiator
- Thermal control of propellant lines and tanks
  - The model was based on a first principles analysis of the radiative heat transfer from the tanks and propellant lines to space. The heat loss through the insulation set the power requirement for the tank and line heaters
  - This modeling included propellant tank MLI and heaters and propellant line insulation and heaters
  - 1 AU thermal environment was used to calculate the heat loss
- S/C insulation
  - The mass of the S/C MLI on the engine bulkhead was modeled to determine the mass of the insulation and heat loss
- Avionics and PMAD cooling

5.0 Software Cost Estimation

5.1 Design Requirements

- Software to control the following systems
  - Power
    - Solar array control
    - Power distribution unit
    - Solar array deployment and drives
  - Propulsion
    - PPU
    - Low pressure assembly
    - High pressure assembly
    - Thruster gimbals
  - Thermal control

5.2 Assumptions

- Estimates based upon "Space Mission Analysis and Design," Chapter 16
- Payload will handle navigation and communications
- Programming language will be C
- Assembly instructions per SLOC of C = 7
- Software on each DCIU is identical

5.3 Summary

Software on each DCIU is identical. Each DCIU controls one thruster independent of the other. DCIUs communicate with science payload using 1554 protocol. DCIUs communicate using 1554 protocol. DCIUs communicate to propulsion subsystems using RS-422.
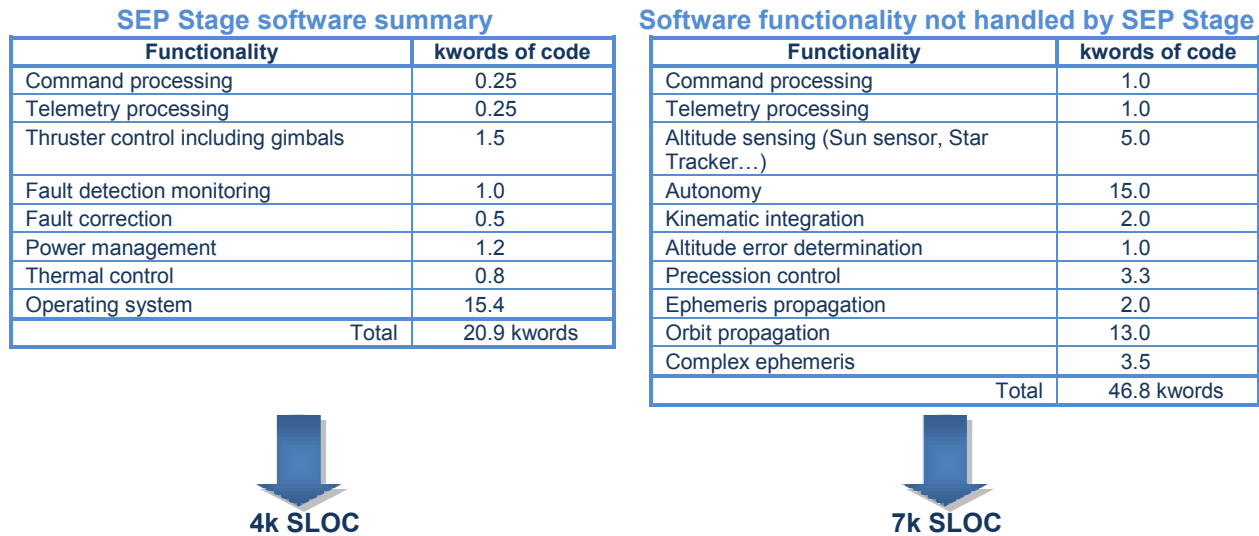
SEP Stage software summary	
Functionality	**kwords of code**
Command processing	0.25
Telemetry processing	0.25
Thruster control including gimbals	1.5
Fault detection monitoring	1.0
Fault correction	0.5
Power management	1.2
Thermal control	0.8
Operating system	15.4
Total	20.9 kwords

Software functionality not handled by SEP Stage	
Functionality	**kwords of code**
Command processing	1.0
Telemetry processing	1.0
Altitude sensing (Sun sensor, Star Tracker…)	5.0
Autonomy	15.0
Kinematic integration	2.0
Altitude error determination	1.0
Precession control	3.3
Ephemeris propagation	2.0
Orbit propagation	13.0
Complex ephemeris	3.5
Total	46.8 kwords

4k SLOC   **7k SLOC**

Figure 5.1.—SEP Stage software sizing.

