NASA/TM—2011-216971 CD–2007–16

COMPASS Final Report: Radioisotope Electric Propulsion (REP) Centaur Orbiter New Frontiers Mission

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February 2011

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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. The team would also like to thank the In Space Propulsion Office for always providing challenging design study ideas.

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1.0 Executive Summary

Radioisotope Electric Propulsion (REP) has been shown in past studies to enable missions to outer planetary bodies including the orbiting of Centaur asteroids (Figure 1.1). Key to the feasibility for REP missions are long life, low power electric propulsion (EP) devices, low mass Radioisotope Power System (RPS) and light spacecraft (S/C) components. In order to determine the key parameters for EP devices to perform these REP missions a design study was completed to design an REP S/C to orbit a Centaur in a New Frontiers (NF) cost cap. The design shows that an orbiter using several long lived (~200 kg xenon (Xe) throughput), low power (~700 W) Hall thrusters teamed with six (150 W each) Advanced Stirling Radioisotope Generators (ASRG) can deliver 60 kg of science instruments to a Centaur in 10 yr within the NF cost cap. Optimal specific impulses (I_{sp}) for the Hall thrusters were found to be around 2000 s with thruster efficiencies over 40 percent. Not only can the REP S/C enable orbiting a Centaur (when compared to an all chemical mission only capable of flybys) but the additional power from the REP system can be used to enhance science and simplify communications.

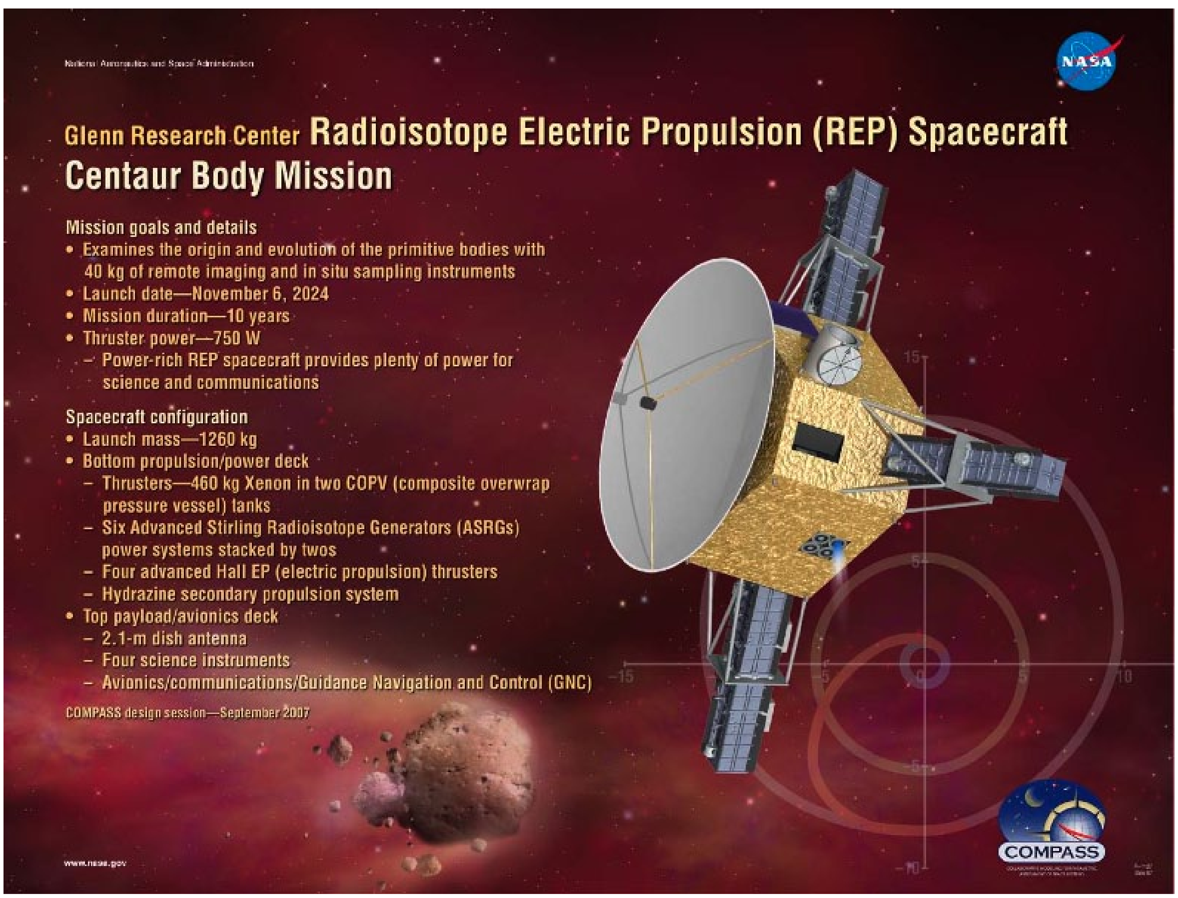

Figure 1.1.—Mission overview.

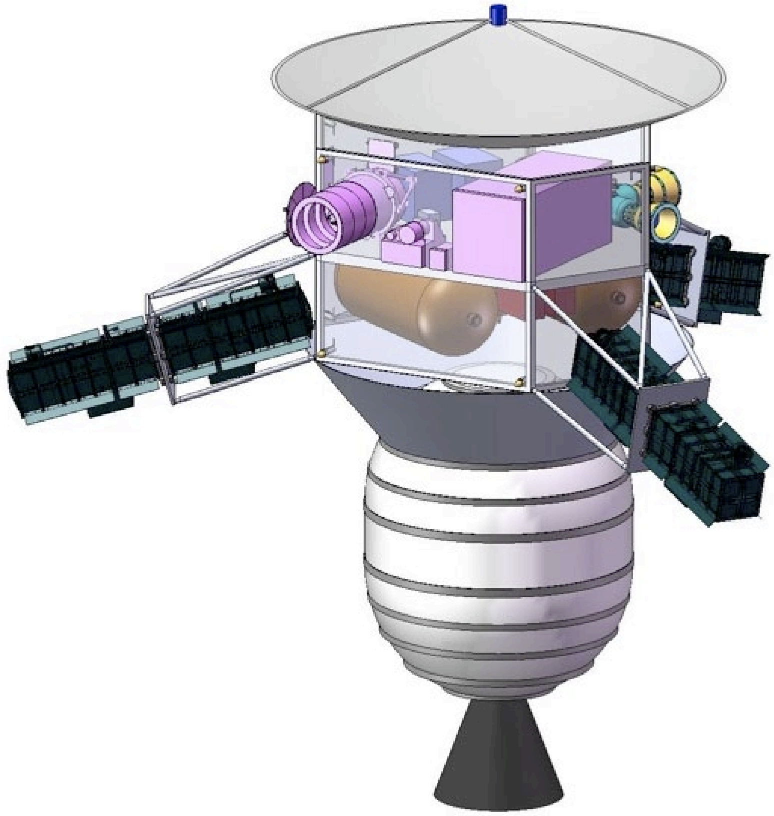

Figure 1.2.—Conceptual REP Science S/C with Star 48 Rocket.

The mission design detailed in this report is a Radioisotope Power System (RPS) powered EP science orbiter to the Centaur Thereus with arrival 10 yr after launch, ending in a 1 yr science mapping mission. Along the trajectory, approximately 1.5 yr into the mission, the REP S/C does a flyby of the Trojan asteroid Tlepolemus. The total ΔV of the trajectory is 8.9 km/s. The REP S/C is delivered to orbit on an Atlas 551 class launch vehicle with a Star 48 B solid rocket stage. Figure 1.2 shows the conceptual S/C reported on in this document.

Table 1.1 summarizes the REP S/C and mission. It is evident that the largest systems are power and propulsion, which enable the timely mission to orbit a Centaur. While the power is more abundant than usual for S/C of this class, it is not large enough to allow for the high I_{sp}s (>3000 s) normally associated with solar electric propulsion (SEP) missions to Mars or nearer the Sun. Thus propellant loading is significant. The additional power used for transit is available for science and communications at the Centaur. Preliminary assessments point towards the use of higher power communications to reduce Deep Space Network (DSN) time and reduce operations costs.

TABLE 1.1—MISSION AND S/C SUMMARY

Mission	10 yr mission, orbit Centaur (Thereus) in 10 yr, flyby Trojan (Tlepolemus) in 1.5 yr, 8.9 km/s	Total mass with growth
Launch	Atlas 551/Star 48, C_3 97.28 km^2/s^2	1260 kg (wet)
Science	Six instruments consuming 60 W, remote imaging (wide angle camera (WAC) and narrow angle camera (NAC) cameras, Laser Detection and Ranging (LIDAR), Near-Infrared Mapping Spectrometer (NIMS) and in-situ sampling (Neutral Gas and Ion Mass Spectrometer (NGIMS), DFM, 130 Mb per day	57 kg
Power	Six ASRG with multilayer insulation (MLI), attached (loaded) in pairs, 900 W end of life (EOL) (10-yr)	199 kg
Propulsion	Primary electric: 3+1 Long Life Hall Thrusters, operated serially, 500 kg Xe propellant Load, 650 W into thruster, 1920 s I_{sp}, 170 kg Xe throughput each, single string power processing units (PPU) (95%), thruster feed, thruster 30,000 hr Secondary chemical: Blow-down hydrazine, 80 kg propellant, eight 0.25 lbf monoprop thrusters	137 kg
Mechanical	Hexagonal aluminum-lithium (Al-Li) bus with propulsion and science decks, capable of carrying 6 g launch loads	118 kg
Communications	220 W, 8 kb/s data rate, Ka Band, 2.1 m antenna	52 kg
Command and Data Handling (C&DH)	RAD 750 computer with x MB storage capacity	45 kg
Attitude, Determination and Control (AD&C)	Two Star cameras, inertial measurement unit (IMU), four reaction wheels, hydrazine propulsion	22 kg
Thermal	MLI and heaters, ASRG isolated	49 kg

2.0 Study Background and Assumptions

2.1 Introduction

The COllaborative Modeling and Parametric Assessment of Space Systems (COMPASS) team was approached by the NASA Glenn Research Center (GRC) In-Space Project to perform a design session to develop REP S/C Conceptual Designs (with Cost/Risk/Reliability) for missions of three different classes: NF Class Centaur Orbiter (with Trojan Flyby), Flagship, and Discovery Class. The designs will allow trading of current and future propulsion systems. The results will directly support technology development decisions. The mission class documented in this final report is the NF class orbiter mission to a Centaur body, with a Trojan flyby along the trajectory.

Past studies have shown that REP can enable orbiters for outer planetary small bodies. The mission design detailed in this report is an REP powered EP science orbiter to the Centaur Thereus with arrival 10 yr after launch, with a 1 yr science mapping mission, to make a total lifetime requirement for all systems of 11 yr. Along the trajectory, approximately 1.5 yr into the mission, the REP S/C does a flyby of the Trojan asteroid Tlepolemus. The total ΔV of the trajectory is 8.4 km/s. The REP S/C is delivered to orbit on an Atlas 551 class launch vehicle with a Star 48 B solid rocket stage. Figure 1.2 shows the conceptual S/C reported on in this document.

The Executive Summary (Section 1.0) provides a framework for where this study fits into NASA's goals. The rest of the document describes the various aspects of the mission including: mission category, propulsion type and engines, mission duration, scientific payload and its requirements, power requirements, communications requirements, launch vehicle and a specific mission target (Thereus). As with any study, there are many options among these mission elements. Many of them are traded for comparison in order to provide optimal scientific results, minimize cost and provide the highest probability of mission success (as defined by established goals). There is a preliminary discussion on past missions that serves as a set of baseline designs. The science mission objective is to determine the origin and evolution of Centaurs and Trojan bodies. For example: What is the density, volume, rotation state, albedo of the "typical" Centaur?

Are Centaurs the direct descendants of Kuiper Belt Objects (KBOs)? How do Centaurs compare to the icy small satellites of Saturn (Phoebe, Hyperion) that are supposedly KBOs? Were Trojan asteroids formed in place, as part of the outer main belt of asteroids, or were they scattered into the Kuiper Belt as the planets migrated outward about 800 M years after the formation of the Earth?

2.2 Purpose

The goal of the REP science S/C is to send a science payload to a Centaur class body. The Centaurs are planetary bodies in orbit about the Sun and located between the orbits of Jupiter and Neptune. These bodies are typically asteroidal to comet-like in appearance and physical makeup. If possible, the mission flight plan will include a flyby of one of the Jupiter Trojan asteroids. The Trojan asteroids occupy one of the Lagrange points created by Jupiter and the Sun.

This study will utilize the COMPASS S/C Conceptual Design team to provide complete Science Class Reference Mission Designs.

Prework that was completed before the official study kickoff

- Determine Science Class missions that are compelling and relevant: Applied Physics Laboratory (APL)
- Setup Mission analyses capabilities for compelling missions: GRC
- Develop a suite of EP systems to trade on S/C designs: GRC
- Develop Concept of Operations (CONOPS) and suite of science instruments and requirements for each science mission class: APL

2.3 Assumptions and Approach

The following section contains the mission description of the class of mission being designed in this REP study. Design details are the most current available at the time of this design session. This study is focusing on a NF Class Mission design.

2.3.1 NASA NF Class Mission Definition

NASA has named New Horizons (NH) as the first NF class mission of the solar system. From the NASA NF Website, the definition of a NF mission at the time of this study was as follows

> *"The Office of Space Science (OSS) of the National Aeronautics and Space Administration (NASA) announces an opportunity to propose for two different types of scientific investigations through the New Frontiers Program: New Frontiers Mission investigations specifically for the planet Venus, the Earth's moon, Jupiter, or a comet nucleus (see Section 2.1); and Mission of Opportunity investigations."*
>
> *(Extract: AO-03-OSS-03 Introduction)*
>
> *New Frontiers Mission investigations are to be completed through space flight missions launched on Expendable Launch Vehicles (ELVs) no later than June 30, 2010, in order to accomplish science objectives in compliance with those stated in Section 2.1 of this Announcement of Opportunity (AO) for a total NASA OSS Cost through mission completion not to exceed $700M in Fiscal Year (FY) 2003 dollars (see Sections 5.1 through 5.11 for a description of both general and specific restrictions).*
>
> *New Frontiers Mission of Opportunity (MO) investigations are part of non-OSS space missions of any size that will be launched no later than December 31, 2008, that require a commitment from NASA before December 31, 2005*

2.3.2 Past Design Starting Points

The REP Centaur Science S/C and mission used the APL NH S/C as its starting point for the baseline design. The APL NH S/C is the first NF class vehicle (NH website). The NH mission is a 10 yr mission to map the Kuiper belt region beyond Neptune, and as such, has both a mission and trajectory similar to the objectives of the Centaur body orbiter to be designed in this COMPASS session.

The initial design of the REP S/C was based on the specifications of the NH S/C. For reference, the NH mission and S/C Design Highlights are as follows

- Launch date/time: January 19, 2006, at 19:00 UTC on an Atlas V 551 using a Star 48B solid third stage
- On-orbit dry mass: 385 kg
- Nominal power output: 228 W
- Life: +15 yr
- Propulsion: Blow down hydrazine: 80 kg of fuel
- Stabilization: Spin and three-axis depending on mission phase
- Thermal: ‘Thermal Bottle’ using heaters/louvers powered by shunts from ASRG
- Science: 32.5 kg/69.7 W total (25.9 W max mode)
- Utilized hibernation mode

The NH S/C supports seven science instruments and is powered by an ASRG. Figure 2.1 is a graphic of the NH science S/C portion without the Star 48B attached. Figure 2.2 shows the NH S/C being assembled in a clean room.

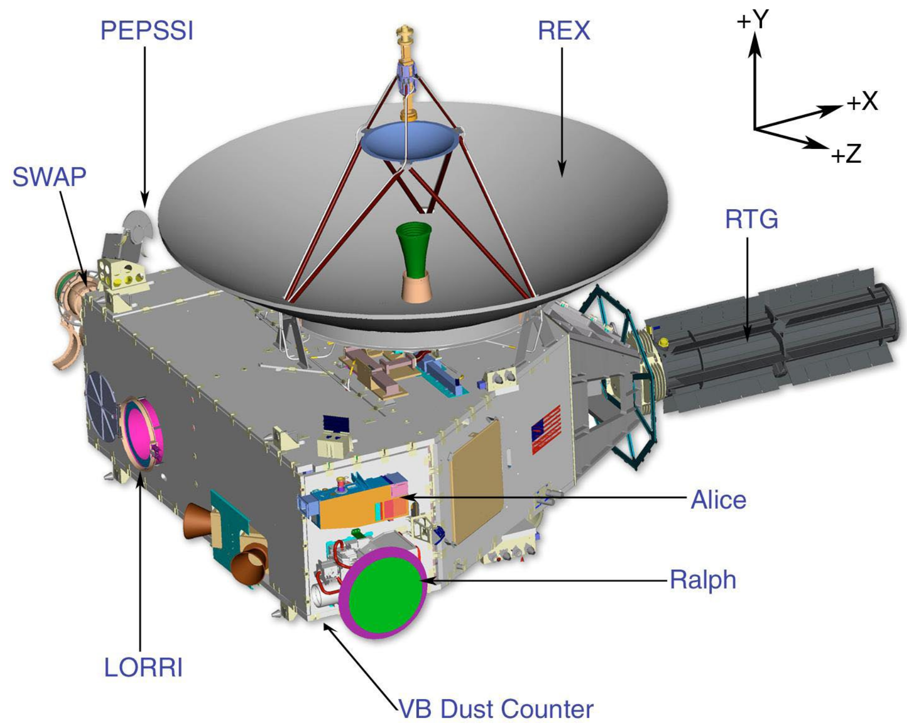

Figure 2.1.—The NH S/C.

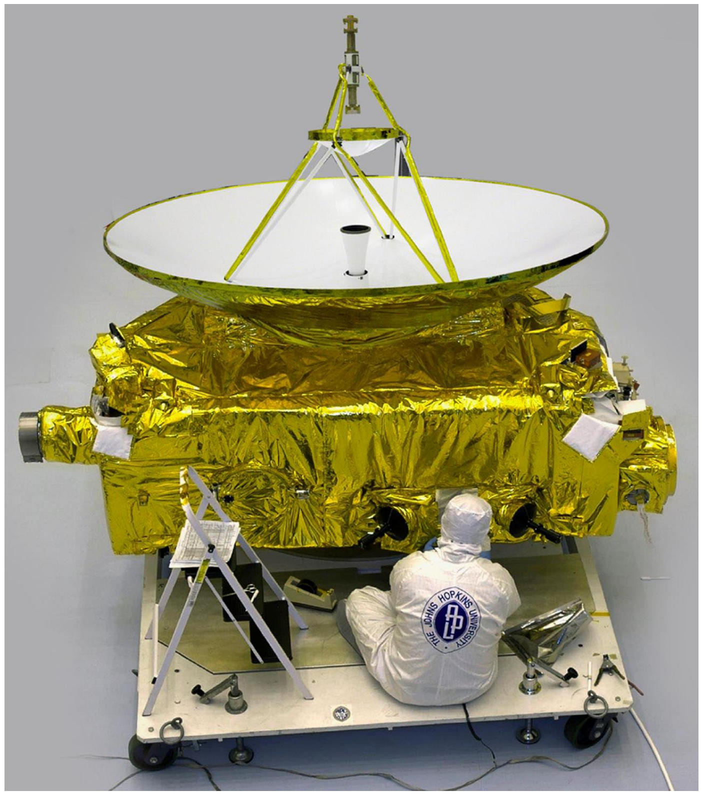

Figure 2.2.—NH S/C clean room assembly.

2.4 Growth, Contingency and Margin Policy

***Mass Growth*:** For dry mass elements in the system design, the COMPASS team uses the American Institute Aeronautics and Astronautics (AIAA) R-020A-1999, "Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles."

Table 2.1 shows the percent mass growth separated into a matrix specified by level of design maturity and specific subsystem. In this document mass growth and weight growth are used interchangeably. Weight growth allowance (WGA) and mass growth allowance (MGA) mean the same thing when expressed as a percent value.

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, and an 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 Discovery 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 4.4 for the power system assumptions.

Science payload mass and power margins assumptions are: The science payload package was designed by the engineering staff at APL. They recommend the following margins be used on their science package. Therefore, the science package uses a different MGA and power margin than the rest of the S/C design.

- 30 percent margin should be *added* for mass estimates (mass estimates do not include any additional shield that may or may not be needed with the REP mission)
- 30 percent margin should be *added* for power estimates

TABLE 2.1.—PERCENT MGA

Code	Design maturity (basis for mass determination)	Percent MGA									
		Electrical/electronic components			Structure	Thermal control	Propulsion	Batteries	Wire Harnesses	Mechanisms	Instrumentation
		0-5 kg	5-15 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	Prerelease 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

2.5 Mission Description

The primary requirement for the mission is orbit a Centaur class body in the minimum amount of time while delivering the maximum payload. Along the trajectory, if possible, a flyby of a Trojan asteroid will be incorporated into the mission profile. The representative Centaur body chosen for the mission was Thereus. Thereus, an average sized Centaur in the Saturn-Uranus region, was found to be representative of several Centaur targets in terms of its known behavior and required delivery ΔV.

Figure 2.3 shows the grouping of Centaur objects in the solar system region around Saturn, Uranus, and Neptune. The orbits of the known Centaurs and Neptune Trojans are show in Green. The inclination of the orbit is the Y-axis. Using the mapping of Centaur bodies in Figure 2.3, and the low thrust mission analysis code VARITOP, a suite of trajectories to Centaur bodies was flown to find a mission that would fit the science requirements and deliver the inert mass from the bottoms-up S/C system design.

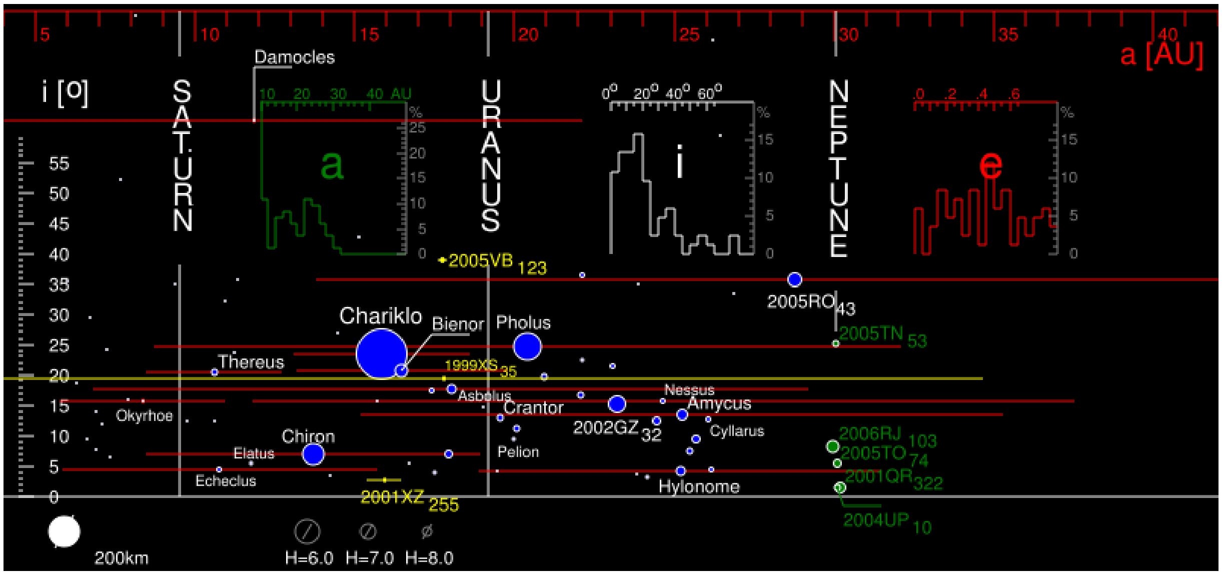

Figure 2.3.—Centaur body distribution.