6.0 Cost, Risk and Reliability

6.1 Costing

6.1.1 Costing Ground Rules and Assumptions

Estimates are for Phase C/D prime contractor costs without fee for the SEP Stage. Estimates developed at the subsystem and component levels using mostly mass-based parametric estimates. Prototype development approach assumed. Quantitative risk analysis performed using Monte Carlo simulation based on mass and Cost Estimation Relationship (CER) uncertainties. Estimates reflect the mean (which is roughly the 64th percentile for the total) unless otherwise indicated.

6.1.2 Cost Summary 1+1 SEP Stage

Table 6.1 shows the cost summary for the SEP Stage.

TABLE 6.1.—COST SUMMARY 1+1 SEP STAGE

| WBS no.		DDT&E	Flight hardware	Ground spares	Total (first flight)
01.2.1	Avionics and communications	4.3	3.6	2.0	9.9
01.2.2	GN&C	0.0	0.2	0.0	0.2
01.2.3	Electrical power subsystem	21.6	20.2	4.1	45.9
01.2.4	Thermal control (non-propellant)	5.2	0.6	0.4	6.2
01.2.5	Structures and mechanisms	9.5	3.1	0.0	12.5
01.2.6	Propulsion	15.3	9.8	5.3	30.4
	Subsystem subtotal	56.0	37.4	11.8	105.1
	Systems integration				
	IACO and STO	7.6	1.7		9.3
	GSE	2.5			2.5
	SE&I	3.1	4.4		7.5
	PM	0.8	1.2		2.0
	LOOS	3.7			3.7
	Systems integration subtotal	17.6	7.3		24.9
	Total prime contractor—Mean	73.6	44.7	11.8	130.1
	Total prime contractor—10 percentile	42.8	31.5	7.3	81.6
	Total prime contractor—90 percentile	115.9	60.2	17.1	193.2
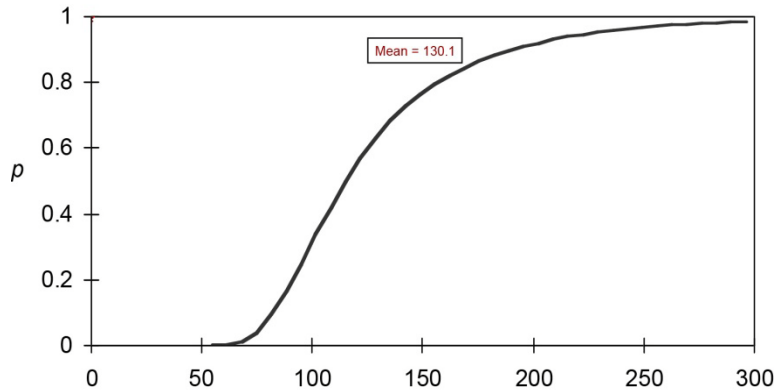
Figure 6.1.—Estimate distribution 1+1 SEP Stage.

6.2 Risk Analysis and Reduction

6.2.1 Assumptions

Each subsystem lead developed a risk for his/her subsystem. Risk attributes are based on the CEV risk values. A risk list is not based on trends of criticality. Some mitigation plans are offered as suggestions.

6.2.2 Risk Summary

These risks, with proper pro-active planning can be mitigated early to avoid becoming problems late in the development life cycle or during the extended mission.

Areas of concern for this mission include

- Stage interface problems
- Separation system issues
- Avionics components not available as well as undefined
- Foreign object debris
- Untested solar array hardware
- Design of the probe S/C
- Possible Electric Thruster limitations

6.3 Reliability

Reliability analysis was not performed on this study. Future iterations of the SEP Stage design will include reliability.

No.	L x C	Team	Risk Title
1	5 x 4	Avionics	Avionics I/O Not Well Defined
2	4 x 4	Avionics	Avionics Components Availability
3	3 x 4	SE&I	Stage Interface Failure
4	3 x 4	Thruster	Electric Thruster Failure to Start/Operate
5	2 x 4	GNC	Navigation Sensor View Blockage
6	2 x 4	Launch Vehicle	Separation System does not work
7	2 x 4	Power	Solar Array Deployment Failure
8	2 x 4	Structures	Potential Structural Failure

Criticality
- High
- Med
- Low

7.0   Trade Space Iterations

Trades were run on the number of active thrusters used to perform the DV of the trajectory. Cases 2 and 3 were classified as Flagship class missions and will be documented in more detail in a future COMPASS final report on the Saturn Flagship SEP Stage (Table 7.1 to Table 7.4).

TABLE 7.1.—SEP STAGE COMPARISON
[SEP Stage comparison details (masses in kg)]

	Flagship mission			
	1+1	2+1	3+1	Delta Case 2 to Case 3
Total SEP Stage dry mass	638.7	1036.1	1160.6	124.5
Total SEP Stage CBE dry mass	555.8	894.1	1003.0	108.9
Bottoms-up growth (lbm)	82.9	142.0	157.6	15.6
Bottoms-up growth (%)	0.15	0.16	0.16	0.00
Required system level growth (30%)	0.15	0.14	0.14	0.00
Additional system level growth	83.9	126.2	143.3	17.1
Total SEP Stage with 30% system growth	722.5	1162.3	1303.9	141.6
Total propellant	311.7	599.9	611.1	11.1
Total wet mass (with 30% growth factor)	1034.2	1762.3	1915.0	152.7
Probe mass	2444.0	3975.0	3946.7	−28.3

TABLE 7.2.—SEP STAGE BOTTOMS-UP MEL COMPARISON

	SEP Stage bottoms-up CBE mass (kg)		
SEP Stage	1+1	2+1	3+1
C&DH	9.5	11.4	13.2
Communications	0.7	0.7	0.7
GN&C	0.8	0.8	0.8
EP subsystem	276.6	495.8	500.2
Thermal control (non-propellant)	32.7	42.9	56.8
Structures and mechanisms	82.6	90.8	117.3
Propulsion	152.9	251.7	313.9
Propellant	311.7	599.9	611.1
Total SEP Stage CBE wet mass	867.5	1494.0	1614.1
Total SEP Stage CBE dry mass	555.8	894.1	1003.0

TABLE 7.3.—SEP STAGE TOTAL MASS COMPARISON

	SEP Stage bottoms-up total mass (kg)		
SEP Stage	1+1	2+1	3+1
C&DH	11.4	13.7	15.8
Communications	0.8	0.8	0.8
GN&C	1.2	1.2	1.2
EP subsystem	320.8	586.0	592.6
Thermal control (non-propellant)	38.6	50.6	67.1
Structures and mechanisms	97.5	107.1	137.7
Propulsion	168.4	276.7	345.4
Propellant	311.7	599.9	611.1
Total SEP Stage CBE wet mass	950.3	1636.0	1771.6
Total SEP Stage CBE dry mass	638.7	1036.1	1160.6

TABLE 7.4.—FLAGSHIP MISSION (CASE 3) COMPARED TO BASELINE (CASE 1)

	SEP Stage Comparison Details (masses in kg)		
	1+1	3+1	Flagship
Total SEP Stage dry mass	638.7	1160.6	521.9
Total SEP Stage CBE dry mass	555.8	1003.0	447.2
Bottoms-up growth (lbm)	82.9	157.6	74.7
Bottoms-up growth (%)	0.15	0.16	0.01
Required system level growth (30%)	0.15	0.14	-0.01
Additional system level growth	83.9	143.3	59.5
Total SEP Stage with 30% system growth	722.5	1303.9	581.4
Total propellant	311.7	611.1	299.4
Total wet mass (with 30% growth factor)	1034.2	1915.0	880.8
Probe mass	2444.0	3946.7	1502.7

7.1   Trade 1 : 1+1

The baseline case documented in this report is the first trade using one active NEXT thruster and one spare.

7.2   Trade 2: 2+1

Details about the second trade are shown in Table 7.5.