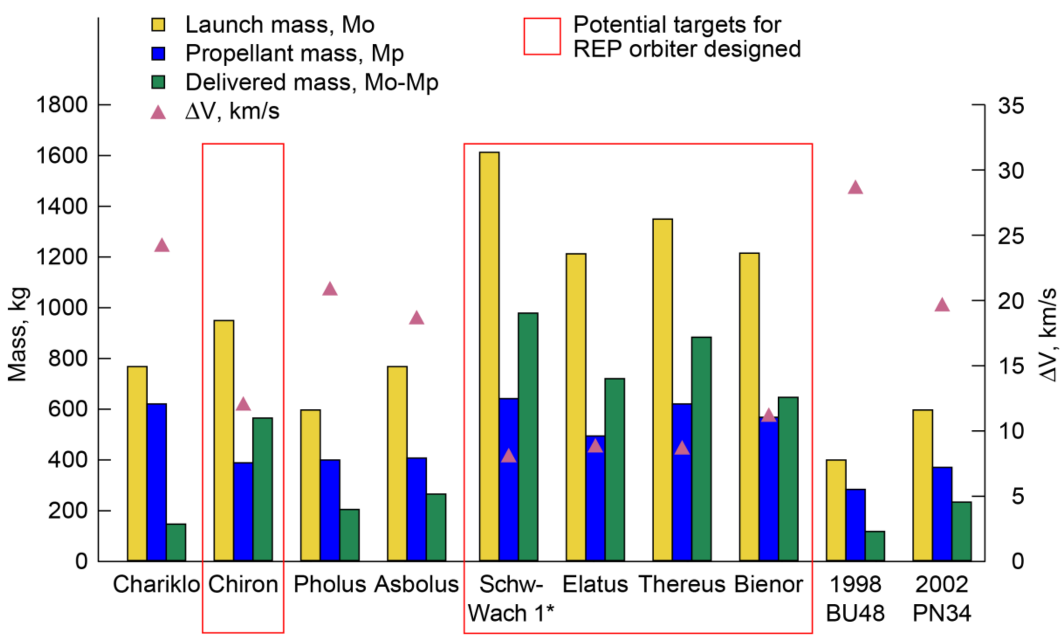

Figure 2.4.—Performance comparison.

While the initial Centaur body selection is done over a wide range of bodies, the iterative process of mission and system design done during the design session limits the selection of Centaur bodies to those with trajectory C_3 is similar to the performance C_3 of the ELV (Atlas 551/Star 48B) to accommodate the inert mass of the S/C design.

Next, Figure 2.4 shows the Centaur body target comparison in terms of mission performance parameters of launch mass, necessary propellant mass and delivered payload mass to the target, as well as total mission ΔV. The two red-boxed areas are the potential target Centaurs that fit within the initial assumed performance of the REP S/C and thrusters.

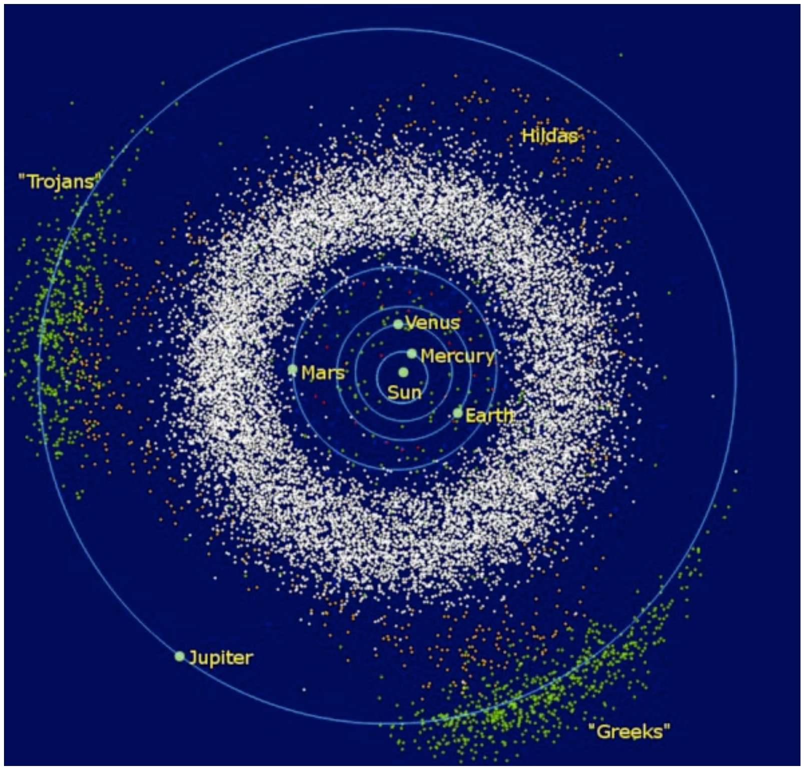

Figure 2.5.—Distribution of Trojan Asteroids in Jupiter's orbit.

Figure 2.5 shows the distribution of the Trojan asteroids in the orbit of Jupiter. The clusters of green objects located behind Jupiter in its orbit are the Trojans, and the clusters in front of Jupiter in its orbit are the Greeks. Both areas in the Jovian orbit have been added into the trade space of target objects for close flyby reconnaissance along the way to the Centaur object.

2.5.1 Trajectory Baseline—Thereus

Preliminary design of the REP Centaur orbiter used the Centaur body, Bienor, as a target. After the REP S/C bottoms-up analysis and sizing, the amount of payload delivered to the Centaur body Bienor (see Appendix D) was determined to be insufficient to accommodate the baseline S/C design and science instruments mass. Therefore, mission analysis was performed to locate another Centaur body trajectory that would place the REP S/C and science instruments at the Centaur before it reaches the Centaur body's perihelion (closest approach to the Sun as required by the mission science drivers, while allowing enough delivered mass to accommodate the REP S/C. Figure 2.6 shows the net delivered mass as a function of trip time to the Centaur body Thereus. It can be observed that net mass delivered for trip times greater than 9 yr begins to level off.

The Centaur Thereus was then chosen to fit those parameters. There is no coast phase in the mission but a 10 percent thrust margin is included in the analysis. The baseline trajectory parameters for the Thereus trajectory were:

- Launch mass (Mo): 1260.2 kg (Atlas 551 performance to C_3)
- C_3: 97.28 km^2/s^2
- Launch date: November 15, 2024
- Fly-by date: August 7, 2026
- Arrival date: November 13, 2034
- Power: 650 W
- I_{sp}: 2057 s
- ΔV: 8.93 km/s

- Propellant (Mp): 450.8 kg (used to perform ΔV)
- Initial mass (Mo): 1260 kg (starting mass from ELV performance to C_3)
- Mo – Mp: (1260 – 450) 809 kg
- Fly-by target: Trojan Asteroid Tlepolemus

This trajectory allowed for an REP inert mass of 809 kg. This inert mass includes the residuals and margin in the Xe propellant. The 450 kg is the ideal Xe propellant used to fly the ideal trajectory ΔV. With the 30 percent margin and the extra Reaction Control System (RCS) propellant, which is not included in the ideal ΔV modeling via the low thrust trajectory, the REP S/C was able to fit inside this “inert” mass box. See Table 4.3 in Section 4.3.3 to see the system masses, with margin, and S/C adapter to come up with the launch margin for the REP S/C.

Figure 2.7 shows the launch window analysis on the baseline Thereus trajectory. A 10-day launch window will result in minimal impact to the trajectory. A 20-day launch window may require slightly longer trip time. Re-optimizing the trajectory to include coast periods could alleviate this issue.

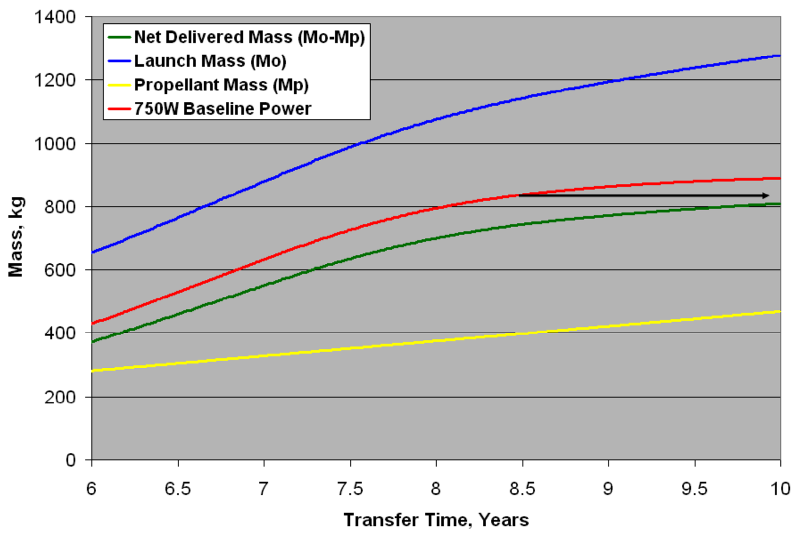

Figure 2.6.—Net mass delivered versus transfer time.

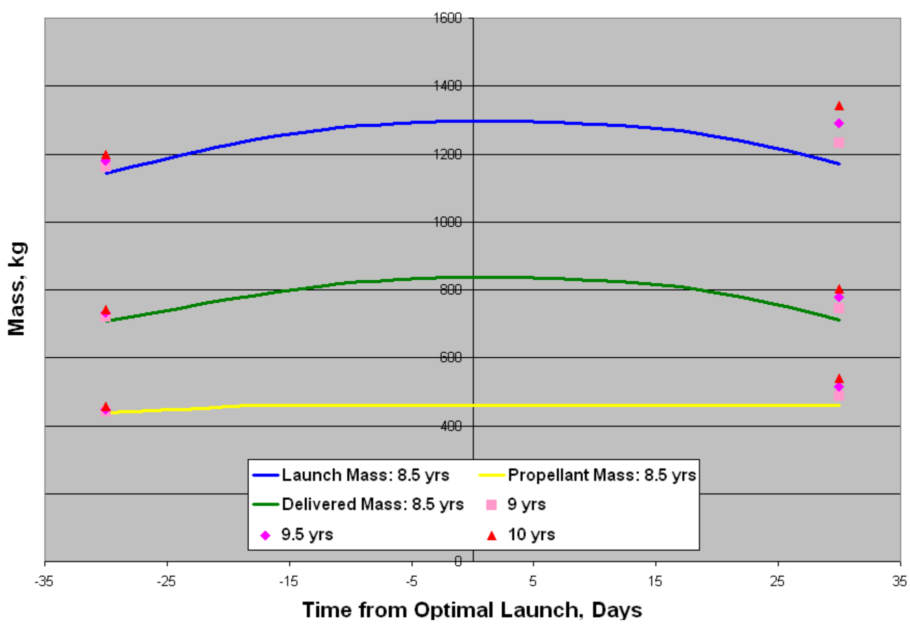

Figure 2.7.—Launch window analysis.

2.5.2 Mission Analysis Analytic Methods

The low thrust mission analysis performed for this mission was done using the low thrust optimization code VARITOP, developed at the NASA Jet Propulsion Laboratory (JPL). Typically a maximum duty cycle of 90 percent is imposed on the trajectory optimization using the updated program, SEPTOP. Because VARITOP does not have that feature, the thrust and therefore efficiency was reduced to 90 percent of the anticipated performance. This interjected 10 percent thrust margin artificially puts 10 percent coasting times into the trajectory to alleviate the launch window analysis implications in Figure 2.7.

The optimization parameters for running this trajectory were as follows in Table 2.2. Note that the mission duration quoted in the mission analysis assumptions is trip time to the Centaur Body. An additional year of analysis is performed at the Centaur making the total mission time 11 yr.

TABLE 2.2—MISSION ANALYSIS ASSUMPTIONS

Launch vehicle	Atlas 551 with Star 48	Optimized
Launch vehicle contingency	10%	Fixed
Epoch date	April 22, 2024	Fixed
Launch date	November 15, 2024	Optimized
Trajectory mission duration	10 yr	Fixed
Thruster power	750 W	Fixed
Thruster efficiency	Hall Curve	Fixed
I_{sp}	1800 s	Fixed

The thruster performance was modeled using the thruster curves shown in Figure 2.8. These are based on a single operating point thruster performance estimates provided by GRCs EP division. The I_{sp} and sensitivity was initially optimized for the mission and then were fixed for parametric analyses on the chosen system. Trades are shown later in Section 6.0.

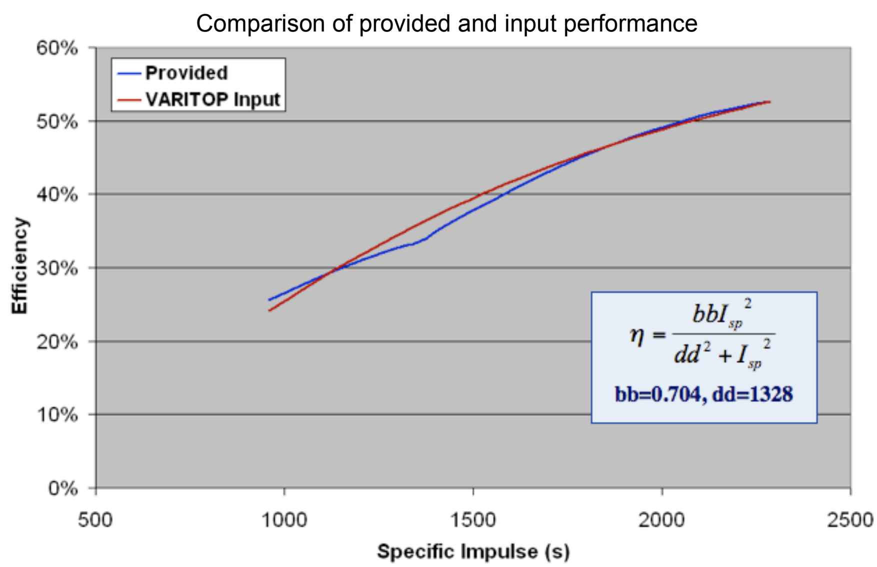

Figure 2.8.—Thruster performance modeling assumptions.

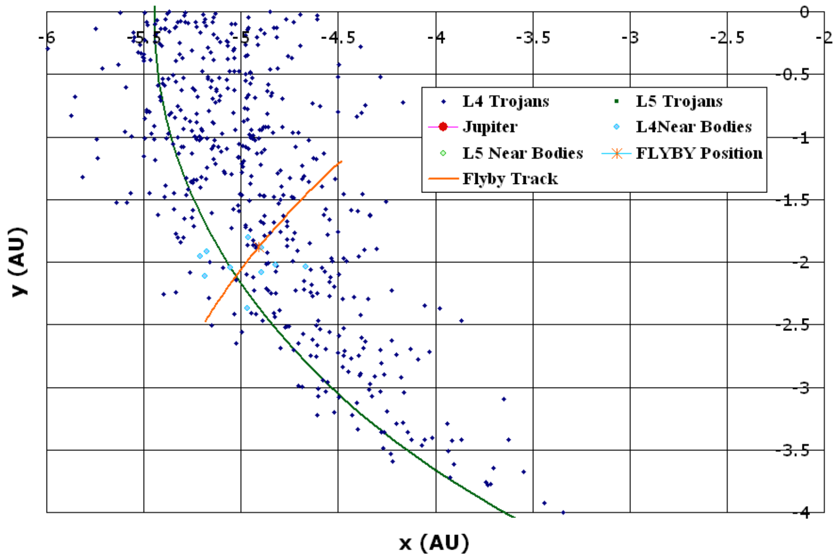

Figure 2.9.—Potential Trojan flybys along trajectory.

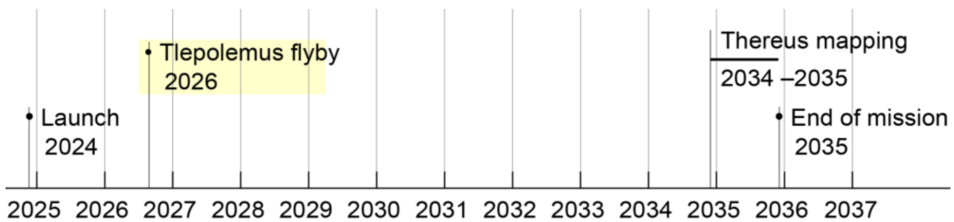

Figure 2.10.—Mission timeline.

2.5.3 Trojan Flyby Analysis

Given the launch dates for the optimal Thereus trajectory, the possibilities of encountering a Trojan asteroid were greater for the Thereus mission than they were for the Bienor mission. Figure 2.9 shows the potential Trojan flybys as the REP science S/C passes through the asteroid belt. With thousands of Trojan asteroids to choose from, it can be seen that multiple Trojans were within flyby range without any targeting necessary by the S/C, and therefore, without extra propellant budgeting necessary.

Based on the available Trojan asteroids during the flyby, the Trojan asteroid, Tlepolemus, was chosen as the flyby target. Tlepolemos is the named Trojan nearest to the Earth in an orbit closer than any other of the Trojans to Earth’s orbit.

2.5.4 Mission Analysis Event Timeline

- Launch date: November 15, 2024
- Fly-by Trojan date: August 7, 2026
- Arrival at Centaur date: November 13, 2034

A benefit of this mission and trajectory is that First Science with a Trojan flyby is available less than 2 yr after launch. This allows for testing of the science instruments for the final Centaur mission along with the capture of interesting science data of a Trojan asteroid. Figure 2.10 shows the relative mission timeline of the major events in the mission: Trojan flyby, Centaur body arrival and science mapping.

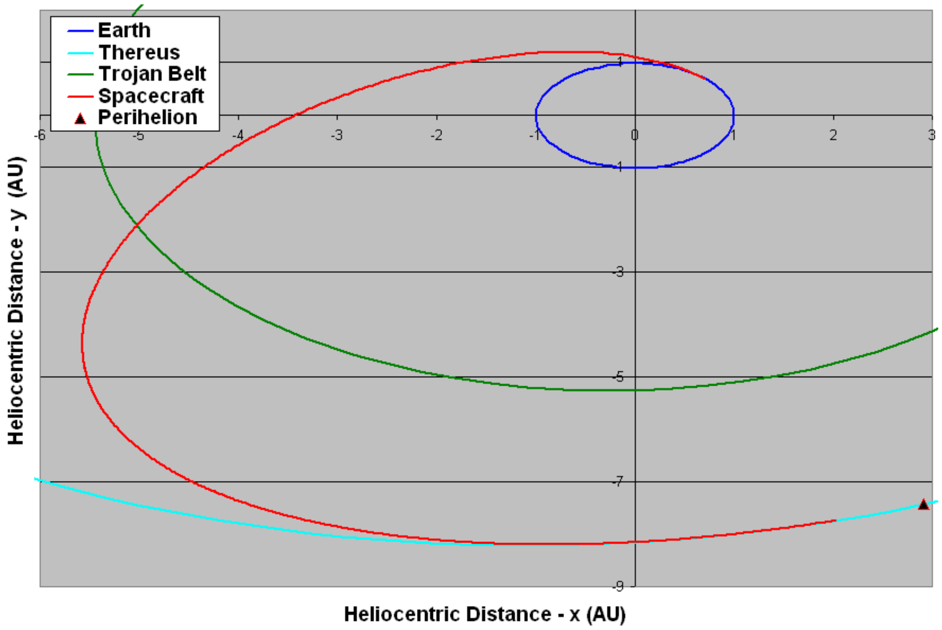

Figure 2.11.—Trajectory from Earth to Thereus, with Trojan Asteroid flyby.

2.5.5 Mission Trajectory Details: Six ASRG Case

Figure 2.11 shows the trajectory to Thereus with a flyby of the Trojan asteroid Tlepolemus. The green orbit is the Trojan asteroid belt, and the aqua blue orbit is that of the Centaur body Thereus. The small triangle on the orbit of Thereus indicates perihelion. The goal of the arrival at Thereus is to arrive at the Centaur body prior to its perihelion, in order to take science data at the Centaur as it goes through its closest approach to the sun and encounters its maximum temperatures and has the highest probability of activity. For reference, the Earth's orbit about the Sun is shown in blue.

The CONOPS of the mission is based on that of the Deep Space One Mission that validated EP technology with the flight of the NSTAR engine. The Ground Trajectory Planning updates monthly. Orbit determination occurs weekly via an on-board camera every 4 to 6 hr. The "Autonav" system can overwrite, during retargeting, a real-time ephemeris, schedules and executes events and navigation update computations. The Reduced State Encounter Navigation (RSEN) operations mode begins several hours prior to encounters and maintains visual lock from the Earth to the REP S/C.

2.6 Launch Vehicle Details

The baseline launch vehicle is the Atlas 551 with a Star 48 solid propellant upper stage. The launch vehicle performance versus launch C_3 is shown in Figure 2.12. The launch vehicle contingency was assumed to be 10 percent, and was generated using the low thrust trajectory code VARITOP. This data assumed that the Star 48 had already performed its burn. Consequently, the mass of the Star 48 will not be included in the REP S/C master equipment list (MEL).

Figure 2.13 shows the packaging of the REP S/C and Star 48 engine in the Atlas 551 short 5 m payload fairing. Note that the Atlas V payload shroud volume more than accommodates the REP Science orbiter S/C, leaving plenty of room for design changes such as a larger antenna diameter or an increase in the number of ASRGs. See Figure B.1 in Appendix B for the dynamic envelop of the REP S/C inside of the launch vehicle fairing.

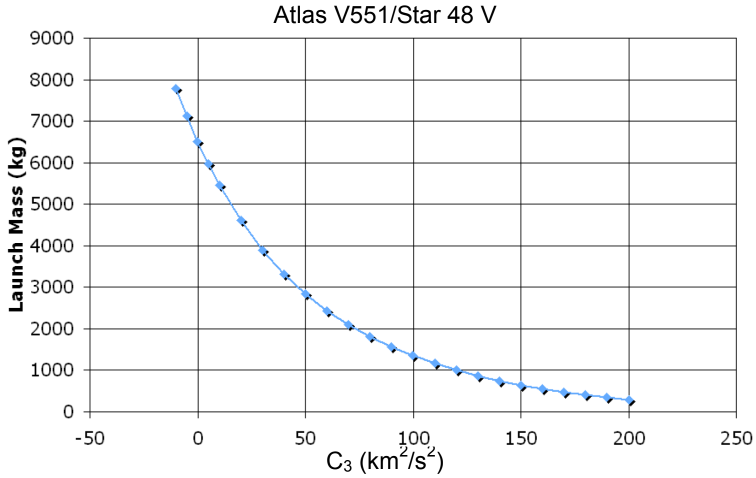

Figure 2.12.—Launch mass versus C_3.

Figure 2.13.—REP S/C in Atlas 551 payload shroud.

2.7 Science and Science Instruments Overview

Typically, the science payload delivered by the bus in a COMPASS design session is the figure of merit (FOM) of the analysis. In this design, the APL science team brought the science instruments to the COMPASS team for inclusion in the REP orbiter design. The goal, then, of the design was to fit the bus and science orbiter package inside of the performance (delivered mass) to the chosen C_3 target by the launch vehicle. Anything left over was considered margin.