TABLE 7.5.—CASE 2 SYSTEM SUMMARY

S/C system details Master equipment list—Mass	CBE (kg)	Growth (kg)	Total Mass (kg)	Percent of dry mass
Science Probe	N/A	N/A	3975	N/A
SEP Stage				N/A
Avionics and Communications	12.1	2.4	14.5	1.4
GN&C	0.8	0.4	1.2	0.1
Electrical Power	495.8	90.2	586.0	56.6
Thermal Control	42.9	7.7	50.6	4.9
Structures	90.8	16.3	107.1	10.3
Propulsion	251.7	25.0	276.7	26.7
Propellant	599.9	0.0	599.9	N/A
SEP Stage total mass without system growth	1494.0	142.0	1636.0	
SEP Stage dry mass	894.1	142.0	1036.1	
SEP Stage inert mass	900.0	N/A	1042.0	
SEP Stage total mass with system growth		126.2	1762.3	
System level growth (contingency) tracking				
Dry mass growth (contingency)		15.9%		
Desired total growth (contingency)		30%		
Desired total growth (contingency)		268.2 kg		
Mass system level growth (contingency)		126.2 kg		

Table 7.6 summarizes the launch vehicle performance hits that lead to the SEP performance to Saturn value. Note that the total SEP Stage wet mass with growth from Table 7.5 (1762) is subtracted from the ELV performance after margin value (5737 kg) in order to arrive at the SEP performance number of 3875 kg. This number is the total mass available to the science probe.

TABLE 7.6.—CASE 2 MISSION ASSUMPTIONS AND SEP PERFORMANCE

Mission assumptions	
Launch vehicle	Delta IV Heavy
ELV performance after margin	5737.1 kg
ELV performance with margin	6374.6 kg
ELV margin (10%)	637.5 kg
Payload adapter	15.0 kg
ELV margin less adapter	622.5 kg
Net ELV margin	9.8%
ELV injected C_3	26 km^2/s^2
SEP performance to Saturn	3975 kg

7.3   Trade 2: 3+1

Table 7.8 summarizes the launch vehicle performance hits that lead to the SEP performance to Saturn value. Note that the total SEP Stage wet mass with growth from Table 7.7 (1920) is subtracted from the ELV performance after margin value (5862 kg) in order to arrive at the SEP performance number of 3942 kg. This number is the total mass available to the science probe.

TABLE 7.7.—CASE 3 SYSTEM SUMMARY

S/C system details Master equipment list—Mass	CBE (kg)	Growth (kg)	Total Mass (kg)	Percent of dry mass
Science Probe	N/A	N/A	3942	N/A
SEP Stage				N/A
Avionics and communications	13.9	2.8	16.7	1.4
GN&C	0.8	0.4	1.2	0.1
Electrical power	500.2	92.4	592.6	50.9
Thermal control	60.5	10.9	71.4	6.1
Structures	117.3	20.3	137.7	11.8
Propulsion	313.9	31.5	345.4	29.6
Propellant	611.1	0.0	611.1	N/A
SEP Stage total mass without system growth	1617.8	158.2	1776.0	
SEP Stage dry mass	1006.7	158.2	1165.0	
SEP Stage inert mass	1012.8	N/A	1171.0	
SEP Stage total mass with system growth		143.8	1919.8	
System level growth (contingency) tracking				
Dry mass growth (contingency)	15.7%			
Desired total growth (contingency)	30%			
Desired total growth (contingency)	302 kg			
Mass system level growth (contingency)	143.8 kg			

TABLE 7.8.—CASE 3 MISSION ASSUMPTIONS AND SEP PERFORMANCE

Mission Assumptions	
Launch vehicle	Delta IV Heavy
ELV performance after margin	5861.7 kg
ELV performance with margin	6513.0 kg
ELV margin (10%)	651.3 kg
Payload adapter	0.0 kg
ELV margin less adapter	651.3 kg
Net ELV margin	10.0%
ELV injected C_3	19 km^2/s^2
SEP performance to Saturn	3942 kg

Appendix A.—Acronyms and Abbreviations

AO	Announcement of Opportunity	ISRU	in situ resource utilization
AU	Astronautical Unit	JPL	NASA Jet Propulsion Laboratory
BOL	beginning of life	KSC	NASA Kennedy Space Center
CAD	computer aided design	LCT	Lunar Communications Terminal
C&DH	Command and Data Handing	LEO	low Earth orbit
CBE	current best estimate	LPA	Low Pressure Assembly
CER	cost estimating relationships	LRS	Lunar Relay Station
CEV	Crew Exploration Vehicle	LSP	Launch Service Program
Comm	Communications	LSTO	Launch Service Task Order
COMPASS	COllaborative Modeling and Parametric Assessment of Space Systems	LVLH	Local Vertical Local Horizontal
		MAC	Media Access Control
		MECO	main engine cutoff
COTS	commercial off the shelf	MEL	Master Equipment List
DCIU	Digital Control Interface Unit	MLI	multilayer insulation
DMR	Design for Minimum Risk	MPU	Makeup Power Unit
DS-1	Deep Space 1	NASA	National Aeronautics and Space Administration
DSN	Deep Space Network		
DTE	direct to Earth	Nav	navigation
ECI	Earth centered inertial	NEXT	NASA Evolutionary Xenon Thruster
EELV	Evolved Expendable Launch Vehicle		
		NIST	National Institute of Standards and Technology
ELV	Expendable Launch Vehicle		
EOL	end of life	NLS	NASA Launch Services
EP	Electric Propulsion	NSTAR	NASA Solar Technology Application Readiness
EPS	Electric Propulsion System		
ESPA	EELV Secondary Payload Adaptor	OMS	Orbital Maneuvering System
FEA	finite element analysis	OTS	off the shelf
FOM	figure of merit	PDU	power distribution unit
FY	fiscal year	PEL	Power Equipment List
GSFC	NASA Goddard Space Flight Center	PMAD	Power Management and Distribution
GLIDE	GLobal Integrated Design Environment	PMS	Propellant Management System
		PN	pseudo-noise
GN&C	Guidance, Navigation and Control	PPU	Power Processing Unit
GRC	NASA Glenn Research Center	RCS	Reaction Control System
HPA	High Pressure Assembly	S/C	spacecraft
HQ	NASA Headquarters	SADA	Solar Array Drive Assembly
IMDC	I M Design Center	SEEGA	solar electric Earth gravity assist
IP	internet protocol	SEVGA	solar electric Venus gravity assist
I_{sp}	specific impulse	SEJGA	solar electric Jupiter gravity assist

SEEJGA	solar electric Earth Jupiter gravity assist	SPACE	System Power Analysis for Capability Evaluation
SEP	Solar Electric Propulsion	SPU	Solar Power Unit
SEPTOP	Solar Electric Propulsion Trajectory Optimization Program	TDRSS	Tracking and Data Relay Satellite System
SLB	Standard LightBand	TWTA	Traveling Wave Tube Amplifier
SN	signal-to-noise	WBS	work breakdown structure

Appendix B.—Rendered Design Drawings

Figure B.1 is the SEP Stage with notional science probe attacked in the solar array deployed configuration. Figure B.2 shows the SEP Stage with the array deployed shown from the bottom of the thrust tube configuration with the NEXT thrusters shown.
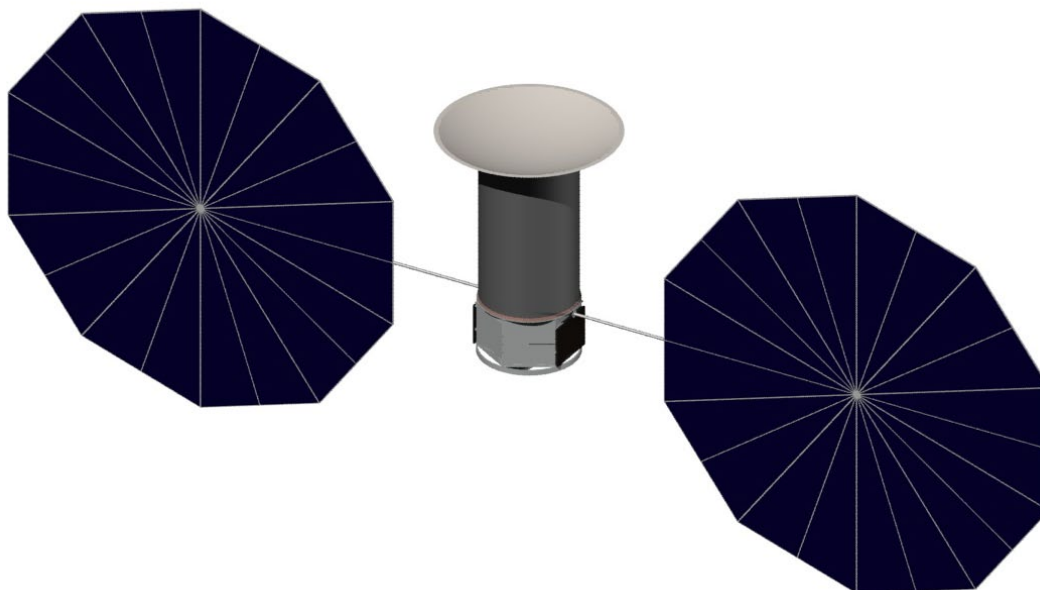