The initial desired science payload from APL consisted of six core instruments base-lined for the Centaur REP mission study. The instruments in order of priority are:

(1) Imager—Long Range Reconnaissance Imager (LORRI)
(2) Thermal mapping—Thermal Emission Imaging System (THEMIS)
(3) Ranging and topography—LIDAR
(4) Imager—WAC
(5) Spectrometer—NIMS
(6) Gas spectrometer—NGIMS

The instruments are not required to have a separate processor (i.e., dual processing unit (DPU)) and will interface directly with the Main Processor of the REP bus. Because Centaurs cross the orbits of the planets, their own orbits are unstable, evolving rapidly. This makes the Centaur mission a challenging one. However, in doing the design, the science instrument NGIMS was removed from the package to save on mass, cost and power.

The proposed mission includes a flyby of a Trojan asteroid with limited science to be performed during that flyby. Once at the Centaur body, up to a year of science mission operations will be performed. Therefore, there must be enough power and attitude control propellant to sustain the vehicle for that amount of time.

Table 2.3 lists the major details (mass, power, cost, etc.) of the six science instruments initially evaluated by APL for inclusion on the REP Centaur conceptual S/C being designed in this session. Note that not all of the instruments were included in the final science package used in the design and reported in the science instrument MEL and therefore flown on the S/C in this design. In order to save on mass, the science package suite was reduced. New THEMIS was the first instrument dropped in order to save mass.

TABLE 2.3.—SCIENCE INSTRUMENTS FOR CENTAUR MISSION

Instrument name	Mass (kg)	Power (W)	Dimensions (cm)	Output data rate	Estimated data volume*	Estimated cost (FY07)
LORRI	8.32	2.11	62 by 48 by 35.5	12.6 Mbps	TBD	TBD
New THEMIS	12.2	14	29 by 38 by 55	614.4 Kbps	TBD	$12 million
WAC	0.6	4	14.5 by 9.2 by 7.6	TBD	TBD	$2 million
NIMS	18	12 W average 13 W peak	83 by 37 by 39 (optics) 20 by 25 by 13 (electronics)	11.52 kbps	TBD	??
LIDAR	5	16.5 W average 20.7 peak	37.5 by 21.6 by 22.9	51 bps	TBD	$8 million
NGIMS	10.50	24	19 by 24.7 by 36.2	1.5 Kbps	TBD	$14 million

*A total of 130 Mbits/day will be baselined and allocated between the instruments

Table 2.4 lists the MEL for the science instruments as modeled in the COMPASS design session. Note that the value in the quantity column of the MEL for NGIMS has been zeroed. As the design progressed, NGIMS was also scaled back out of the science instrument suite in the interest of saving mass. New THEMIS was entered back into the Science instrument suite during the analysis.

TABLE 2.4.—REP SCIENCE S/C TOP LEVEL BOTTOMS-UP MEL

WBS no.	Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
01	REP S/C (Payload and Stage)	-	-----	1071.00	11.68	125.14	1196.14
01.1	Science Payload	-	-----	44.12	30.00	13.24	57.36
01.1.1	APL science instruments	-	-----	25.52	30.00	7.66	33.18
01.1.1.a	LORRI	1	8.32	8.32	30.00	2.5	10.82
01.1.1.b	New THEMIS	1	12.2	12.20	30.00	3.66	15.86
01.1.1.c	LIDAR	1	5.0	5.00	30.00	1.50	6.50
01.1.1.d	NGIMS	0	10.5	0.00	30.00	0.00	0.00
01.1.2	Additional science instruments	-	-----	18.60	30.00	5.58	24.18
01.1.2.a	WACS	1	0.60	0.60	30.00	0.18	0.78
01.1.2.b	NIMS	1	18.00	18.00	30.00	5.40	23.40

2.7.1 Science CONOPS

Orbital CONOPS

- The S/C will orbit the Centaur Body for at least 1 yr
- The S/C will orbit the body in the plane of Earth's sky (i.e., the S/C will always be able to communicate with the Earth and will not eclipse behind the body)
- Before orbit insertion, upon approach to the body, the LORRI instrument will be required to capture at least two optical navigation images per day
- The minimum daily science collection will be 130 Mb/day while in Thereus orbit
  - The instruments and recorder allow for a higher collection rate if desired
- Primary ground contact operations concept:
  - Three 8 hr contacts per week using the DSN—70 m Dish, yielding 24 hr/wk of contact time and ~19.5 hr/wk of actual downlink time (at 6.5 hr/contact).
  - Possible ~1 Gbits/day of science/housekeeping collection
    - Assumed minimum 45 kbits/s downlink at 70 m dish with 100 W of radio frequency (RF) radiated power
  - DSN total 1 yr orbital cost: FY07 $7.4M ($23M for 1 yr of 7 tracks per week)

Trojan Flyby CONOPS

- Science collection at the Trojan flyby consists of two data collection periods
  - For 10 days leading up to closest approach to the asteroid, the LORRI instrument will capture at least six images per day
    - Assume 50 percent duty cycle of the EP thrusters during this time
  - The 3 days of closest approach (1 day before, 1 day during, 1 day after) will consist of all instruments collecting data
    - Assume 0 percent duty cycle of the EP thrusters during this time
  - It is estimated that ~ 8 Gbits of science data will be collected during the entire flyby
- Total non-thrusting period during asteroid flyby would be approximately 12 days broken down as follows
  - 5 days of non-thrusting leading up to closest approach
  - 3 days of non-thrusting during closest approach
  - 52 hr of data downlink for 8 Gbits during non-thrusting periods
    - Assume 70 m dish and S/C can downlink at minimum 45 kbits/s
  - 2 days of contingency

2.8 System Design Trade Space

The COMPASS design trade space will cover the following mission classes and subdesigns within those classes of mission as noted herein. This first document will only deal with the initial baseline NF class mission, similar to the NH mission. A second design session will be run to study a Flagship mission (see COMPASS document CD–2007–18).

2.9 Baseline System Design

Figure 2.14 shows the basic design of the Centaur REP driven S/C. From the top down in this diagram, the REP S/C consists of a 2.1 m antenna dish located on the top for relay to the DSN. Below the dish sits the payload deck where the science instruments (shown in purple and grey) and avionics instruments are mounted. There are six ASRG units, providing at least 900 W of total S/C power, mounted by twos (dark blue grey) to the S/C bus superstructure, pointing out perpendicularly from the main body of the S/C. Also on the s/c are two Xe tanks (orange), a helium (He) pressurization tank, four Xe Hall EP Thrusters, and a Star 48 solid rocket engine sitting directly below the propellant management equipment.

The REP S/C, shown in Figure 2.15, will be launched on an Atlas 551 with Star 48 Upper Stage (similar to NH launch). The payload will be located in the middle of the REP/Star 48 motor stack just below the antenna dish.

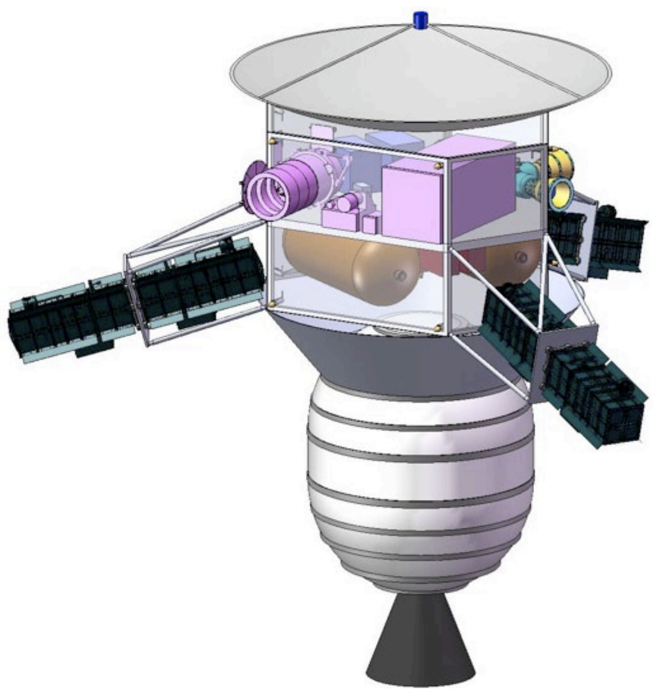

Figure 2.14.—Baseline conceptual REP S/C design on top of Star 48 rocket motor.

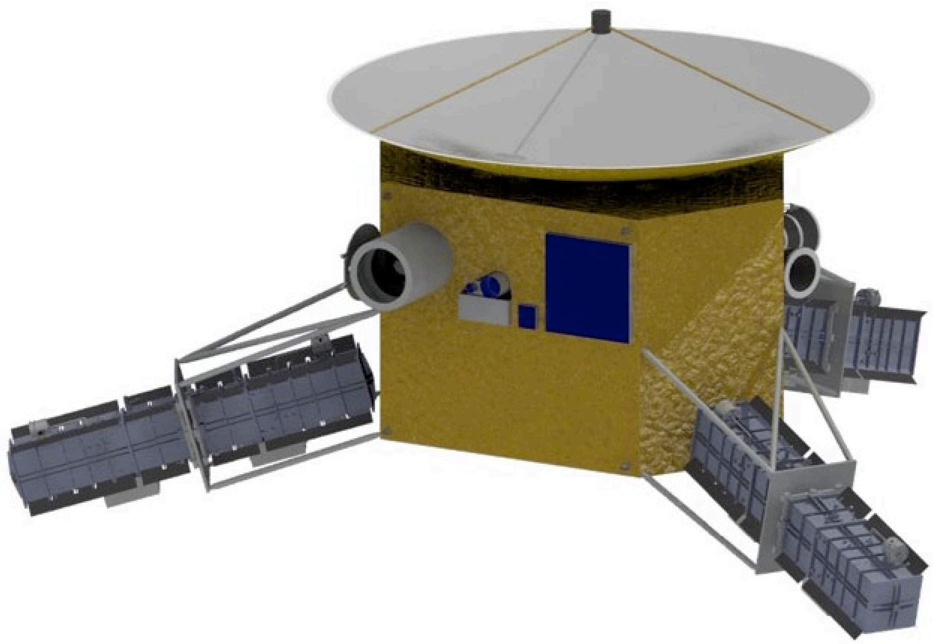

Figure 2.15.—Conceptual REP science S/C.

To first order, the S/C configuration is built around the following major components

- Bottom propulsion/power deck
  - ~500 kg of Xe in two Composite Overwrapped Pressure Vessel (COPV) tanks
  - Six Advanced RPS
    - Stacked by twos
    - Supported by ends with plate struts
  - Four advanced Hall or Ion propulsion systems
    - Thrusting mostly tangential
  - Hydrazine secondary propulsion system
- Top payload/avionics deck
  - 2.1 m dish antenna
  - Four Science payloads—side pointing
  - Avionics/Communications/GN&C
- Bottom launch vehicle interface

The main design challenge for an REP S/C is minimizing S/C masses while integrating a significant Xe propellant and multiple RPS. Fortunately, Xe propellant is very dense and can be stored in carbon fiber overwrapped tanks. The selected RPS was ASRG due to their high efficiency and subsequently small plutonium load for the power delivered. The best balance of power and thrust was found to be around 900 W EOL, with 700 W being fed to the electric thruster power processors. The remaining 200 W was for housekeeping. This power requirement was provided by six ASRGs, each containing about 0.9 kg of plutonium for a total of 5.4 kg. By comparison the NH S/C, using an 80 percent loaded RTG had 11 kg of plutonium.

Besides the obvious advantage of enabling orbiting a Centaur, the REP approach also provides much more power to the science and communications system once the asteroid is reached when compared to a chemical flyby system. During this design it was found that the power was actually more valuable to the communication system. While additional power for science will require additional mass allowance, the communication and data storage systems and operations can be reduced by allowing for higher power communications and either a smaller antenna or reduced DSN time.

2.10 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.

2.10.1 GLIDE Study Share

GLIDE (GLobal Integrated Design Environment) 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. The study team members can find documents saved at the following permissions-limited, webdav-accessible, share.

Study Share: https://glide.grc.nasa.gov/REP_Sept2007

2.10.2 GLIDE Study Container (Architecture)

GLIDE Architecture: REP_Sept2007

2.10.3 GLIDE Study Container(s)

Six ASRG Case: Thereus_6ASRG

Eight ASRG Case: Bienor1

Study Description

NF Class Mission to the Centaur Bienor 1. Although the target has been shifted to Thereus, per the mission section, the study container was not renamed with that new target name. This information is specific to the inner workings of the COMPASS design team and session and is noted here for posterity in case future work must build upon these studies.

3.0 Baseline Design (Six ASRG)

3.1 Top Level Design (MEL and PEL) for Six ASRG Case

The six ASRG configuration of the REP Centaur orbiting S/C is built on a hexagonal, two platform design with ASRGs attached by twos. The main propulsion system consists of four, long life, Hall thrusters, serially operated, and mounted to the side of the S/C. The bottom deck, to which the Hall thrusters are attached, carries all the electric and chemical propulsion as well as various power conversion systems. The top deck of the S/C, below the antenna, contains all of the avionics and science instruments. The surface area of the top face of the S/C is dominated by the 2.1 m Ka-band antenna. The S/C is designed to optimize ASRG output power and provide science pointing while providing communications.

3.1.1 MEL for Six ASRG Case

The bottoms-up MEL for the final six ASRG mission designed is shown in Table 3.1.

TABLE 3.1.—BASELINE SIX ASRC CASE—TOP LEVEL MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP S/C (Payload and Stage)	-	-	1071.00	11.68	125.14	1196.14
Science Payload	-	-	44.12	30.00	13.24	57.36
REP Bus	-	-	1026.88	10.90	111.91	1138.79
Attitude Determination and Control	-	-	18.40	20.00	3.68	22.08
Command and Data Handling	-	-	33.30	34.26	11.41	44.71
Communications and Tracking	-	-	39.00	34.10	13.30	52.30
Electrical Power Subsystem	-	-	169.82	16.97	28.82	198.64
Thermal Control (Non-Propellant)	-	-	42.81	15.00	6.42	49.23
Propulsion	-	-	107.99	26.56	28.68	136.67
Propellant	-	-	517.57	0.00	0.00	517.57
Structures and Mechanisms	-	-	97.99	20.00	19.60	117.59

3.1.2 Power Equipment List (PEL) for Six ASRG Case

The power listing for nominal loads is modeled using a 900 W max power (with 30 percent margin on all but the EP system) budget. A 30 percent margin is added on all but the REP systems (thruster able to handle lower powers down to 500 W). Table 3.2 lists the concepts of operations and what items are turned on at which point on the mission trajectory.

TABLE 3.2.—PEL PER SUBSYSTEM OVER MISSION PHASES

	Propulsion (W)	Avionics (W)	Comm. (W)	Thermal (W)	GN&C (W)	Power (W)	Science (W)	CBE Total (W)	30% Margin	Total (W)
Launch	0	24	0	33	27	63	0	146	48.63	195
Star 48 Operation	0	24	420	33	27	63	0	566	174.63	741
S/C separation	16	24	420	33	27	63	0	582	174.63	757
S/C checkout	16	24	420	33	36	63	60	652	195.48	847
REP Thrusting	700	24	0	33	29	63	2	850	49.86	900
REP Coast	16	24	0	33	29	63	2	166	49.86	216
Communications	16	53	420	33	29	63	2	615	184.44	799
Flyby	16	53	420	33	29	63	60	673	201.87	875
Centaur Targeting	700	24	0	33	29	63	2	850	49.86	900
Centaur Science	16	53	0	33	29	63	60	253	75.87	329
Centaur Communications	16	53	420	33	29	63	60	673	201.87	875

3.2 System Level Summary

Table 3.3 breaks out the system level summary of the REP S/C designed in this COMPASS session. The bottoms-up masses of the subsystem with the growth estimates applied per line item in the subsystem yielded a total growth on the dry mass of 22.6 percent. Since the desired total growth is 30 percent per COMPASS operating procedure, an additional 7.4 percent of dry mass is carried as system level growth. This mass is “flown” through the mission, and adds to the inert mass that the propulsion system has to navigate with the trajectory ΔV.

The performance of the Atlas 551 launch vehicle to the mission C_3 of 97 km^2/s^2 is 1260 kg. A S/C adapter mass of 10 kg is taken out of that number to give the available launch performance of 1250 kg to the C_3 as reported in Table 3.3. The total wet mass of the S/C of 1237 kg is subtracted from the available launch performance. The remaining 13 kg is the launch margin available. This mass is not flown along the trajectory. In future iterations, the science payload may be increased to get as close to a zero launch margin as possible and still fit in the launch vehicle performance.

TABLE 3.3.—SYSTEM INTEGRATION SUMMARY SHEET: SYSTEM LEVEL GROWTH TRACKING

COMPASS study: Radioisotope Electric Propulsion (REP)					Study date: November 26, 2007
GLIDE container: *REP_Sept2007: Thereus_6ASRG*					
REP Spacecraft MEL rack-up (mass)				COMPASS REP design	
WBS no.	Main subsystems	CBE Mass (kg)	Growth (kg)	Total mass (kg)	Aggregate growth (%)
01	REP Spacecraft (payload and bus)	1071.0	125.1	1196.1	------
01.1	Science Payload	44.1	13.2	57.4	30.0
01.2	*REP Bus*	*1026.9*	*111.9*	*1138.8*	------
01.2.1	AD&C	18.4	3.7	22.1	20.0
01.2.2	C&DH	33.3	11.4	44.7	34.3
01.2.3	Communications and tracking	39.0	13.3	52.3	34.1
01.2.4	Electric power	169.8	28.8	198.6	17.0
01.2.5	Thermal control	42.8	6.4	49.2	15.0
01.2.6	Propulsion	108.0	28.7	136.7	26.6
01.2.7	Propellant	517.6	-----	--------	------
01.2.8	Structures and mechanisms	98.0	19.6	117.6	20.0
	Estimated REP S/C dry mass	553	125	678.6	22.6
	Estimated REP S/C wet mass	1071	125	1196.1	------
	System level growth calculations				Total growth (%)
	Desired system level growth	553	166	719.5	30.0
	Additional growth (carried at system level)	-------	41	--------	7.4
	Total wet mass with growth	1071	166	1237.0	
	Available launch performance to C_3 (kg)			1250.2	
	Launch margin available (kg)			13.2	

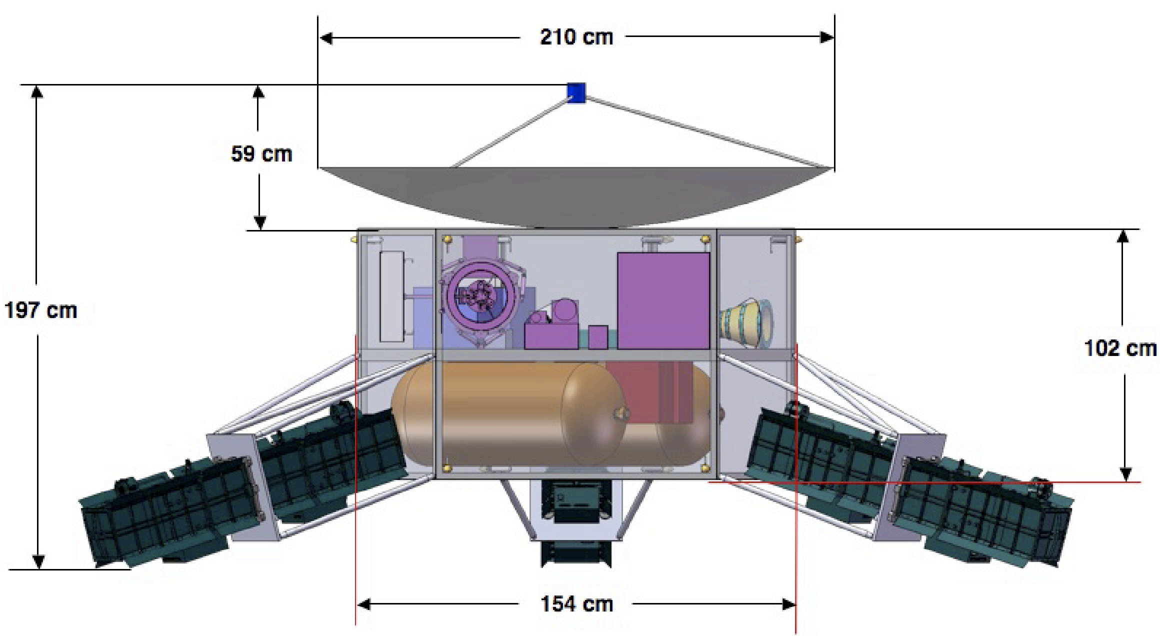

Figure 3.1.—REP Science S/C dimensions.

3.3 Design Concept Drawing and Description

Figure 3.1 shows a side view of the REP Centaur S/C, without the Star 48 engine attached, with dimensions. All dimensions are in metric units.

4.0 Subsystem Breakdown

4.1 Attitude Control System (ACS)

The starting design is borrowed from NH

- ACS hydrazine
- Two Star Trackers (Adcole Corporation) Figure 4.1. These star trackers were the ones used on the NH S/C (Adcole)
- Eight Sun Sensors (EDO Corp Barnes Engineering Division)
- Four Reaction Wheels (Valley Forge Bearcat 5 Nms)
- GN&C software run on main C&DH computers

Table 4.1 lists the items in the ACS MEL for the COMPASS REP S/C design. All growth allowances follow the AIAA MGA schedule in Section 2.4. Note that an IMU from Northrop Grumman was added to the MEL (Figure 4.2).

Figure 4.1.—Adcole Star Tracker.

Figure 4.2.—Northrop Grumman Scalable Inertial Reference Unit (NG SIRU) for space.

TABLE 4.1.—ACS BOTTOMS-UP MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	----	1071.00	11.68	125.14	1196.14
REP Bus	-	----	1026.88	10.90	111.91	1138.79
AD&C	-	----	18.40	20.00	3.68	22.08
GN&C	-	----	18.40	20.00	3.68	22.08
Sun sensors	8	0.01	0.04	20.00	0.01	0.05
Reaction wheels	4	1.27	5.08	20.00	1.02	6.10
Star Trackers	2	3.19	6.38	20.00	1.28	7.66
IMU	1	6.90	6.90	20.00	1.38	8.28

Figure 4.3 shows the avionics deck of the REP S/C. This deck is where all the electronics are located. The graphic has all non-electronics items in the S/C invisible, and highlights the components of Avionics, ACS, Power and Communications, aside from the main antenna, and the science instruments (all labeled at the bottom of the graphic).

4.1.1 ACS Trades

No trades were done on the ACS, except a conceptual trade of whether or not to include reaction wheels.

4.1.2 ACS Analytical Methods

The design was based on current hardware.

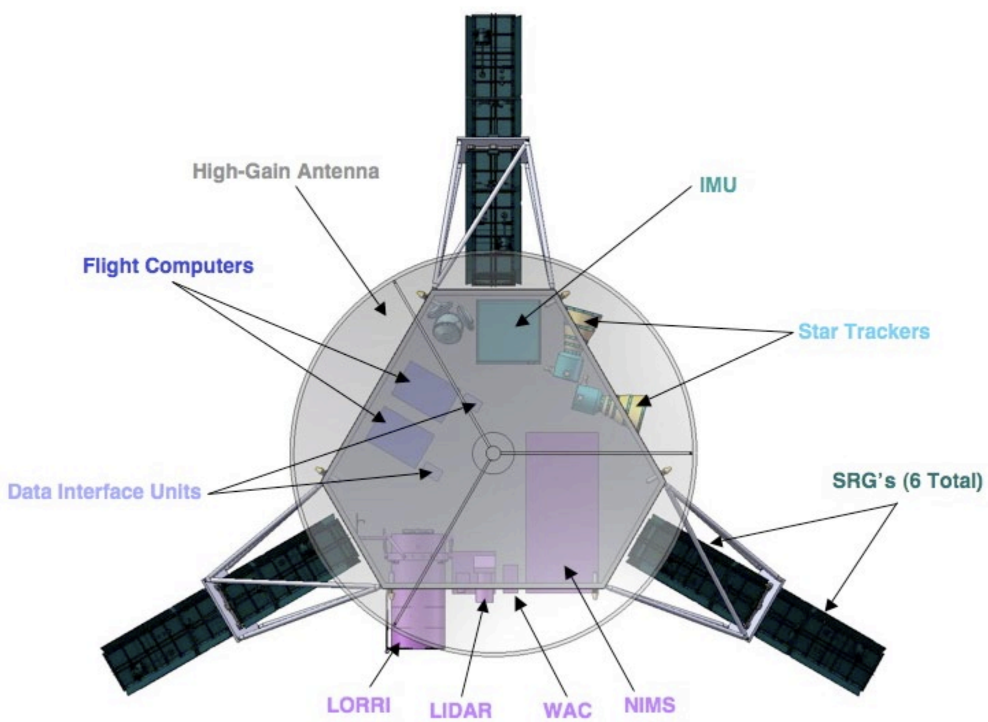

Figure 4.3.—Science and avionics deck of REP S/C.

4.1.3 ACS Recommendation

Analysis into the amount of ΔV necessary for station keeping and attitude control throughout the mission life needs to be performed to determine whether the 50 m/s assumption is sufficient. Additionally, further research is necessary to determine whether the star trackers and sun sensors are capable of operating at the distances of the Centaur bodies at the EOL of the trajectory.

4.2 Communications

4.2.1 Communications Requirements

Provide uplink and downlink capability throughout the primary and/or extended mission. Meet science mission requirements during Thereus orbital operations of 8 hr/day of downlink pointed to Earth with a minimum 6.3 kbps downlink at 70-m dish (or about 147 Mbits/day of downlink including a minimum of 10 percent for housekeeping).

4.2.2 Communications Assumptions

Assume the DSN will be capable of supporting Ka-Band downlink via a 70-m (or equivalent antenna array) antenna by through 2024. The design is based on the NH concept of two onboard, integrated electronics modules (IEMs). The overall harness requirements are reduced if the NH IEM design is implemented.

4.2.3 Communications Design and MEL

- REP orbiter communications subsystem consists of
  - A fixed 2.1-m diameter X/Ka-Band high gain antenna (HGA)
  - Two IEMs, based on the NH, housing many S/C functions, including C&DH, instrument interface circuitry, telemetry interface, solid state recorder, and receiver and exciter sections of the communications subsystem
  - Two 200-W Traveling Wave Tube Amplifier (TWTA) to provide high power radio frequency communications power (RF downlink output)
  - RF switch assembly to interconnect antenna with two TWTAs and the rest of the communications subsystem
  - Cabling
- Ka-Band downlink to 70-m ground stations
  - Ka-Band downlink frequency: 32 GHz
- X-Band support between orbiter and Earth’s 70-m ground stations
  - Forward frequency: 8.4 GHz
  - Return frequency: 7.75 GHz
  - Use of a fixed 2.1-m HGA
  - 200 W RF power
  - TT&C will share the uplink and downlink bandwidth

Table 4.2 lists the items in the Communications system MEL for the COMPASS REP S/C design. All growth allowances follow the AIAA MGA schedule in Section 2.4 and do not contain the additional 8.8 percent carried at the system level.

TABLE 4.2.—REP COMMUNICATIONS SYSTEM MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	----	1071.00	11.68	125.14	1196.14
REP Bus	-	----	1026.88	10.90	111.91	1138.79
Communications and Tracking	-	----	39.00	34.10	13.30	52.30
X/Ka HGA	-	----	27.00	31.48	8.50	35.50
Transmitter/receiver	2	4.00	8.00	30.00	2.40	10.40
Power amp	2	3.00	6.00	30.00	1.80	7.80
Switch unit	0	0.00	0.00	0.00	0.00	0.00
Antenna	1	9.00	9.00	30.00	2.70	11.70
Band pass filter	0	0.00	0.00	0.00	0.00	0.00
Band reject filter	0	0.00	0.00	0.00	0.00	0.00
Sensor	0	0.00	0.00	0.00	0.00	0.00
Cabling	2	2.00	4.00	40.00	1.60	5.60
Diplexer	0	0.00	0.00	0.00	0.00	0.00
Coupler	0	0.00	0.00	0.00	0.00	0.00
Miscellaneous no. 1	0	0.00	0.00	0.00	0.00	0.00
Miscellaneous no. 2	0	0.00	0.00	0.00	0.00	0.00
Ka-band antenna	-	----	0.00	0.00	0.00	0.00
Transponder	0	0.00	0.00	0.00	0.00	0.00
RF assembly	0	0.00	0.00	0.00	0.00	0.00
Processing module	0	0.00	0.00	0.00	0.00	0.00
Antenna	0	0.00	0.00	0.00	0.00	0.00
Communications instrumentation	-	----	12.00	40.00	4.80	16.80
Coaxial cable	2	6.00	12.00	40.00	4.80	16.80
Installation—mounting and circuitry	0	0.00	0.00	0.00	0.00	0.00

4.2.4 Communications Trades

Further analysis on optimization of the communications components was not performed.

4.2.5 Communications Analytical Methods

The link budgets provides for values of RF transmit powers at 40 and 200 W and antenna gains for Ka-Band and X-Band downlinks. Link margins of 3 dB or better exist for all links.

4.2.6 Communications Risk Inputs

Heritage of success of design and components assumed.

4.2.7 Communications Recommendation

In the future, further analysis should be done to consider the communications system used in the NH mission, as its performance behavior is characterized during flight mission ops. Figure 4.4 shows a detailed block diagram of the NH full S/C design. This includes C&DH as well as communications.

4.3 Command and Data Handling (C&DH)—Avionics

4.3.1 Avionics Requirements

The design requirements, from the science payload and the REP Bus, for the C&DH system were as follows

- Storage for 7 days of data or fly-by (TBD, est. 8 to 16 GB)
- Avionics for systems command, control, and health management
- Payload control will be done by the C&DH system

- Single fault tolerant avionics

4.3.2 Avionics Assumptions

- All electronics are ≥65 Krad avionics
- Cabling mass is estimated as 50 percent of the avionics hardware
- Avionics spares are cold spares to minimize power consumption
- NH S/C was used as the starting point for the avionics hardware design

4.3.3 Avionics Design and MEL

All avionics components used in the design are based on commercially available components from British Aerospace (BAE) and SEAKR Engineering, Inc. (SEAKR). There are two independent avionics boxes to provide for single fault tolerance. Each avionics box contains a GN&C/C&DH RAD6000 processor, 256 MB GN&C solid-state memory card, solid state recorder (SSR) card, a communications interface card, and a payload interface card. The 1553 processor is used for communications between the GN&C processor and hardware, i.e., Star trackers, IMUs, etc. The GN&C and C&DH computers communicate via the 1553 bus. This new REP avionics design, as shown below, is based on the NH S/C. With the exception of GN&C and C&DH, the processors for each IEM are combined into one, using either RAD6000 or RAD750.

Table 4.3 lists the components used in the COMPASS C&DH MEL design. These are the inputs from the subsystem lead. All growth allowances follow the AIAA MGA schedule in Section 2.4 and do not contain the additional 8.8 percent carried at the system level.

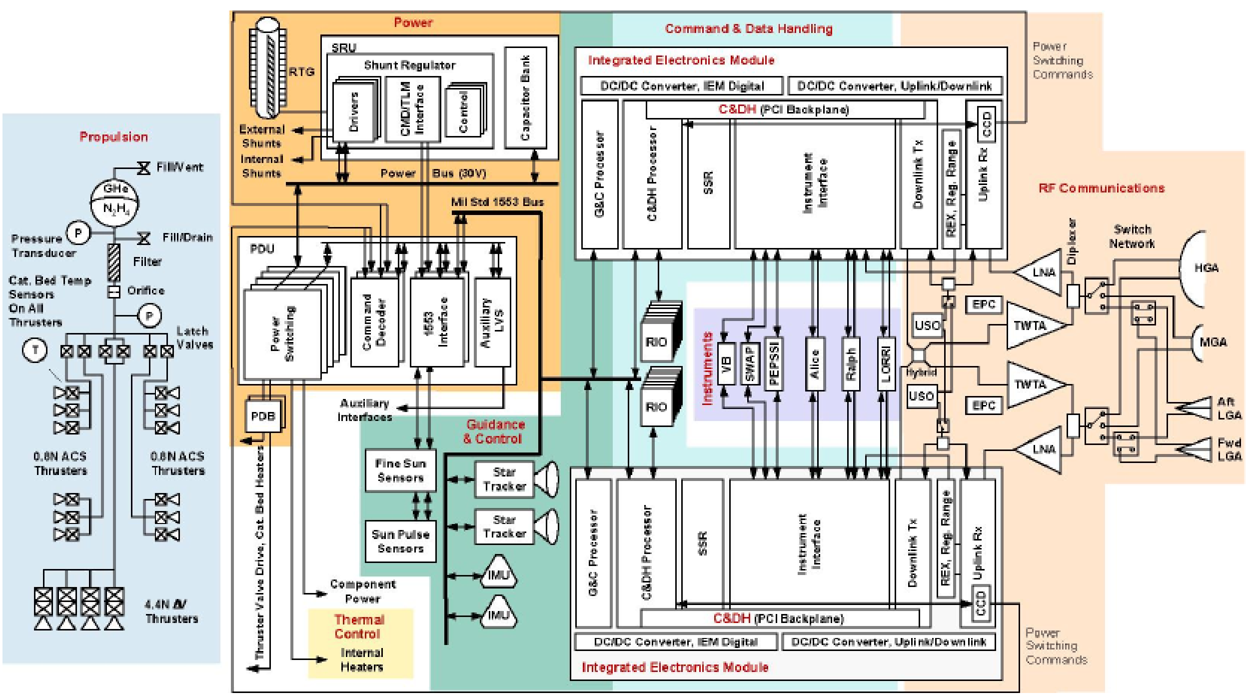

Figure 4.4.—NH full S/C design.

TABLE 4.3.—AVIONICS MEL MASS DETAILS

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	------	1071.00	11.68	125.14	1196.14
REP Bus	-	------	1026.88	10.90	111.91	1138.79
C&DH	-	------	33.30	34.26	11.41	44.71
Flight computer	2	8.00	16.00	25.00	4.00	20.00
Command and telemetry computer	0	0.00	0.00	0.00	0.00	0.00
Data interface unit	2	2.00	4.00	30.00	1.20	5.20
Data bus operations amplifier	0	0.00	0.00	0.00	0.00	0.00
Operations recorder	2	1.10	2.20	30.00	0.66	2.86
Command and control harness (data)	1	11.10	11.10	50.00	5.55	16.65
Instrumentation and wiring	-	------	0.00	0.00	0.00	0.00
Operational Instrumentation, sensors	0	0.00	0.00	0.00	0.00	0.00
Data cabling	0	0.00	0.00	0.00	0.00	0.00

4.3.4 Avionics Trades

Off-the-shelf (OTS) components used in the design. Further analysis of the choice of components to needs be done.

4.3.5 Avionics Analytical Methods

Much of the design is based on the NH S/C.

4.3.6 Avionics Risk Inputs

Assuming heritage on components.

4.3.7 Avionics Concerns, Comments, Recommendations

- No ultra-stable oscillator (USO)/Atomic Clock included in avionics hardware. Should it be included in communications system?
- Processing power of the RAD6000 is assumed to be adequate for GN&C, C&DH, and science payload.
- Storage requirements are driven by fly-by storage needs (which are still being estimated).
- Only one SSR would be active at a time and thus susceptible to single event upsets (SEU).
- Total radiation dose is a concern with all deep space missions. This preliminary design has attempted to use only hardware which has already been proven in a deep space mission to assure the life of the electronics over the 12-yr mission.

4.4 Electrical Power System

4.4.1 Power Requirements

Six ASRG’s (12 general purpose heat source (GPHS)) are designed to provide 960 W to power the REP S/C at beginning of life (BOL). The system is designed to provide 900 W to the REP S/C at EOL (10-yr). There are negligible thermal interactions between the ASRG’s. Figure 4.5 shows a typical ASRG with the main components called out in the graphic. The six ASRGs are connected together with via a Shunt Regulator/Bus Protection (RBI) assembly. This RBI isolates the ASRG’s from S/C bus and each other and follows load demands from S/C bus. There is an approximately 6 percent loss through the RBI and monitoring circuitry (94 percent of power flows through to loads) with 53 W used for fault detection/ monitoring. Included in this system is a bus capacitance of 3000 μf which provides some bus rigidity. Power cabling and harness systems design assumes a 1 percent line loss.

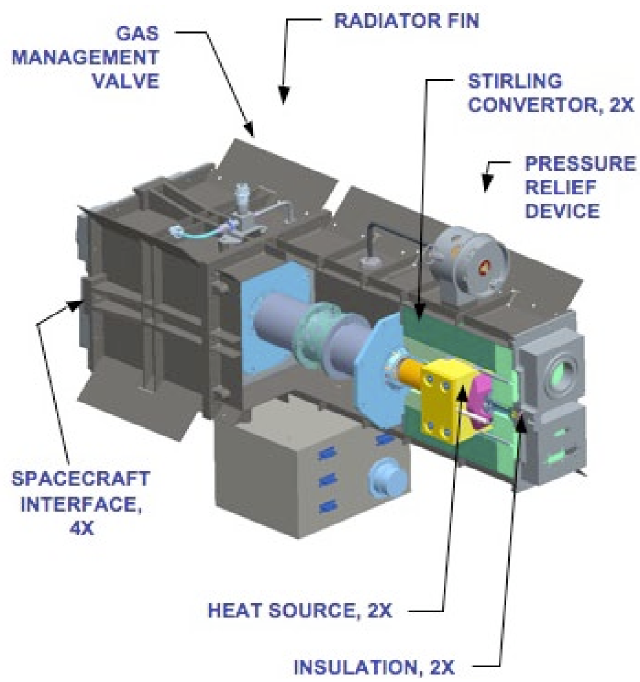

Figure 4.5.—ASRG computer aided design (CAD) model.

4.4.2 Power Assumptions

This baseline power system design consisted of six ASRG for the generation of power.

4.4.3 Power Design and MEL

Minimize power for non-propulsion during EP operation (minimize plutonium needed). Specific performance details on each ASRG unit are as follows

- Power: 160 W at 28 ± 0.2 V BOL
  150 W at 28 ± 0.2 V EOL (10 yr)
- End of mission (EOM) (Deep Space (14 yr)) -126 W
- Mass: 19.5 kg with mounting isolator plate
- Envelope: 30.5-cm W, 46-cm H, 76-cm L (12-in. W, 18-in. H, 30-in. L)
- Specific Power: 8 We/kg

ASRG Design Attributes

- Two Stirling converters
  - Co-axially aligned for dynamic balance
  - One GPHS module per converter
- Integrated, single-fault tolerant controller
- Autonomous operation and fault isolation from S/C
- S/C disturbance torque requirement < 35 N-m
  - Based on 1000 kg, 1-m cube S/C with 5-µrad pointing accuracy and a safety factor of 5

Table 4.4 lists the items in the power system MEL for the COMPASS REP S/C design. All growth allowances follow the AIAA MGA schedule in Section 2.4. Figure 4.6 shows the power and propulsion deck of the REP S/C. The ASRGs are mounted to the bus structure via trusses, at a 120° angle between sets of pairs around the perimeter of the main bus.

TABLE 4.4.—POWER SYSTEM MEL FOR REP BUS

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	-------	1071.00	11.68	125.14	1196.14
REP Bus	-	-------	1026.88	10.90	111.91	1138.79
Electrical Power Subsystem	-	-------	169.82	16.97	28.82	198.64
Radioisotope Power System	-	-------	116.82	10.00	11.68	128.50
RPS main system	6	19.47	116.82	10.00	11.68	128.50
Miscellaneous no. 2	0	0.00	0.00	0.00	0.00	0.00
Power management and distribution	-	-------	26.75	15.00	4.01	30.76
Power management/control electronics	0	0.00	0.00	0.00	0.00	0.00
Power distribution/monitoring wiring harness	0	0.00	0.00	0.00	0.00	0.00
DC Switchgear/Shunt Regulator	1	26.75	26.75	15.00	4.01	30.76
Miscellaneous no. 2	0	0.00	0.00	0.00	0.00	0.00
Power cable and harness subsystem	-	-------	26.25	50.00	13.13	39.38
Spacecraft bus harness	1	5.25	5.25	50.00	2.63	7.88
PMAD Harness	1	5.25	5.25	50.00	2.63	7.88
Electric propulsion harness	1	5.25	5.25	50.00	2.63	7.88
RPS to S/C harness	1	5.25	5.25	50.00	2.63	7.88
Power cabling	1	5.25	5.25	50.00	2.63	7.88

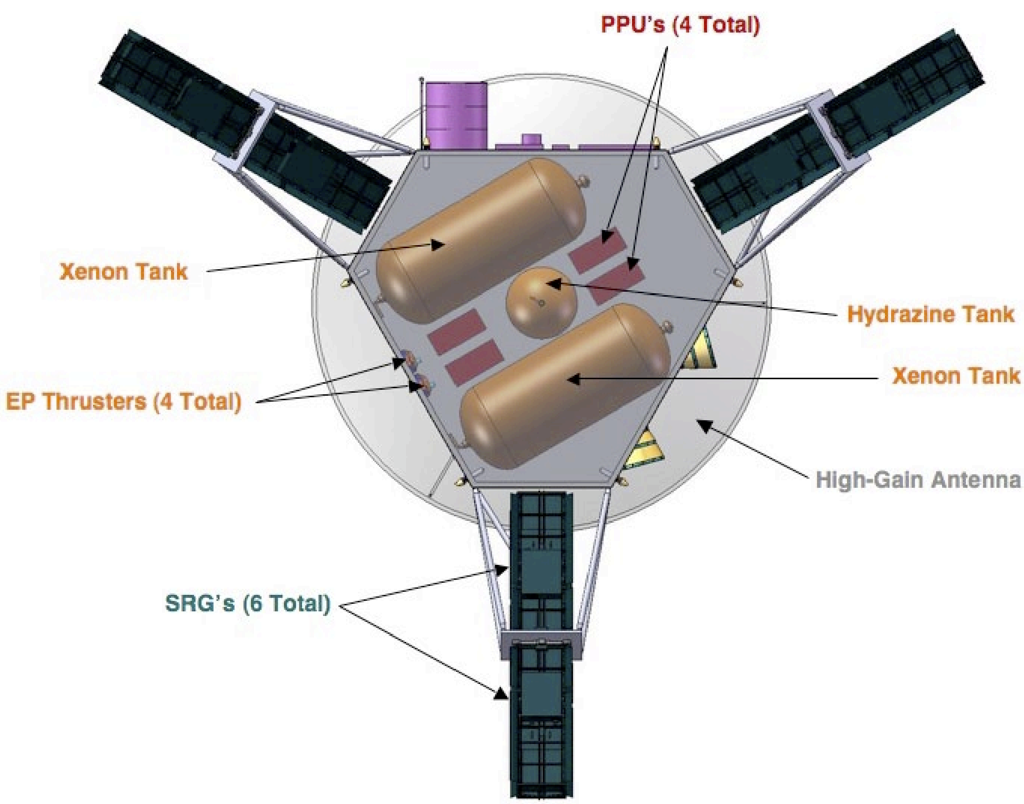

Figure 4.6.—Power and propulsion system deck.

4.4.4 Power Trades

For the power system, two other power system design options were considered but rejected

Option 1 (Not used since more power was needed than the 10/1 power ratio allowable on the dual-windings allowed. Significant advantages to the mission exist if the full ASRG power is available during non-thrusting periods)

- Direct drive the Hall thrusters
- Use dual wound alternator (providing 600 V and 28 V, 100 Hz AC)
- 10/1 power ratio on dual alternators
- The 600 V AC converts to 400 V DC
- Power to Thrusters EOM 646 W
- Power to Payload EOM 76 W

Option 2 (not used due to the heavy converter needed.)
- Each ASRG provides 28 V DC as designed
- DC/DC conversion to 400 V DC for hall thruster
- The current mass estimate of single 600 W DC/DC converter at 30 kg
- Eight ASRG provides (1120 W BOL, 1040 W EOL) 750 W power into the thruster with excess 14 W EOL
- Loss of a single SRG would then provide ~650 W into Thruster
- If the spacing of the ASRGs is changed to a 90°: Does this provide better viewing for radiative coupling between the ASRGs? How does this affect structure and loads balancing?

Table 4.5 lists the impact of trade in the number of ASRGs and total power available, as well as excess power to be radiated as mentioned in the Option 2 analysis up above.

TABLE 4.5.—TRADE ANALYSIS OF VARYING NUMBER OF SRGS

Number SRGs	4	5	6	7	8
Power (EOL, 10 yr)	130	130	130	130	130
Total power EOL (W)	520	650	780	910	1040
Into thruster (W)	250	400	500	650	750
PPU, line loss	25	40	50	65	75
Housekeeping (cruise only)	155	155	155	155	155
Housekeep margin (30%)	46.5	46.5	46.5	46.5	46.5
Excess	44	9	29	–7	14

4.4.5 Power Analytical Methods

ASRGs based on current hardware designs.

4.4.6 Power Recommendations for Future Analysis

- Would a change to a 90° separation between the ASRGs provide a better or worse radiative coupling between the ASRGs?
- Plate/strut support in ‘middle’
  - Vibration/thermal leak to science payload instruments
- Science Payload view of SRGs, is a 90° or 120° (current design) separation better? Should the ASRG's be offset above or below the flight instrument deck?

Is there necessary plutonium availability?

4.5 Structures and Mechanisms

The following section describes the design structures and mechanisms methodology and details.

4.5.1 Structures and Mechanisms Requirements

The REP S/C structure must contain necessary hardware for research instrumentation, avionics, communications, propulsion and power. It must be able to withstand applied loads from launch vehicle and provide minimum deflections, sufficient stiffness, and vibration damping. The goal of the design is to minimize weight of the components that make up the structure of the S/C bus, and must fit within confines of launch vehicle.

4.5.2 Structures and Mechanisms Assumptions

Because of the use of the Star 48 engine, the structure must be able to withstand an axial acceleration load of up to 6g maximum. The launch vehicle also imparts a maximum of 3.5g lateral acceleration. The structure must also accommodate a Conical Star 48 adapter. The Hall thrusters must thrust through the center of mass (COM) and need to be canted. The structural design and S/C layout must accommodate a thruster gimbal angle.

The basic assumptions made in the design process of the S/C bus structure were

- Material: Al alloy 2090-T3
- Space frame with tubular members
- Composite sandwich structure shelf assumed to be 100 percent Al composition using Al 2090-T3 face sheets and an Al honeycomb core with the trade name, Alcore Higrid.
- Welded and threaded fastener assembly

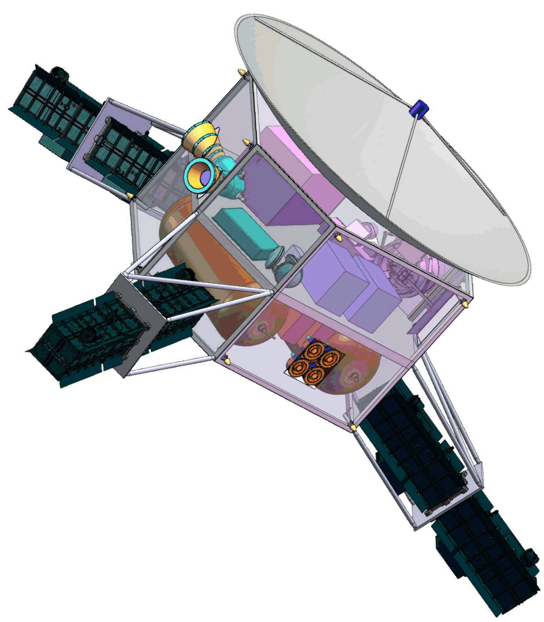

Figure 4.7.—REP Main Bus.