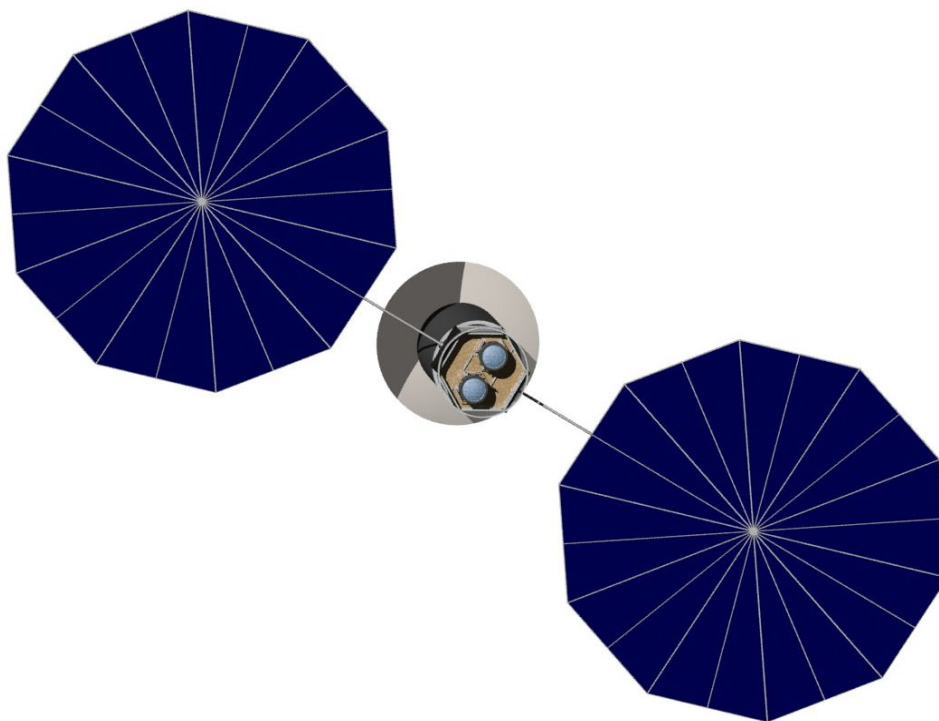Figure B.1.—SEP Stage with Science Probe in deployed array configuration.

Figure B.2.—SEP Stage with array deployed, looking at NEXT thruster face.

Appendix C.—Study Participants

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13. SUPPLEMENTARY NOTES

14. ABSTRACT
The results of the NASA Glenn Research Center (GRC) COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) internal Solar Electric Propulsion (SEP) stage design are documented in this report (Figure 1.1). The SEP Stage was designed to deliver a science probe to Saturn (the probe design was performed separately by the NASA Goddard Space Flight Center's (GSFC) Integrated Mission Design Center (IMDC)). The SEP Stage delivers the 2444 kg probe on a Saturn trajectory with a hyperbolic arrival velocity of 5.4 km/s. The design carried 30 percent mass, 10 percent power, and 6 percent propellant margins. The SEP Stage relies on the probe for substantial guidance, navigation and control (GN&C), command and data handling (C&DH), and Communications functions. The stage is configured to carry the probe and to minimize the packaging interference between the probe and the stage. The propulsion system consisted of a 1+1 (one active, one spare) configuration of gimbaled 7 kW NASA Evolutionary Xenon Thruster (NEXT) ion propulsion thrusters with a throughput of 309 kg Xe propellant. Two 9350 W GaAs triple junction (at 1 Astronomical Unit (AU), includes 10 percent margin) ultra-flex solar arrays provided power to the stage, with Li-ion batteries for launch and contingency operations power. The base structure was an Al-Li hexagonal skin-stringer frame built to withstand launch loads. A passive thermal control system consisted of heat pipes to 'north and south' radiator panels, multilayer insulation (MLI) and heaters for the Xe tank. All systems except tanks and solar arrays were designed to be single fault tolerant.

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