4.5.3 Structures and Mechanisms Design and MEL

The basic structural design of the REP S/C shown in Figure 4.7 consists of

- Tubular space frame in a hexagonal configuration
- Deck consists of a composite sandwich architecture with an Al 25 mm thick low density honeycomb core and 1.5 mm thick Al face sheets to mount hardware
- Thin sheets utilized as sheer panels and to enclose the structure
- Struts are used to support the paired ASRGs externally. The ASRGs are mounted to plates with vibration isolators.

All growth allowances follow the AIAA MGA schedule in Section 2.4.

The initial assumptions used in the design were: 2600 kg, 6g axial loading, 3.5g lateral loading (not concurrent with max axial loading), Al 2090-T3 and a 1.4 safety factor. The maximum stress in axial members was set at 58 MPa. Initial assumptions for lateral load capacity were based on S/C heritage design. The Secondary structural mass (called installations in the MEL above in Table 4.6) assumes a 4 percent factor of the mounted hardware current best estimate (CBE) mass.

TABLE 4.6.—REP SIX ASRG S/C STRUCTURES AND MECHANISMS BOTTOMS-UP MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	------	1071.00	11.68	125.14	1196.14
REP Bus	-	------	1026.88	10.90	111.91	1138.79
Structures and Mechanisms	-	------	97.99	20.00	19.60	117.59
Structures	-	------	78.41	20.00	15.68	94.09
Primary structures	-	------	40.95	20.00	8.19	49.15
Main bus structure	1	40.95	40.95	20.00	8.19	49.15
Secondary structures	-	------	37.45	20.00	7.49	44.94
Balance mass	0	0.00	0.00	0.00	0.00	0.00
Tank supports and bracketry	1	18.18	18.18	20.00	3.64	21.82
SRG support structure	3	5.51	16.54	20.00	3.31	19.85
SRG vibration isolation hardware	3	0.91	2.73	20.00	0.55	3.27
Mechanisms	-	------	19.58	20.00	3.92	23.50
Adaptors and separation	-	------	4.72	20.00	0.94	5.67
Spacecraft adapter	1	4.72	4.72	20.00	0.94	5.67
Separation mechanism (pyros)	0	0.00	0.00	0.00	0.00	0.00
Miscellaneous no. 1	0	0.00	0.00	0.00	0.00	0.00
Installations	-	------	14.86	20.00	2.97	17.83
Science payload installation	1	1.76	1.76	20.00	0.35	2.12
C&DH installation	1	1.33	1.33	20.00	0.27	1.60
Communications and tracking installation	1	1.56	1.56	20.00	0.31	1.87
GN&C installation	1	0.74	0.74	20.00	0.15	0.88
Electrical power installation	0	0.00	0.00	0.00	0.00	0.00
Thermal control installation	1	3.03	3.03	20.00	0.61	3.63
Electric Propulsion installation	1	6.44	6.44	20.00	1.29	7.73

4.5.4 Structures and Mechanisms Trades

Trades were conducted on the use of composite for the main bus compartment structure. The frame of the main bus is sized to accommodate volume and space requirements for antenna, ASRGs, and instrumentation while fitting within confines of the launch vehicle.

4.5.5 Structures and Mechanisms Analytical Methods

Preliminary structural analysis and modeling was performed using the given launch loads and dimensions of the desired S/C bus. An additional installations mass was held for each subsystem in the mechanisms section of the structures system. These installations were modeled using 4 percent of the CBE dry mass of each of the subsystems. No growth margin was applied to that installation mass.

4.5.6 Structures and Mechanisms Risk Inputs

Risk analysis is still to be performed.

4.5.7 Structures and Mechanisms Recommendation

A more detailed structural analysis for loads and vibrations using a modeling tool, i.e., finite element analysis (FEA) would be beneficial in further modeling the structure with assurance of sustaining launch loads. Analysis is needed to look into using the current shelf face sheets, outer sheets, and/or support struts substituted with graphite/polymer composites for further weight savings but possibly at increased cost.

4.6 Propulsion and Propellant Management

4.6.1 Propulsion and Propellant Management Requirements

The REP Centaur mission is a relatively long life mission. The goal of each subsystem and the propulsion system is to minimize the overall mass of the REP bus in order to deliver the maximum amount of payload.

The mission requirements that impact the choice of EP system components are as follows

- Approximately 10 yr mission (~ 600 kg throughput)
- 500 to 1000 W total power
- Essentially constant power from ASRG power supply
- Preliminary optimum I_{sp} ~ 2050 s

4.6.2 Propulsion and Propellant Management Assumptions

The baseline system redundancy assumption was based on single string units. In other words, a propulsion system unit consists of a string of thruster, PPU, gimbal, and Propellant Management System (PMS). Spares or redundant units are assumed to consist of all of the above subsystems. It is important to note that the mission is already modeled with a 90 percent duty cycle. So, 10 percent of the time, the S/C is coasting along its trajectory.

The baseline propulsion system design consists of the following items

- One active 750 W long life Hall engine with two extra engines for lifetime issues and one cold spare
- Four PPU: no cross-strap
- Two-axis range of motion: TBD
- 5.0 percent Xe navigation allowance, 3.6 percent Xe residuals
- PSI cylindrical COPV Xe tanks
- OTS hydrazine system with NH heritage
- Instruments to be fully operational with 900 W ASRG power supply and thrusters full on.

Hall thruster and PPU performance and masses were based on published or in-house calculations by the GRC RPP branch. Thruster performance over a range of I_{sp} was examined as a series of custom designs, rather than a single thruster design throttled over a range of I_{sp}. Thruster mass was assumed to be

50 percent greater than a commercial thruster (SPT-70) which operates at a similar power level. PPU performance and mass were based on a single module of a PPU unit under development and test at GRC.

4.6.3 Propulsion and Propellant Management Design Trades

The trades considered in designing the two major propulsion systems on the S/C (main and RCS) are as follows

- ACS hydrazine
  - OTS blow-down similar to NH
  - Single tank with ~20 kg hydrazine

The possible main EP system options to be considered for this design are

- New Advanced Technology Small Hall Thruster (Figure 4.8)
  - Based on ongoing High Voltage Hall Accelerator (HiVHAC) program at GRC
  - Optimized design to allow up to 2000 s I_{sp} at powers below 1 kW
  - Allows long life needed for mission (Figure 4.9)
- Derated HiVHAC (Figure 4.10)
  - Maximum I_{sp} at 1 kW ~1570 s
  - Performance inadequate for range of REP missions
- Commercial-off-the-shelf (COTS)
  - SPT-70/BPT-600
  - 600 W, ~ 1500 s
  - Limited life/throughput (35 to 50 kg)
- Low power (20 cm) Ion

The Advanced Hall thruster option was initially chosen both for its potential for direct drive operation (see power system discussion), and for its superior performance in terms of efficiency (or equivalently, thrust-to-power) at the low power levels characteristic of REP. The 20 cm ion thruster projected performance was inferior to that of the Hall below 1 kW and at 2000 s or less I_{sp}. The commercial Hall thrusters increased system mass and complexity through the increased number of propulsion strings (13 or more) needed to meet lifetime and redundancy requirements.

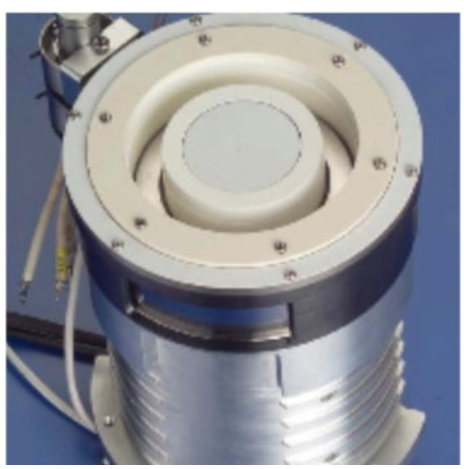

Figure 4.8.—Advanced Technology Small Hall Thruster.

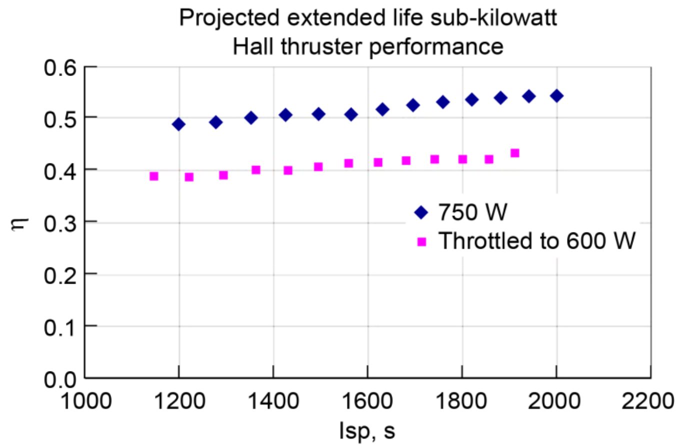

Figure 4.9.—Assumed performance of Small Hall Thruster.

Figure 4.10.—HiVHAC Hall Electric Thruster.

The possible EP thruster system options, once an EP thruster type has been chosen, to be considered are

- Hall
  - Standard PPU
  - "Direct drive" from Stirling alternator

Because of limitations in the ASRG alternator design, the "direct drive" option was discarded and a standard PPU option was selected.

4.6.4 Propulsion and Propellant Management Design and MEL

Figure 4.10 is a graphic of the Advanced Technology Small Hall Thruster being considered in this conceptual design.

4.6.4.1 Main EP System (Xe)

The main EP system is comprised of

- Four extended life, high I_{sp} Hall Thrusters (three operating, one active spare)
  - Thruster performance
    - 30,000 hr life, 300 to 700 V
- Standard PPU
- Two cylindrical, COPV high pressure (2800 psi) Xe tanks

- Propellant distribution system: Single string PMS to each thruster from balanced tank feed
- Thermal details of prop system
  - Number of heaters on tanks, etc.
- Total propellant
  - 540 kg used
  - 8.6 percent residual + margin

Figure 4.11 is a schematic of the EP system and propellant management tankage, etc. The main EP subsystem is comprised of: four HiVHAC Hall Thrusters—three operating, one spare, gimbals on each thruster was used for thrust vector control, and two COPV titanium (Ti)-lined high-pressure cylindrical storage tanks for the Xe propellant (nominal) storage. The Xe distribution system was based on newly developed pressure and flow control units and four PPU for delivering power to each ion thruster.

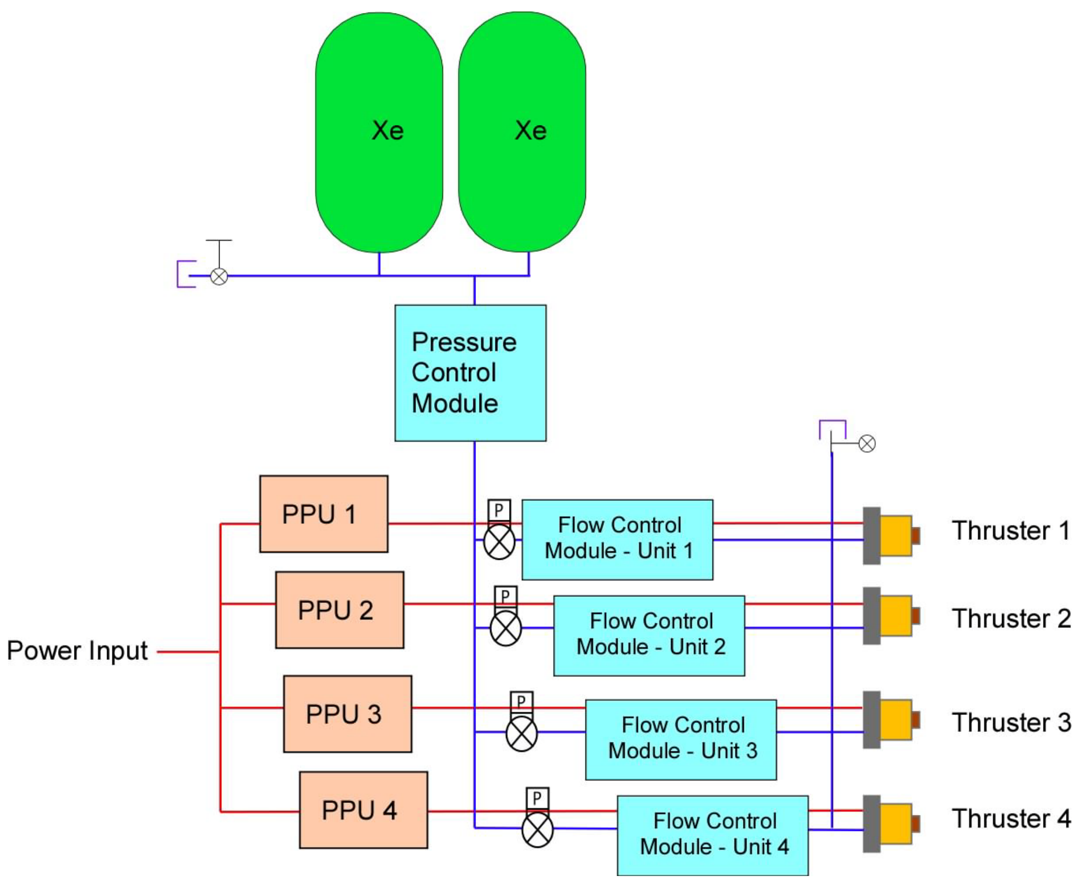

Figure 4.11.—EP system schematic.

4.6.4.2 Secondary RCS System (Hydrazine)

The attitude-reaction control propulsion subsystem was comprised of: eight 0.25 lbf monoprop thrusters placed around the S/C body The Rocket Research MR-103H monomethyl hydrazine and nitrogen tetroxide bipropellant system (MMH/NTO) thrusters were used. Fuel was stored in an Al-Li Metallic Tank Single spherical tank using a blow down pressurization with discrete He pressurization system (Cassini heritage). The propellant distribution system used a design similar to systems developed for the Constellation program, including fault tolerance configuration. Multiple tank and line heaters were included in the mass model to prevent propellants from freezing. Additionally, insulation was included for the same elements. The Instrumentation included was a nominal suite of temperature and pressure sensors. Figure 4.12 shows the notional setup from hydrazine tank to the hydrazine thrusters and illustrated the feed system linkages between the two.

Table 4.7 lists the propulsion system hardware MEL. All growth allowances follow the AIAA MGA schedule in Section 2.4.

Table 4.8 lists the propellant used in this mission. Note, the margins and residuals are called out as separate line items in this mass listing, and no additional weight growth schedule (WGS) is necessary on the propellants.

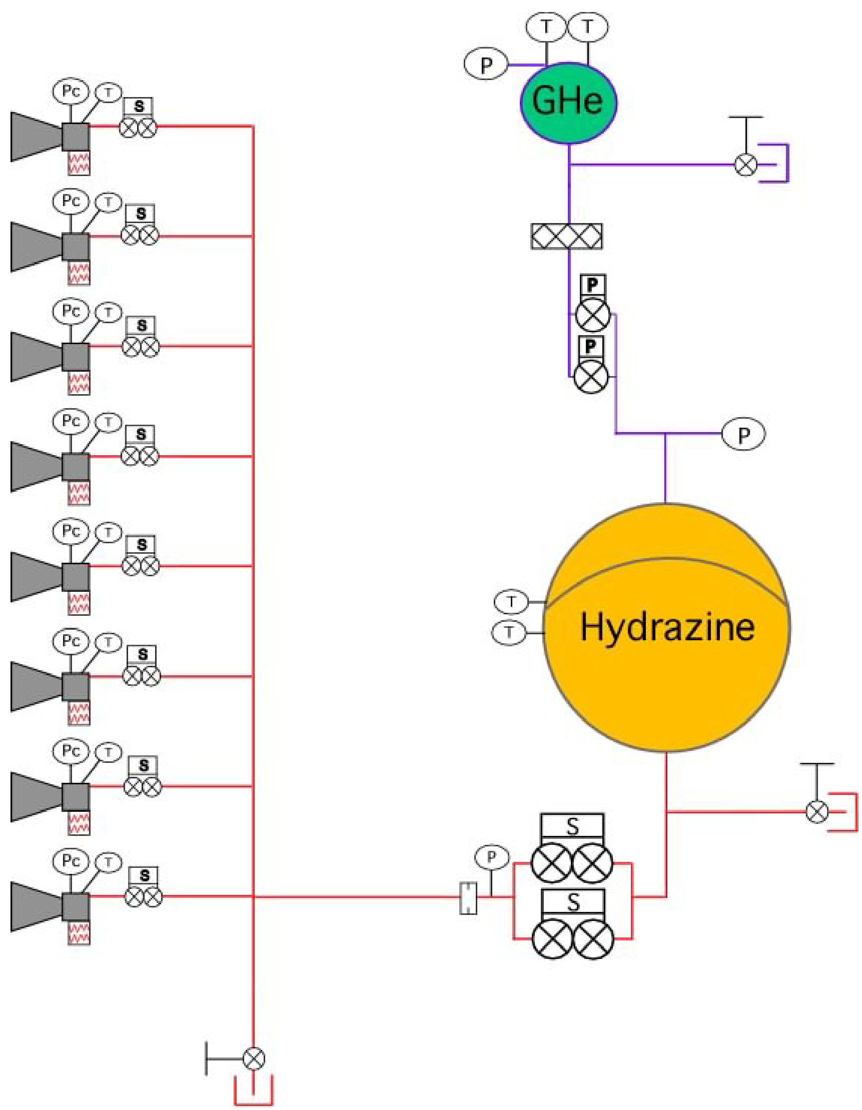

Figure 4.12.—RCS system schematic.

TABLE 4.7.—ELECTRIC AND CHEMICAL PROPULSION SYSTEM MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	-------	1071.00	11.68	125.14	1196.14
REP Bus	-	-------	1026.88	10.90	111.91	1138.79
Propulsion	-	-------	107.99	26.56	28.68	136.67
Primary EP system	-	-------	9.00	12.00	1.08	10.08
Primary EP thrusters	4	2.25	9.00	12.00	1.08	10.08
EPS power processing and control	0	0.00	0.00	0.00	0.00	0.00
EPS structure	-	-------	0.00	0.00	0.00	0.00
EP thruster pod	0	0.00	0.00	0.00	0.00	0.00
EP thruster boom	0	0.00	0.00	0.00	0.00	0.00
Miscellaneous no. 1	0	0.00	0.00	0.00	0.00	0.00
EPS thermal control subsystem	-	-------	0.00	0.00	0.00	0.00
EPS MLI	0	0.00	0.00	0.00	0.00	0.00
EPS heaters and sensors	0	0.00	0.00	0.00	0.00	0.00
Miscellaneous no. 1	0	0.00	0.00	0.00	0.00	0.00
Propellant management	-	-------	66.29	31.18	20.67	86.96
Xe propellant tank(s)	2	27.39	54.77	30.00	16.43	71.21
High pressure feed system	1	7.62	7.62	30.00	2.29	9.90
Low pressure feed system	0	0.00	0.00	0.00	0.00	0.00
Residual Xe propellant (nondeterministic)	0	0.00	0.00	0.00	0.00	0.00
Temperature sensors	1	3.90	3.90	50.00	1.95	5.85
PPU	-	-------	16.00	12.00	1.92	17.92
PPU mass	4	4.00	16.00	12.00	1.92	17.92
Cabling	0	0.00	0.00	0.00	0.00	0.00
RCS	-	-------	16.70	30.00	5.01	21.71
RCS tank subassembly	1	2.79	2.79	30.00	0.84	3.62
RCS propellant management subassembly	1	9.45	9.45	30.00	2.83	12.28
RCS thruster subassembly	2	2.23	4.46	30.00	1.34	5.80

TABLE 4.8.—PROPELLANT MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	-	--------	1071.00	11.68	125.14	1196.14
REP Bus	-	--------	1026.88	10.90	111.91	1138.79
Propellant	-	--------	517.57	0.00	0.00	517.57
Primary EP propellant	-	--------	489.53	0.00	0.00	489.53
Primary EP propellant used	1	450.76	450.76	0.00	0.00	450.76
Primary EP propellant residuals (unused)	1	16.23	16.23	0.00	0.00	16.23
Primary EP propellant performance margin (unused)	1	22.54	22.54	0.00	0.00	22.54
RCS propellant	-	--------	27.94	0.00	0.00	27.94
RCS used	1	27.26	27.26	0.00	0.00	27.26
RCS residuals	1	0.68	0.68	0.00	0.00	0.68
Pressurant	1	0.10	0.10	0.00	0.00	0.10

4.6.5 Propulsion and Propellant Management Risk Inputs

The EP system is expected to have a performance not previously demonstrated at the power levels of interest. Because the chosen trajectory does not have large coast periods, the mission performance would not be achievable without reaching the thruster performance goals. Also, the mission is not very “robust” in that missed thrust periods could place the mission at risk. Mitigation would include imposing coast period requirements or mandating a propulsion system with a less than optimal I_{sp}, therefore higher thrust; to increase the capability to makeup mission thrust periods. Effect of thrust on coast periods is shown in the initial analysis done for the Bienor mission in Figure D.4.

A development schedule risk lies in the choice of an undeveloped thruster technology. Because the projected thruster performance and life have not been demonstrated, additional schedule and cost uncertainty are introduced. The assigned levels of risk for the various consequence categories are

- Cost: 4
- Schedule: 4
- Performance: 4
- Safety: 1
- Risk mitigation can be achieved by appropriate scheduling assumptions and by incorporating adequate preliminary testing into the development program.

4.6.6 Propulsion and Propellant Management Recommendation

Future trades to reduce mass on the main propulsion system are as follows

- Lower power ion thruster (8 cm) for possible mass benefit or mission benefit
- Re-optimize mission for lower I_{sp} or higher power to capture HiVHAC or COTS regime

Further trades on the secondary propulsion system to reduce mass are

- Utilize primary propulsion for some maneuvering, and modeling the trade between attitude control using only wheels (control moment gyros) versus propulsion or a combination of the two.

4.7 Thermal Control

4.7.1 Thermal Requirements

The thermal requirements for the REP Centaur mission were to provide a means of cooling the S/C during operation as well as provide heat to vital components and systems to maintain a minimum temperature throughout the mission. The goal of the thermal control system is to provide for the rejection of heat and maintain a safe operating environment for the electronics and other systems on the S/C.

The maximum heat load to be rejected by the thermal system was 125 W from the electronics, and the desired operating temperature for the electronics and propellant was 300 K. The ASRGs have dedicated built in thermal control systems and therefore were not part of the S/C thermal system.

4.7.2 Thermal Assumptions

The thermal subsystem used in the COMPASS SEP Enceladus design (CD–2007–09) was used as a starting point for these analyses. The system was modeled for Deep Space Operation. The radiator always sees deep space with a small (0.05) view factor to the Sun.

The assumptions utilized in the analysis and sizing of the thermal system were based on the operational environment. It was assumed that the worst case operational conditions would be in near Earth space. The following assumptions were utilized to size the thermal system.

- The view factors for the radiator to the Earth, lunar surface and ASRG radiators were assumed to be 0.1, 0.25 and 0.1, respectively
- The maximum angle of the radiator to the Sun was 15°
- The radiator temperature was 320 K

4.7.3 Thermal Design and MEL

The thermal system is used to remove excess heat from the electronics and other components of the system as well as provide heating to thermally sensitive components during periods of inactivity.

Excess heat is collected from a series of Al cold plates located throughout the interior of the S/C. These cold plates have heat pipes integrated into them. The heat pipes transfer heat from the cold plates to the radiator, which radiates the excess heat to space. The portions of the heat pipes that extend from the S/C body and are integrated to the radiator are protected with a micrometeor shield. The radiator has exterior louvers on it to provide some control over its heat transfer capability.

The radiator was sized with approximately 50 percent margin in its heat rejection area. This added margin insures against unforeseen heat loads, degradation of the radiator and increased view factor toward the sun or other thermally hot body not accounted for in the analysis.

To provide internal heating for the electronics and propulsion systems a series of electric heaters are utilized. These heaters are controlled by an electronics controller, which reads a series of thermocouples through a data acquisition system.

MLI is also utilized on the S/C, and propellant system to regulate and maintain the desired temperatures.

Table 4.9 lists the items in the thermal system MEL for the COMPASS REP S/C design. All growth allowances follow the AIAA MGA schedule in Section 2.4 and do not contain the additional 8.8 percent carried at the system level. Systems modeled: Micrometeor Shielding on Radiator, Radiator Panels, Thermal Control of Propellant Lines and Tanks, S/C Insulation, Avionics and Power Management and Distribution (PMAD) Cooling.

TABLE 4.9.—THERMAL SYSTEM MEL

Description REP Centaur Mission Six ASRG (October 11, 2007)	Qty	Unit mass (kg)	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	---	-----	1071.00	11.68	125.14	1196.14
REP Bus	---	-----	1026.88	10.90	111.91	1138.79
Thermal control (nonpropellant)	---	-----	42.81	15.00	6.42	49.23
Active thermal control	---	-----	16.90	15.00	2.54	19.44
Heaters	15	1.00	15.00	15.00	2.25	17.25
Thermal control/heaters circuit	2	0.20	0.40	15.00	0.06	0.46
Data acquisition	1	1.00	1.00	15.00	0.15	1.15
Thermocouples	50	0.01	0.50	15.00	0.08	0.58
Miscellaneous no. 1	0	0.00	0.00	15.00	0.00	0.00
Miscellaneous no. 2	0	0.00	0.00	15.00	0.00	0.00
Passive thermal control	---	-----	23.76	15.00	3.56	27.33
Heat Sinks	4	3.46	13.85	15.00	2.08	15.93
Heat Pipes	1	1.02	1.02	15.00	0.15	1.17
Radiators	1	2.34	2.34	15.00	0.35	2.69
MLI	1	3.77	3.77	15.00	0.57	4.33
Temperature sensors	25	0.01	0.25	15.00	0.04	0.29
Phase change devices	0	0.00	0.00	15.00	0.00	0.00
Thermal coatings/paint	1	0.93	0.93	15.00	0.14	1.07
Micrometeor shielding	0	0.00	0.00	15.00	0.00	0.00
S/C RTG MLI	1	0.00	0.00	15.00	0.00	0.00
S/C engine MLI	1	1.60	1.60	15.00	0.24	1.84
Semi-passive thermal control	---	-----	2.15	15.00	0.32	2.47
Louvers	1	1.35	1.35	15.00	0.20	1.55
Thermal Switches	4	0.20	0.80	15.00	0.12	0.92

4.7.4 Thermal Trades

No additional trades were run in this design session. The thermal system was designed to match the requirements.

4.7.5 Thermal Analytical Methods

The radiator panel area has been modeled along with a rough estimate of its mass. The model was based on a 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. Worst-case thermal environment assumptions were used to size the radiator. See Table 4.10 for thermal environment constant assumption.

TABLE 4.10.—THERMAL ENVIRONMENT CONSTANT ASSUMPTIONS

Variable	Value
Radiator solar absorptivity	0.14
Radiator emissivity	0.84
Radiator sun angle	90°
Radiator operating temperature	320 K
S/C radiator dissipation power	250 W

Power requirements and mass have been modeled. This modeling included propellant tank MLI and heaters and propellant line insulation and heaters. Worst case thermal environment assumptions were used to calculate the heat loss (Figure 4.13).

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. See Table 4.11 for thermal system details.

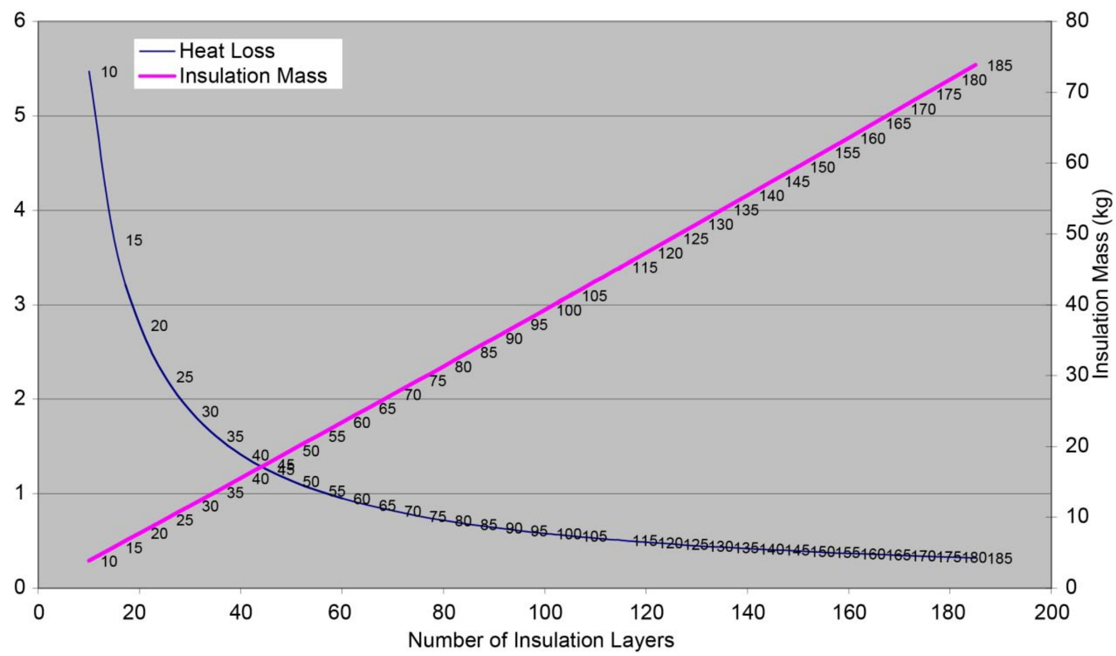

Figure 4.13.—Insulation mass versus number of layers of insulation.

TABLE 4.11—THERMAL SYSTEM DETAILS

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

The mass of the S/C MLI on the engine bulkhead was modeled to determine the mass of the insulation and heat loss (Table 4.12). The model was based on a first principles analysis of the heat transfer from the S/C through the insulation to space. The near Earth thermal environment was used to size the insulation for the maximum expected operating heat load during flight.

TABLE 4.12.—S/C MLI 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)	25
MLI layer spacing (d_i)	1.0 mm
S/C radius (r_{sc})	1.145 m

Mass estimates for the ATCS system have been completed. The components of the system included: Cold plates and heat pipes (for cooling plate assumptions see Table 4.13). The model was based on a first principle 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.

TABLE 4.13.—COOLING PLATE 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

For more detailed information on the thermal analysis a summary white paper titled “Preliminary Thermal System Sizing,” was produced. This paper is presently under publication as a NASA Contractor Report (CR).

4.7.6 Thermal Risk Inputs

The risks associated with the thermal system are based mainly on the failure of a component or multiple components of the system. The majority of the system operation is passive and therefore has a fairly high reliability. Some of the major failure mechanisms are listed below.

- Heat pipe failure. This can be due to cracking due to thermal stresses, micro-meteor impact or design defect. This likelihood of this type of failure is low. The impact of this failure would be a loss of all or a portion of the S/C's capability.
- Heater system failure. This would most likely be due to wire breakage or a controller failure. The likelihood of this type of failure is low. The impact of this failure would be a loss of certain components or propulsion capability once the vehicle is exposed to an extended period of cold
- Radiator louver failure. The thermal controller on the system can fail due to an electronics failure or power failure. Subsequently this will cause a failure of the radiator louvers reducing the effectiveness of the radiator and limiting control of the thermal system. The louvers can also experience mechanical failure causing them to be held in a fixed position or limiting their range of motion.

To improve the reliability of the system and compensate for the identified failure risks, the following system design changes can be made.

- Redundant heat pips can be utilized for each cold plate. The heat pipes can be individually run to the radiator to provide independent cooling paths. The radiator can be separated into two independent units providing additional redundancy.
- Redundant heating system controllers can be utilized. The heaters can be wired individually so that a single heater failure does not bring down any additional heaters. Additional insulation can be added to the S/C to insure that the interior components do not drop below their desired minimum temperature based on a know shadow period of operation.
- The radiator louvers can be designed to fail opened to a specified angle. This will enable the radiator to continue to operate, although not optimally, for the remainder of the mission.

5.0 Cost, Risk and Reliability

5.1 Costing: Six ASRG Configuration

The following items represent the assumptions in the costing analysis of the six ASRG REP S/C design. S/C costs reflect 50 percent confidence level. The ASRG is assumed to be flight ready by its own development project. The S/C fee is assumed at 10 percent and is not applied to science instruments (assumed to be furnished equipment). The NASA project office and technical oversight is based on 5 percent of all other costs. The costing for Phase A is based on 5 percent of S/C costs. The Launch services cost is based on guidance from 2003 NF AO. The 25 percent reserves are not applied to Launch Services or RPS costs per 2003 NF AO. Table 5.1 shows the estimations for the REP S/C and science instruments life cycle costs (LCC).

TABLE 5.1.—REP S/C AND SCIENCE INSTRUMENTS LCC

REP Thereus—NF Mission	FY08 $M
NASA project office/technical oversight	31
Phase A	17
S/C with science instruments*	315
S/C prime contractor fee (10%)	28
Launch services	172
Mission operations	94
Reserves (25%)	98
Life cycle cost	756

Table 5.2 shows the costing per work breakdown structure (WBS) line items in the REP S/C MEL in FY08 $M.

TABLE 5.2.—REP S/C COST PER MEL LINE ITEM IN FY08 $M

WBS element	Element name	DDT&E total	Flight hardware	S/C total
0.1.1	Science Payload	22.7	12.3	35.0
01.1.1.a	LORRI	6.0	2.6	8.6
01.1.1.c	LIDAR	4.8	2.1	6.9
01.1.1.d	NGIMS			
01.1.2.a	WAC	1.2	0.5	1.7
01.1.2.b	NIMS	10.7	7.1	17.8
01.2.1	AD&C	9.0	7.6	16.6
01.2.1.a.a	Sun Sensors	1.4	3.0	4.5
01.2.1.a.b	Reaction Wheels	1.0	0.8	1.8
01.2.1.a.c	Star Trackers	0.9	1.8	2.7
01.2.1.a.d	IMU	5.7	1.9	7.6
01.2.2	C&DH	11.8	3.7	15.5
01.2.2.a.a	Flight Computer	2.6	3.0	5.5
01.2.2.a.c	Data Interface Unit	0.3	0.3	0.6
01.2.2.a.e	Operations Recorder	0.1	0.1	0.2
01.2.2.a.f	Command and Control Harness (data)	3.1	0.3	3.4
	Flight Software/Firmware	5.8		5.8
01.2.3	Communications and Tracking	8.9	4.6	13.5
01.2.3.a	X/Ka HGA			
01.2.3.a.a	Transmitter/Receiver	3.0	1.2	4.2
01.2.3.a.b	Power Amp	1.4	1.1	2.5
01.2.3.a.d	Antenna	2.3	1.6	3.9
01.2.3.a.h	Cabling	0.6	0.2	0.8
01.2.3.c.a	Coaxial Cable	1.7	0.4	2.1
01.2.4	Electrical Power Subsystem	3.8	125.4	129.2
01.2.4.a.a	RPS Main System		123.4	123.4
01.2.4.b	PMAD	2.7	1.1	3.8
01.2.4.c	Power Cable and Harness Subsystem (C and HS)	1.1	0.9	2.0
01.2.5	Thermal Control (Non-Propellant)	3.6	1.0	4.7
01.2.5.a	Active thermal control	0.5	0.8	1.2
01.2.5.b	Passive and semi-passive thermal control	3.2	0.3	3.4
01.2.6	Propulsion	10.0	6.8	16.8
01.2.6.a.a	Primary EP thrusters	0.8	1.2	2.0
01.2.6.b.a	Xe propellant tank(s)	1.4	0.7	2.1
01.2.6.b	Balance of propellant management system	2.0	0.7	2.7
01.2.6.c.a	PPU mass	2.1	2.8	4.9
01.2.6.d	RCS			
01.2.6.d.a	RCS tank subassembly	0.1	0.0	0.2
01.2.6.d.b	RCS propellant management subassembly	1.7	0.8	2.5
01.2.6.d.c	RCS thruster subassembly	1.9	0.5	2.4
01.2.8	Structures and mechanisms	4.7	4.0	8.7
01.2.8.a	Structures	4.5	3.8	8.3
01.2.8.b.e	S/C adapter	0.2	0.2	0.4
Subtotal		74.6	165.4	240.1

TABLE 5.2.—REP S/C COST PER MEL LINE ITEM IN FY08 $M

WBS element	Element name	DDT&E total	Flight hardware	S/C total
Systems Integration		46.3	29.1	75.3
	Integration, Assembly and Check Out	3.8	5.2	9.0
	System Test Operations	4.4		4.4
	Ground Support Equipment	7.9		7.9
	System Engineering and Integration	14.8	16.4	31.1
	Project Management	8.4	7.5	15.9
	Launch Operations and Orbital Support	7.0		7.0
	Total Prime Cost	120.9	194.5	315.4

5.2 Reliability

5.2.1 Reliability Methodology

A first-order reliability analysis was performed based on limited information available during the conceptual design phase. Hardware elements, which have major impact on mission success (such as propulsion and power systems), were given the most attention. Many of the component failure rates utilized were assumed using engineering judgment due to a lack of knowledge—partly because new technologies are used with unknown reliability parameters and partly because relevant reliability databases are not generally accessible. Hence, the reliability numbers cited below are considered to be ballpark-accurate only and subject to considerable revision pending detailed analyses.

Figure 5.1 shows the general methodology used to estimate the reliability of the REP S/C. To cope with the limited study duration, a simple spreadsheet reliability model was developed that contained numerous and/or gates that mimicked the behavior of a full-blown fault tree analysis. This tool affords a convenient technique to generate approximate reliability values relatively quickly for alternative design options. It is a deterministic methodology, however, and therefore yields a single reliability value that is subject to considerable uncertainty. The systems uncertainty analysis (SUA) tool was therefore invoked to assess the impact of the uncertainties on the mission reliability as well as to identify the most important uncertainties.

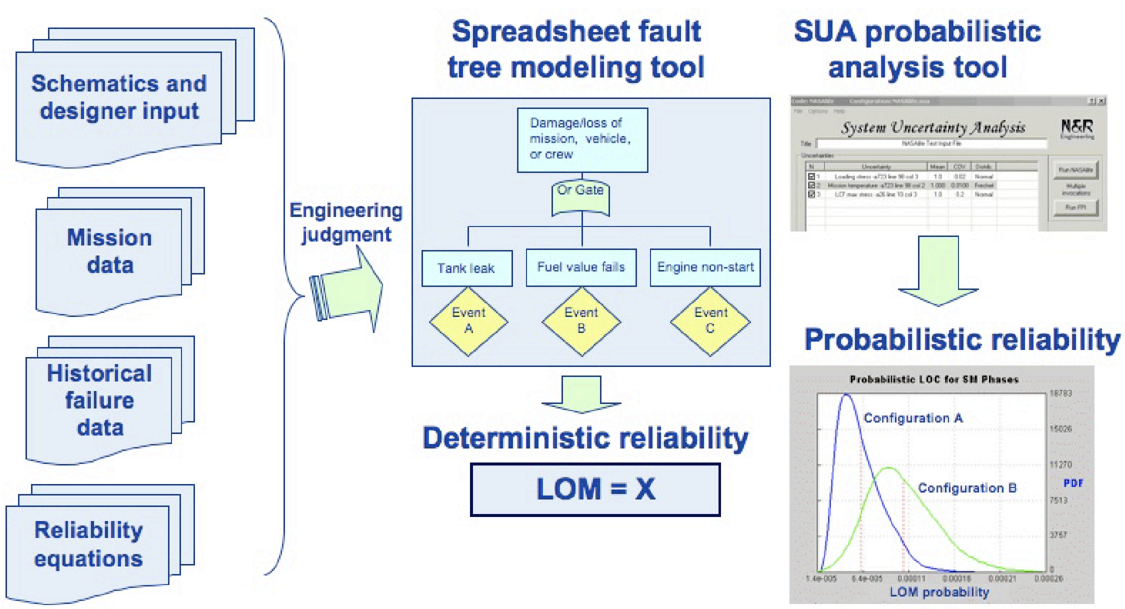

Figure 5.1.—Reliability prediction methodology.

5.2.2 Reliability Assumptions

The baseline S/C propulsion systems were assumed to consist of

- Four extended-life Hall-effect thrusters (three of the thrusters are activated sequentially one-at-a-time—3 yr = 26298 hr duty for each—for 9-yr of S/C thrusting and the fourth thruster is a standby spare).
- An RCS system consisting of eight MMH/NTO thrusters placed around the S/C body (the RCS provides some additional ΔV capability for minor orbit adjustments).

An important uncertainty is the failure rate of the advanced Hall-effect thrusters. Only two data points were available to estimate this failure rate—successful life tests of 16,265 and 30,352 hr on key elements of current-technology thrusters applicable to life-limiting elements of advanced Hall Thrusters. Two procedures are available for estimating mean-time-to-failure (MTTF) of hardware that has demonstrated lifetimes without failure

- Using “Calculating MTTF When You Have Zero Failures” by Relex Software, MTTF was estimated as 1.4427×30352 hr = 43789 hr, which results in an estimated thruster failure rate $\lambda = 1/\text{MTTF} = 2.284\times10^{-5}$/hr;
- A second “rule-of-thumb” is the 1/3 rule, in which estimated MTTF = demonstrated life divided by 1/3 = 30352/0.33333 = 91056 hr, resulting in an estimated failure rate of 1.098×10^{-5}/hr.

Feedback from thruster designers indicated the first estimate is more likely at this point.

The ASRG power system is the second critical element of the S/C. The baseline configuration assumed that six ASRG units would operate for the entire 10-yr (= 87,660 hr) mission to provide electrical power and that one of these units could fail without causing complete mission failure (although operational compromises would certainly be required). Estimates of an ASRG MTTF was reverse-engineered from reliability results performed for a February 2007 ASRG Engineering Unit Final Design Review (FDR). For the FDR study option which assumed the use of high-strength components, 3 yr in fueled storage, and 14 yr operating life, an ASRG mission reliability of 0.9906 had been estimated. Assuming the MTTF for the storage portion of the mission was 10 times that of the operational phase, an ASRG operational MTTF of 13.900,000 hr was calculated (resulting in an estimated failure rate of 7.1942×10^{-8}/hr), which is significantly longer than values used in earlier analyses.

Additional hardware assumption changes made from phase 1 of this study included

(1) Propulsion—Decreased the common cause factor (CCF) value from 0.10 to 0.05 by assuming that extensive ground testing would precede flight.
(2) RCS—Reduced duty life (once every 2 wk for 20 min rather than the original 1 min every 6 hr);
(3) Avionics—Two redundant pairs of star trackers rather than a single pair.
(4) Engine gimbals—Replaced the initial guess of 1.0×10^{-6}/demand with a Cassini gimbal failure rate 2 orders of magnitude less.

5.2.3 Reliability Results

Reliability analyses were performed for the 10-yr mission starting with the baseline configuration with three of the thruster strings (all the elements downstream of the Xe propellant tank and regulator—thruster, flow controller, and PPU) activated serially for 9 yr of EP thrusting. As the sensitivity results in Figure 5.3 indicate, the Hall thruster system configuration has the biggest impact on overall S/C reliability. Thruster system and overall S/C reliabilities determined for different numbers of thruster spares and values of thruster MTTF are summarized in Table 5.3.

TABLE 5.3.—RELIABILITY CASES FOR VARYING NUMBER OF THRUSTERS AND MTTFS

Total no. of Hall thrusters*	No. of Hall thruster spares	Assumed Hall thruster MTTF (hr)	Thruster strings system reliability	Overall s/c reliability
4	1	43789	0.807	0.754
5	2	43789	0.930	0.869
6	3	43789	0.953	0.891
4	1	91056	0.932	0.871
5	2	91056	0.966	0.903
6	3	91056	0.970	0.906

* Total minus spares = 3 operating thrusters (serially one at a time)

The highlighted result shows the value of adding an additional thruster string to the configuration (adding two additional spares has little impact) and technology efforts to improve thruster MTTF to the higher of the estimated values.

Table 5.4 summarizes the reliability contributions of the various S/C systems to the overall S/C reliability (exclusive of the science payload) for the highlighted configuration.

TABLE 5.4.—RELIABILITY SUMMARY FOR SIX REP ASRG'S AND FIVE HIGH-MTTF THRUSTERS CONFIGURATION

	Contribution
Propulsion system (including RCS)	41.3%
Electrical power	0.9%
AD&C	8.1%
Communication	18.3%
Thermal control	31.0%
Structure and mechanical systems	0.4%
Science (not modeled)	-------
Mission reliability	0.903

The impact of four major failure rate uncertainties (thruster, Xe tank, RCS, ASRG) on mission reliability is displayed in Figure 5.2 in terms of the probability and cumulative density functions (pdf and cdf). Each uncertainty was assumed to be normally distributed with a 0.50 coefficient of variation. These uncertainties have a large impact on predicted mission reliability: the 5 to 95 percentile range of mission success is 0.88 to 0.93 out of 1.0.

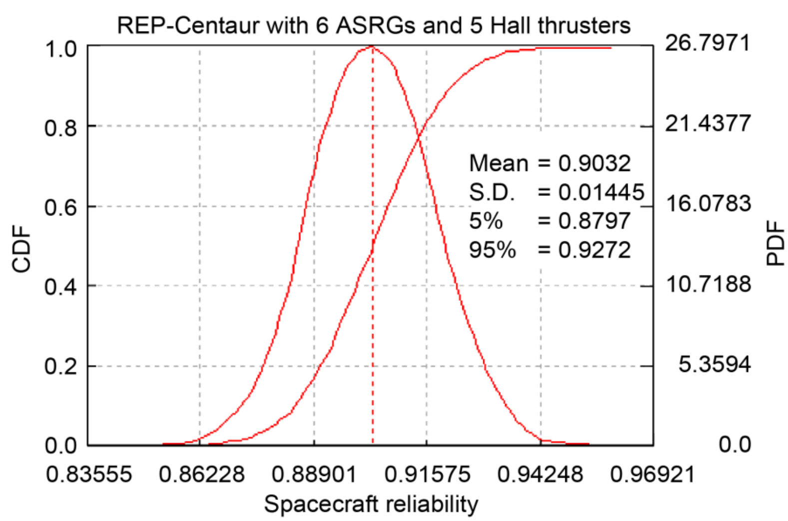

Figure 5.2.—Probabilistic S/C reliability (capturing major uncertainties).

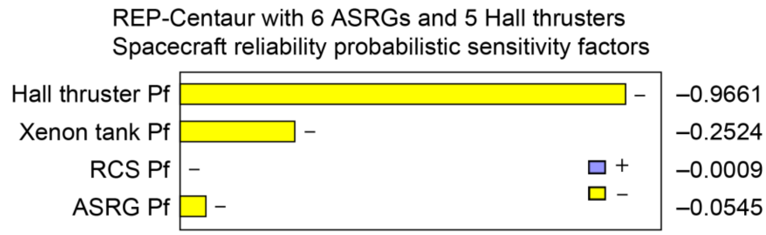

Figure 5.3.—Probabilistic sensitivity factors.

The probabilistic sensitivity factors are shown in Figure 5.3. The longer bars identify the more important uncertainties. In this case, the thruster failure rate (P_f= 1.0982×10^{-5}/hr) uncertainty is dominant.

5.2.4 Interpretation of the Reliability Results

The low mission reliability estimate is somewhat discouraging at first glance. Furthermore, a large improvement is not possible with just a single added redundancy since the unreliability is spread relatively evenly over many components. This suggests that reliability improvement would come from many design improvements rather than just one.

It should be noted, however, that the modeling fidelity for this design exercise was quite low. Many failure rates were merely educated guesses due to the rapid pace of the analysis and especially efforts to come up with reasonable estimates for the hardware elements driving the overall S/C reliability. Hence, it is not known whether the low values reflect the reality of substantial risks in long duration deep space missions or are artificially low because of the modeling fidelity. A more in-depth reliability analysis is required to resolve this uncertainty.

6.0 Trades

The original baseline at the start of the COMPASS session was an eight ASRG configuration for the bus. Upon costing, the number of ASRGs was reduced to six, and that design became the new baseline. Both designs are reported here in the trade section.

6.1 Case 1—REP Centaur Orbiter With Six ASRGs—Baseline

This design became the baseline of the mission since it accomplished the science goals and fit inside the NF cost box. The details of this design trade are documented in this report in detail.

6.2 Case 2—REP Centaur Orbiter With Eight ASRGs

Initially, the REP Centaur orbiter design contained eight ASRGs. This design was able to deliver the payload required in a reasonable amount of trip time. It did not, however, fit inside the NF cost cap. Therefore, the first trade performed was to reduce the number of ASRGs down to six, and adjust the mission in order to fly to the centaur body with reduced power available to the thrusters. In order to reduce the S/C mass, one of the science instruments (NGMIS) was dropped off of the science payload (see Section 2.7). Note that New THEMIS was put back into the science instrument listing at the final design.

6.2.1 Eight ASRG REP Centaur Orbiter Summary

The eight ASRG case is still a NF Class Mission, to be launched on an Atlas 551 using a Star 48 solid motor.

- 8.5 yr trip time to Centaur Thereus
  - Flyby a Trojan in 2 yr
- 1 yr Centaur Science Period
  - Science during Centaur perihelion for maximum activity
    - Non-Perihelion targeting possible if active instruments used
  - 55 kg/60 W of passive science instruments
  - 6 kbps with DSN
- Eight ASRG
  - 1040 W EOL (10 yr) power output
  - Less plutonium than NH
- 3+1 advanced, long life Hall thrusters
  - Thruster throughput >4 times current flight systems
- Relatively high power communications system
  - Using EP power when coasting
  - Could allow 'high' data rates, less DSN time, less cost
- Avionics, GN&C, hydrazine propulsion, thermal systems similar to other Deep Space Missions

Figure 6.1 shows the eight ASRG configuration of the Centaur orbiter from the beginning of this study. This was originally the baseline but was supplanted by the six ASRG case.

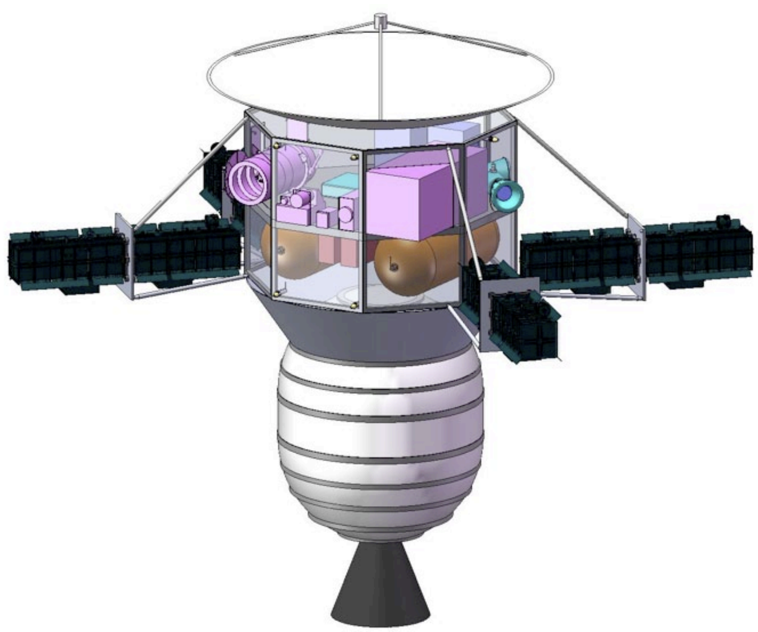

Figure 6.1.—Concept design using eight ASRGs.

6.2.1.1 Eight ASRG Mission Trajectory—Thereus

After the REP S/C bottoms-up analysis and sizing, the amount of payload delivered to Bienor was found to be insufficient to accommodate the baseline S/C design and science instruments mass. Therefore, analysis was performed to locate another centaur body and trajectory that would place the REP S/C and science instruments at the Centaur before perihelion for science requirements, and allow enough delivered mass to accommodate the REP S/C.

The Centaur Thereus was chosen to fit those parameters. The baseline trajectory parameters for the Thereus trajectory were

- Launch Mass (Mo): 1296.6 kg
- C_3: 95.33 km^2/s^2
- Launch Date: November 6, 2024
- Fly-by Date: July 29, 2026
- Arrival Date: May 8, 2033
- Power: 750 W
- I_{sp}: 2000 s
- ΔV: 8.59 km/s
- Propellant (Mp): 469.73 kg
- Mo – Mp: 836.87 kg
- Fly-by Target: Prylis

6.2.1.2 Mission Trajectory Details: Eight ASRG Case

Figure 6.2 shows the baseline trajectory to Thereus for the REP Science Orbiter using eight ASRG power levels (750 W). Earth's orbit is in blue in the center, Thereus's is in the aqua color on the outside. The REP trajectory is in red, taking it through the Trojan asteroid belt shown in green. This baseline trajectory is for the thruster operating at 2000 I_{sp} and a power level of 750 W.

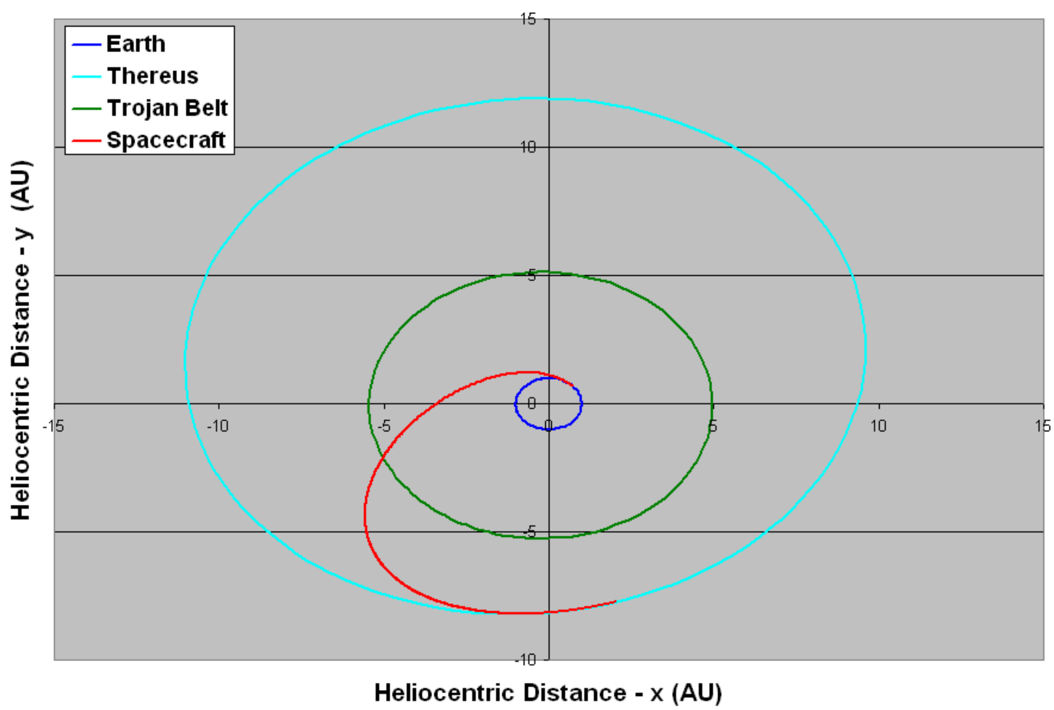

Figure 6.2.—Trajectory to Thereus.

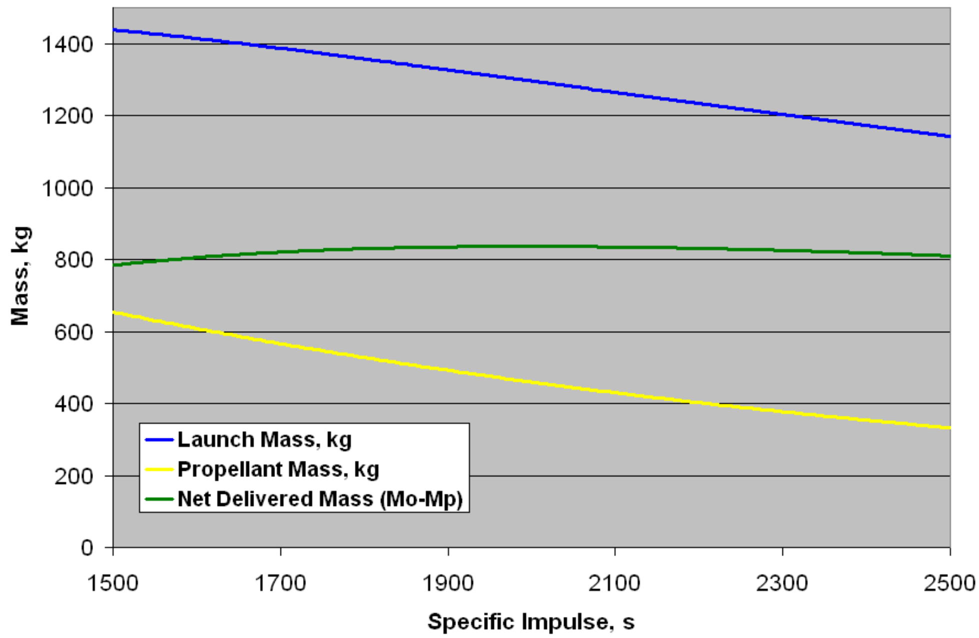

Figure 6.3.—Launch, propellant and delivered mass as a function of I_{sp}.

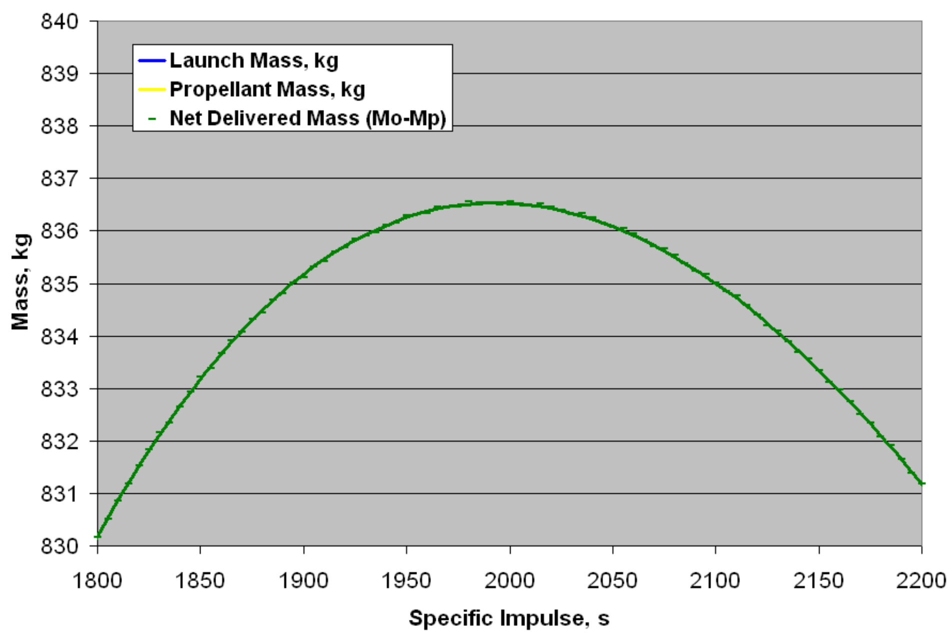

Figure 6.4.—Delivered Mass as a function of I_{sp}.

The Centaur Thereus is a more difficult target to hit than Bienor. Thereus is a difficult target to reach in case of a failure in an RPS for baseline missions beyond 8 yr. Should there be an RPS failure inside the 8.5-yr mission lifetime, the possibility of reaching Thereus becomes more difficult. There might not be the propellant budget in the current design to accommodate this sort of failure. Figure 6.3 shows the relationship of mass and I_{sp} for the Thereus trajectory for differing power levels. Bienor showed approximately linear relationship of mass versus trip time during period of interest.

Analysis was performed on the sensitivity of the trajectory to I_{sp}. Figure 6.3 shows that the launch mass and propellant mass are greatly affected by changed in I_{sp}.

However, Figure 6.4 shows that net delivered mass was very insensitive to I_{sp} changes, varying only about 6 kg through a variation of 400 s in I_{sp}.

6.2.2 Eight ASRG System Summary Masses (MEL and System Sheet)

The following table is the system summary of the eight ASRG design configuration.

The MEL (Table 6.1) captures the bottoms-up estimation of CBE and growth percentage line item by item from the subsystem engineer.

The system integration in Table 6.2 summarizes 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. The total additional growth amounted to 8.8 percent (51 kg). The bottoms-up average growth on the dry mass of the REP S/C totaled 21.2 percent (122 kg). This was for the dry mass of 698.8 kg. With the additional system level growth, the total dry mass is 750 kg. Given the available launch performance of 1296.6 kg to the C_3 targets, and the 10 kg S/C adaptor taken out of that, the remaining mass available to the REP S/C is 1286.6 kg. The total wet mass, with 30 percent total growth applied on top of it yields a total wet mass of the REP S/C of 1278 kg, which fits in with an 8.5 kg margin.

TABLE 6.1.—EIGHT ASRG CASE TOP LEVEL MEL

Description REP Centaur Mission (September 4, 2007))	CBE mass (kg)	Growth (%)	Growth (kg)	Total mass (kg)
REP Spacecraft (Payload and Stage)	1105.19	11.07	122.36	1227.55
Science Payload	42.42	30.00	12.73	55.15
REP Bus	1062.77	10.32	109.64	1172.41
AD&C	13.52	20.00	2.70	16.22
C&DH	33.30	34.26	11.41	44.71
Communications and Tracking	39.00	34.10	13.30	52.30
Electrical power subsystem	234.76	16.56	38.88	273.64
Thermal control (non-propellant)	36.49	18.00	6.57	43.06
Propulsion	85.98	25.67	22.07	108.05
Propellant	528.78	0.00	0.00	528.78
Structures and mechanisms	90.94	16.17	14.70	105.65

TABLE 6.2.—EIGHT ASRG REP CONFIGURATION SYSTEM SUMMARY

REP Spacecraft Master Equipment List Rack-up (Mass)		COMPASS REP Design			
WBS	Main subsystems	CBE Mass (kg)	Growth (kg)	Total mass (kg)	Aggregate growth (%)
01	REP Spacecraft (payload and bus)	1105.2	122.4	1227.6	-----
01.1	Science Payload	42.4	12.7	55.1	30.0
01.2	*REP Bus*	*1062.8*	*109.6*	*1172.4*	
01.2.1	AD&C	13.5	5.4	16.2	40.0
01.2.2	C&DH	33.3	11.4	44.7	34.3
01.2.3	Communications and Tracking	39.0	13.3	52.3	34.1
01.2.4	Electric Power	234.8	38.9	273.6	16.6
01.2.5	Thermal Control	36.5	6.6	43.1	18.0
01.2.6	Propulsion	86.0	22.1	108.1	25.7
01.2.7	Propellant	528.8	------	--------	-----
01.2.8	Structures and Mechanisms	90.9	14.7	105.6	16.2
	Estimated REP S/C dry mass	576	122	698.8	21.2
	Estimated REP S/C wet mass	1105	122	1227.6	
	System Level Growth Calculations				Total growth
	Desired System Level Growth	576	173	749.3	30.0
	Additional Growth (carried at system level)	-----	51	--------	8.8
	Total Wet Mass with Growth	1105	173	1278.1	-----
	Available launch performance to C_3 (kg)			1286.6	
	Launch margin available (kg)			8.5	

7.0 Challenges, Lessons Learned, and Areas For Future Study

In order to determine what are the key parameters for EP devices to perform these REP missions a design study was completed to design an REP S/C to orbit a Centaur in a NF cost cap. The design shows that an orbiter using several long lived (~200 kg Xe throughput), low power (~700 W) Hall thrusters teamed with six (150 W each) ASRG can deliver 60 kg of science instruments to a Centaur in 10 yr within the NF cost cap. Optimal I_{sp}s for the Hall thrusters were found to be around 2000 s with thruster efficiencies over 40 percent. Not only can the REP S/C enable orbiting a Centaur (when compared to an all chemical mission only capable of flybys) but the additional power from the REP system can be reused to enhance science and simplify communications.

Key to the feasibility for REP missions are long life, low power EP devices, low mass RPS and light S/C components. Performance of the REP mission could be improved by increasing ASRG power density and increasing thruster lifetime and efficiency.

7.1 Six ASRG REP Centaur Orbiter Summary

7.1.1 Potential Mass Reductions

- Improved ASRG (greater power density)
  - Allows for elimination of one to two ASRGs from the baseline design
- ASRG mounting methods
  - Trade of structure and thermal radiative coupling (effective SRG power output)
- Optimize structures, more composites
- Propulsion
  - Higher EP I_{sp} is shown to not be an important driver
  - Efficiency and longer life thrusters are a major driver
- Communications
  - High data rates cost and mission selling rate: reduced data rate not recommended
  - Lighter antennas designs are recommended

7.1.2 Areas for Future Work

Further work is needed to trade ASRG mounting methods, optimizing structures (utilize more composites), and optimizing communications performance/mass based on the additional power available from the ASRG system.

Fix the issues open in the current design.

- Exploring Centaur Design Space
  - Propulsion: Other Hall, ion options
  - Power: Other power levels, power supplies
  - Other targets
- Flagship and Discovery designs
- Inputs to EP Technology Development Program

Appendix A.—Acronyms and Abbreviations

ACS	Attitude Control System
AD&C	Attitude, Determination and Control
AIAA	American Institute for Aeronautics and Astronautics
Al	aluminum
ANSI	American National Standards Institute
AO	Announcement of Opportunity
APL	Applied Physics Laboratory
ASRG	Advanced Stirling Radioisotope Generators
BAE	British Aerospace
BOL	beginning of life
C&DH	Command and Data Handling
CAD	computer aided design
CBE	current best estimate
CCF	common cause factor
COM	center of mass
Comm	Communications
COMPASS	COllaborative Modeling and Parametric Assessment of Space Systems
CONOPS	Concept of Operations
COPV	Composite Overwrapped Pressure Vessel
COTS	commercial-off-the-shelf
DPU	dual processing unit
DSN	Deep Space Network
ELV	Expendable Launch Vehicle
EOL	end of life
EOM	end of mission
EP	Electric Propulsion
FEA	finite element analysis
FOM	figure of merit
FY	fiscal year
GLIDE	GLobal Integrated Design Environment
GN&C	Guidance, Navigation and Control
GPHS	general purpose heat source
GRC	NASA Glenn Research Center
He	helium
HGA	high gain antenna
HiVHAC	High Voltage Hall Accelerator
IEM	integrated electronics module
IMU	Inertial Measurement Unit
I_{sp}	specific impulse
JPL	NASA Jet Propulsion Laboratory
KBO	Kuiper Belt Objects
LCC	life cycle costs
Li	lithium
LIDAR	Laser Detection and Ranging
LORRI	Long Range Reconnaissance Imager
MEL	Master Equipment List
MGA	Mass Growth Allowance
MLI	multilayer insulation
MMH/NTO	monomethyl hydrazine and nitrogen tetroxide bipropellant system
MO	Mission of Opportunity
MTTF	mean-time-to-failure
NAC	narrow angle camera
NASA	National Aeronautics and Space Administration
NG SIRU	Northrop Grumman Scalable Inertial Reference Unit
NGIMS	Neutral Gas and Ion Mass Spectrometer
NH	New Horizons
NIMS	Near Infrared Mapping Spectrometer
NIST	National Institute of Standards and Technology
OSS	Office of Space Science
OTS	off-the-shelf
PEL	Power Equipment List
PMAD	Power Management and Distribution
PMS	Propellant Management System
PPU	Power Processing Units
RBI	Regulator/Bus Protection

RCS	Reaction Control System
REP	Radioisotope Electric Propulsion
RF	radio frequency
RPS	Radioisotope Power Systems
RSEN	Reduced State Encounter Navigation
S/C	spacecraft
SEAKR	SEAKR Engineering, Inc.
SEP	Solar Electric Propulsion
SEU	single event upset
SRG	Stirling Radioisotope Generator
SSR	solid state recorder
SUA	Systems Uncertainty Analysis
TBD	to be determined
THEMIS	Thermal Emission Imaging System
Ti	titanium
TT&C	Telemetry, Tracking and Command
TWTA	Traveling Wave Tube Amplifier
USO	ultra-stable oscillator
WAC	wide angle camera
WBS	work breakdown structure
WGA	weight growth allowance
WGS	weight growth schedule
Xe	xenon

Appendix B.—Six ASRG Design Rendered Drawings

Figure B.1 shows the dynamic envelope of the REP S/C and Star 48 motor inside the Delta IV fairing. Figure B.2 shows the rendered image of the REP Stage with the science payload side shown. Figure B.3 shows the rendered image of the REP Stage with the Hall Thruster side shown.

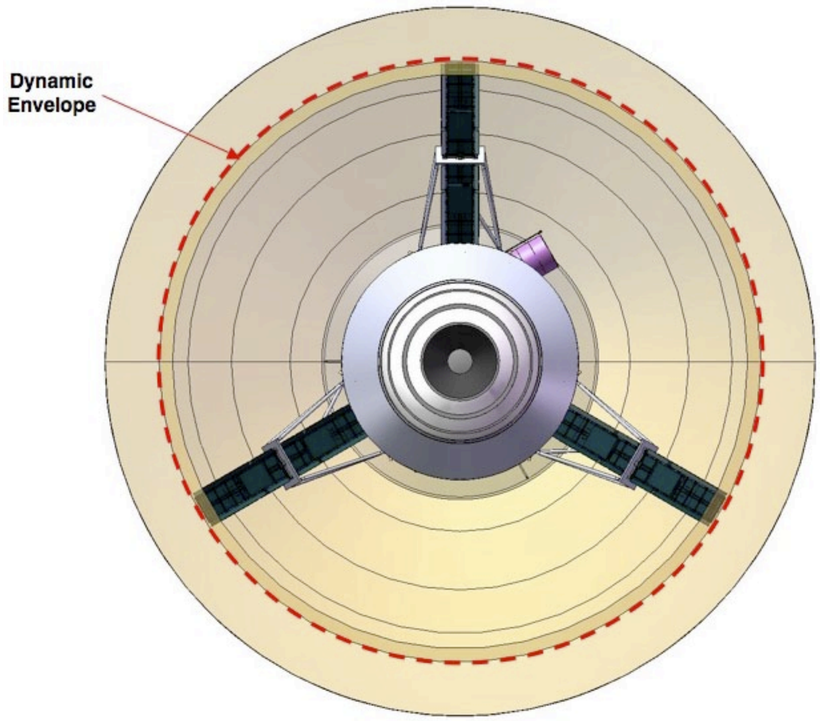

Figure B.1.—REP S/C inside Delta IV Fairing.

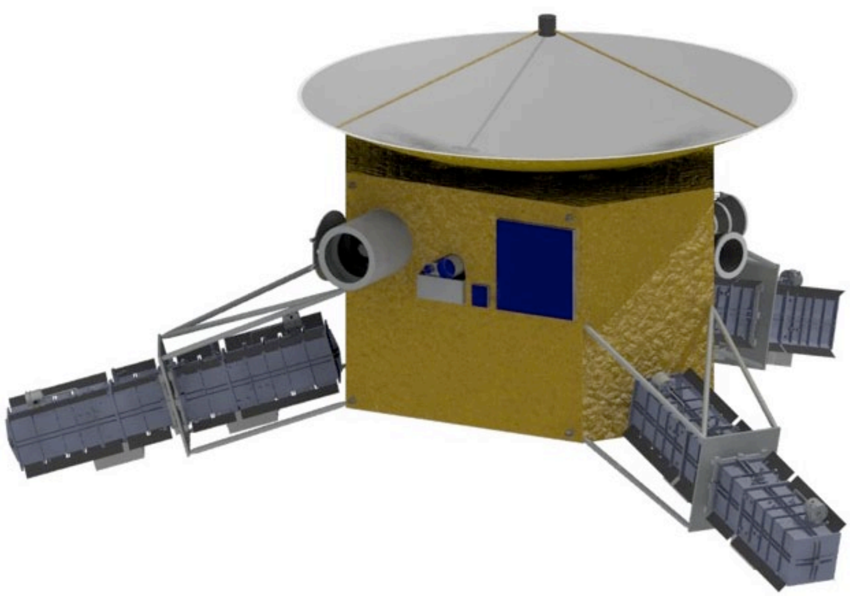

Figure B.2.—REP Stage after Star Motor separation.

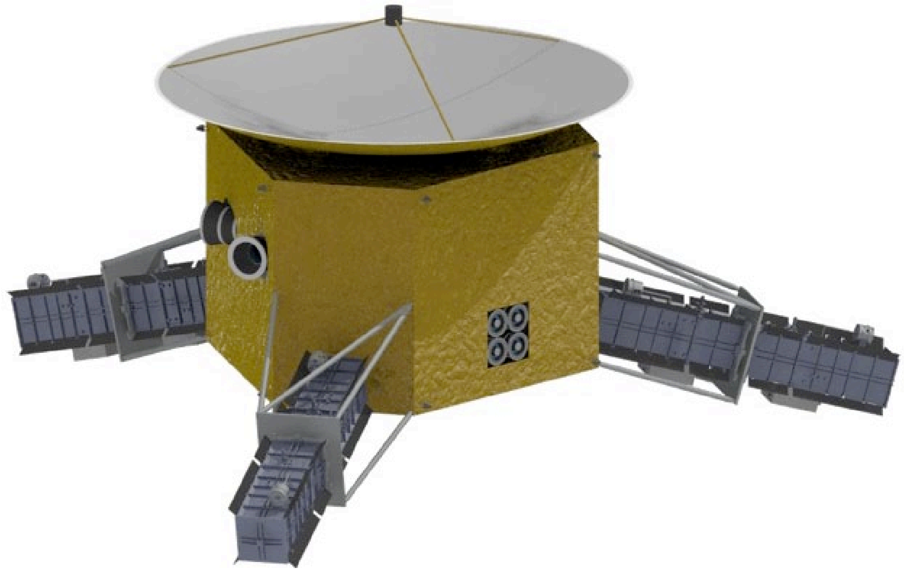

Figure B.3.—Thruster face of the REP S/C.

Appendix C.—Eight ASRG Design Rendered Drawings

Figure C.1 shows the dynamic envelope of the REP S/C and Star 48 motor inside the Atlas 551 fairing. Figure C.2 shows the dimensions of the Eight ASRG Case. Figure C.3 shows the REP S/C face with the four hall thrusters pointing out of the power and propulsion deck. Figure C.4 displays the science package face of the REP S/C. Figure C.5 shows the Avionics and flight deck of the REP S/C with all major components labeled. Figure C.6 presents the Power and Propulsion deck of the REP S/C with all major components labeled.

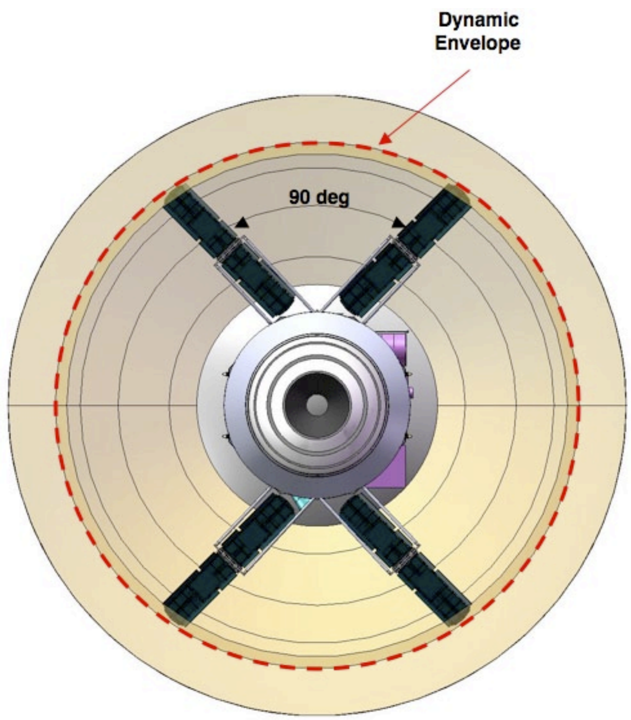

Figure C.1.—Dynamic envelope of the REP S/C inside the Atlas 551 Fairing.

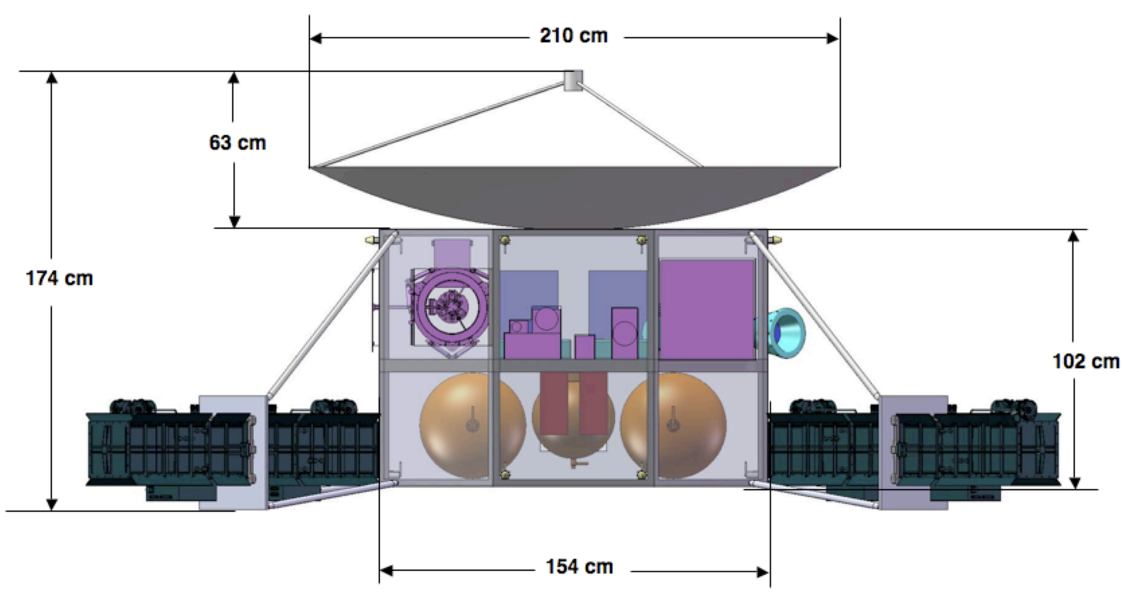

Figure C.2.—Dimensions of the eight ASRG Case.

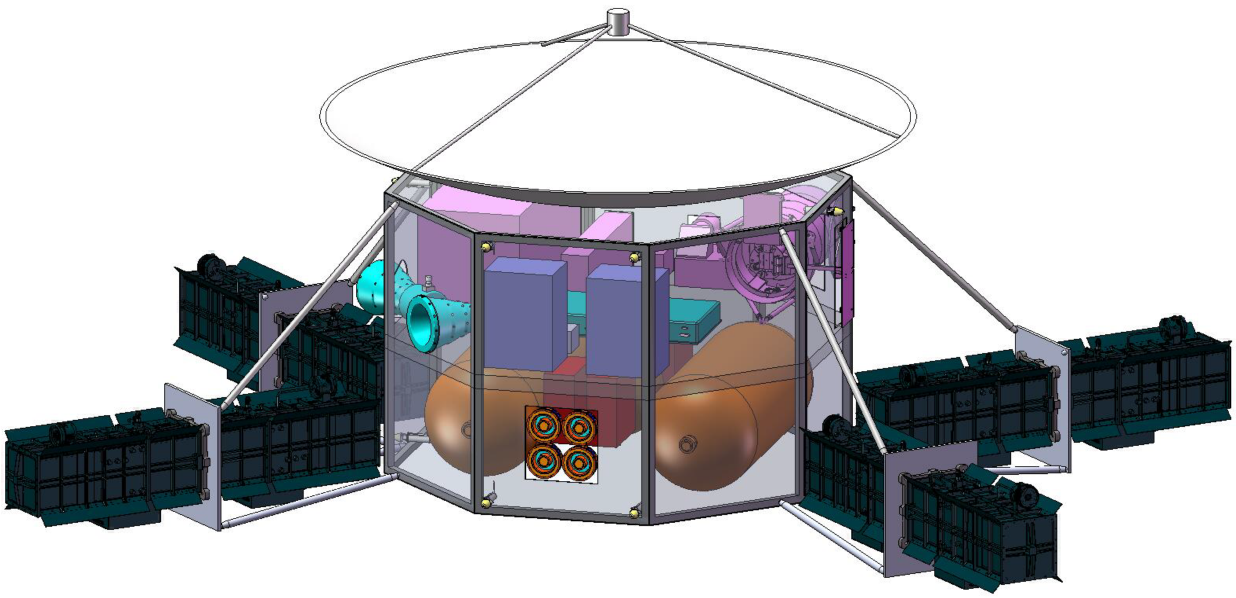

Figure C.3.—EP Thruster face of the eight ASRG REP Design.

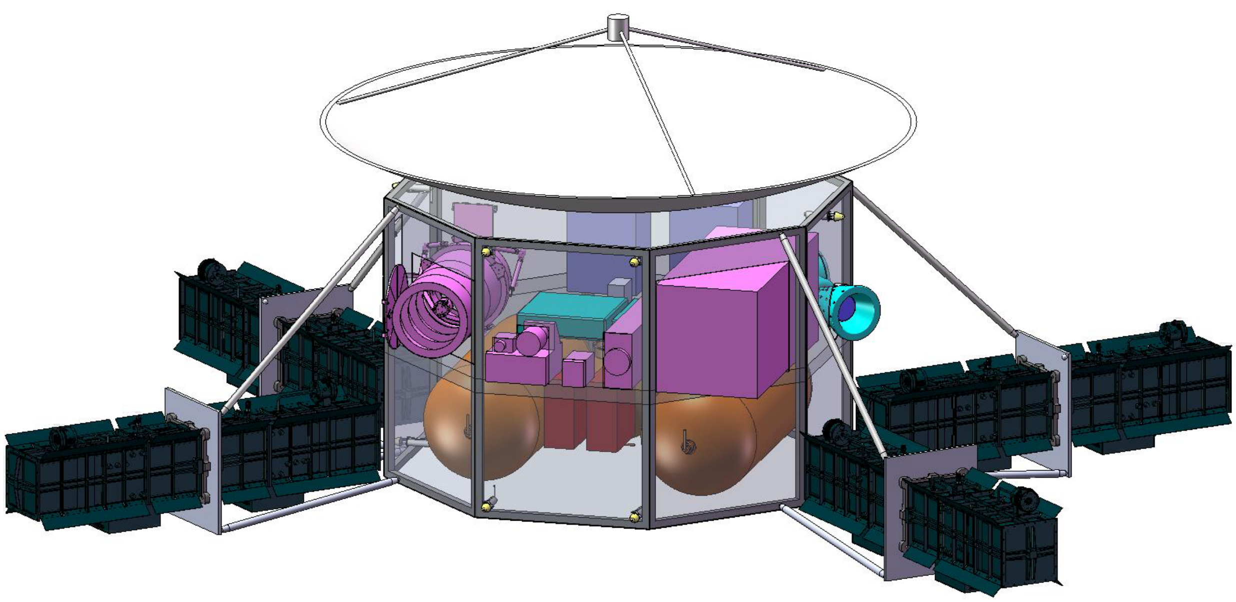

Figure C.4.—Science package face of the REP S/C.

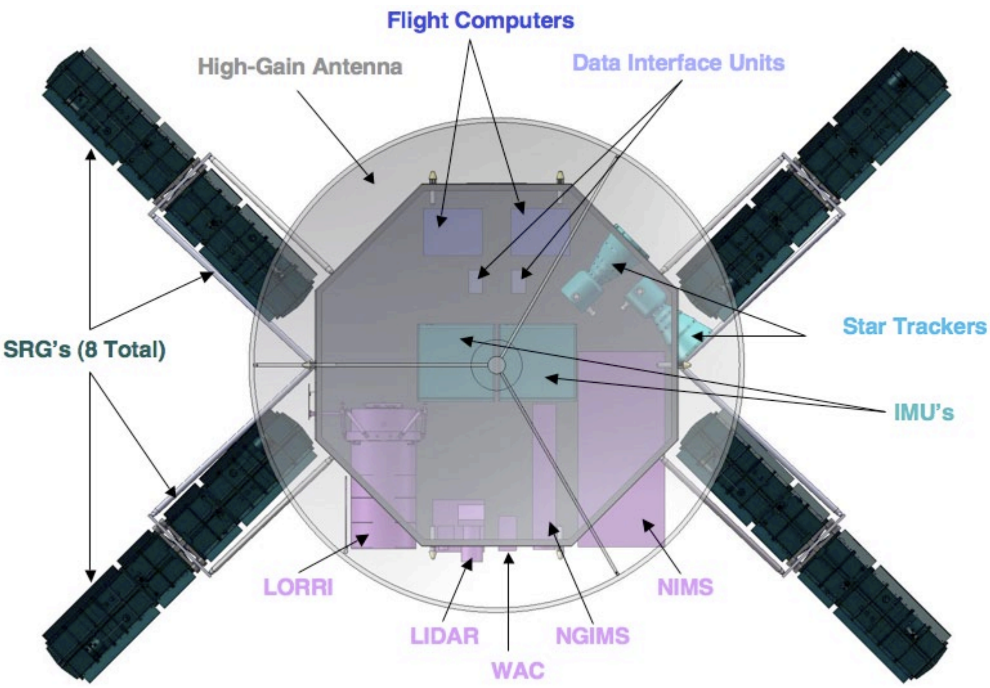

Figure C.5.—Avionics deck of the eight ASRG design.

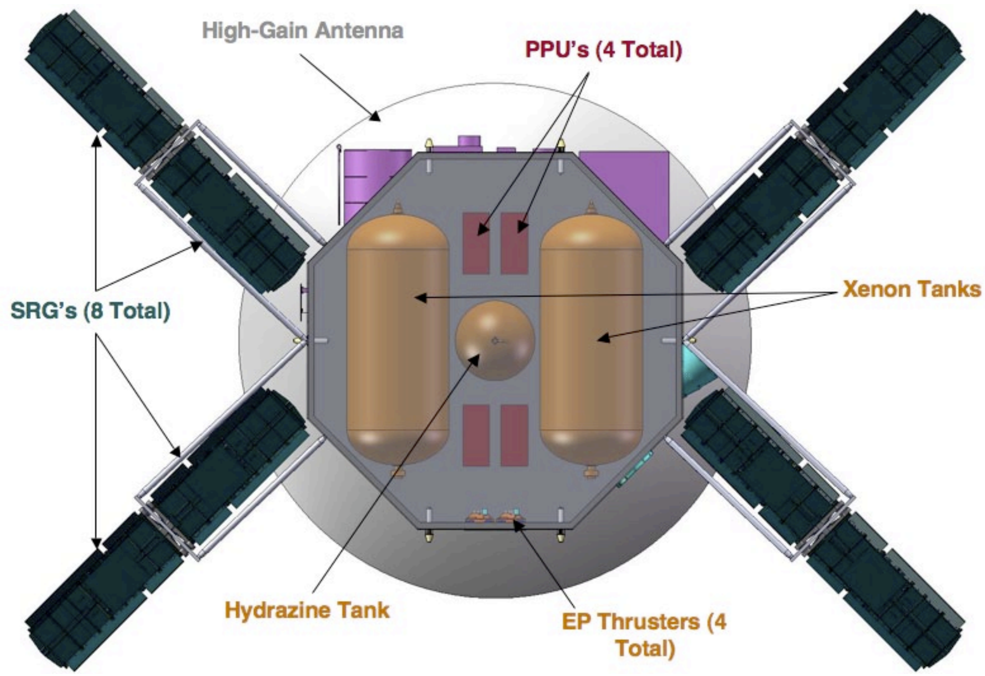

Figure C.6.—Power and propulsion deck of the eight ASRG design.

Appendix D.—Mission Analysis of Centaur Body Bienor

D.1 Initial Potential Centaur Target Trajectory Analysis

A Centaur class body, Bienor, was chosen as the first representative body for this Centaur mission. Bienor (Asteroid 54598, Bienor, is a Centaur-type asteroid discovered on August 27, 2000, by the Deep Ecliptic Survey project at Cerro Tololo near Pedernales, Coquimbo, Chile. It has a period of 66 yr, 305 days) is representative of this class of body. If possible, the mission will add on a Trojan asteroid body flyby.

Several trades were run on the affect of thruster power for trip times to see the impact on net delivered mass. Performance is very sensitive to both trip time and available power. A trade was performed at Hall Effect Thruster performance fit (I_{sp} may be too high) to analyze the affect of power on trip time and delivered mass. It was noted that optimal I_{sp} rises quickly with available power and trip times greater than 9 yr.

Figure D.1 shows the net delivered mass to the Centaur, Bienor, as a function of trip time for three varying power levels of the REP S/C thruster system (500 to 750 W).

Figure D.2 shows net delivered mass to the Centaur Bienor as a function of thruster input power for a series of different trip time trajectories. The mission was baselined on a 750 W power system. The trades in the chart were made varying mission and launch date used a total power of 750 W.

Figure D.3 shows the launch mass (initial mass), propellant mass, and net delivered mass to Bienor1 as a function of trip time at the 750 W ASRG configuration. The requirement on trip time is to arrive at Bienor1 in about a 10-yr total trip time. Therefore, the initial mass of the REP S/C can be no more than roughly 1100 kg at launch.

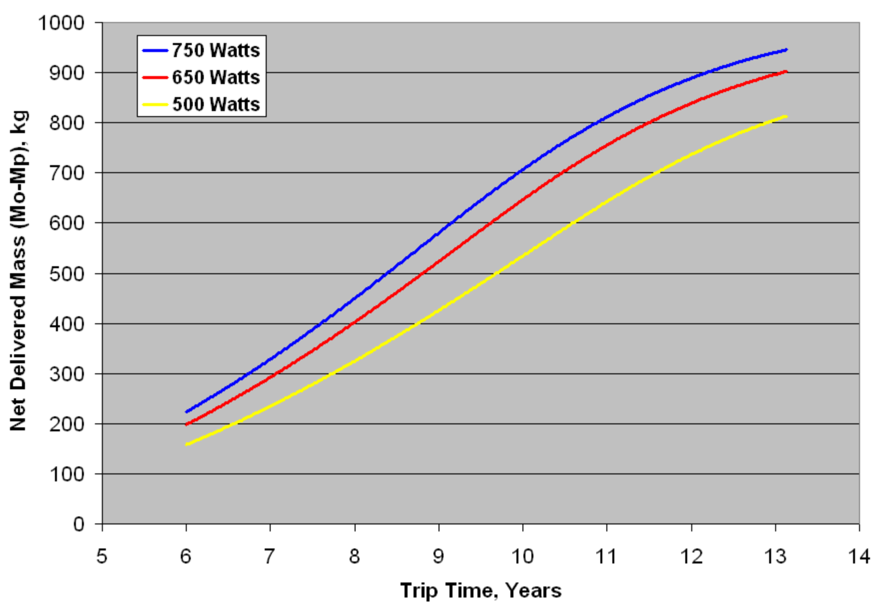

Figure D.1.—Bienor delivered mass versus trip time and thruster system.

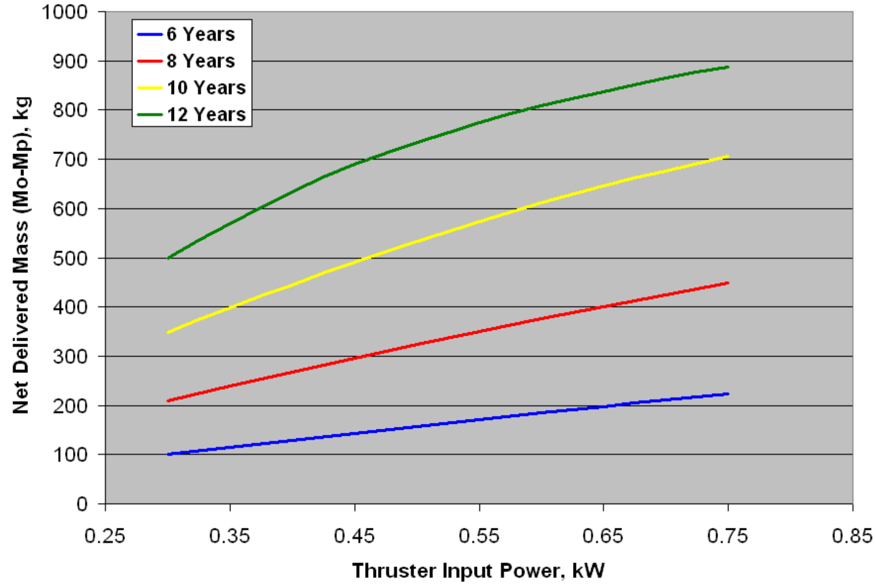

Figure D.2.—Bienor delivered mass versus thruster input.

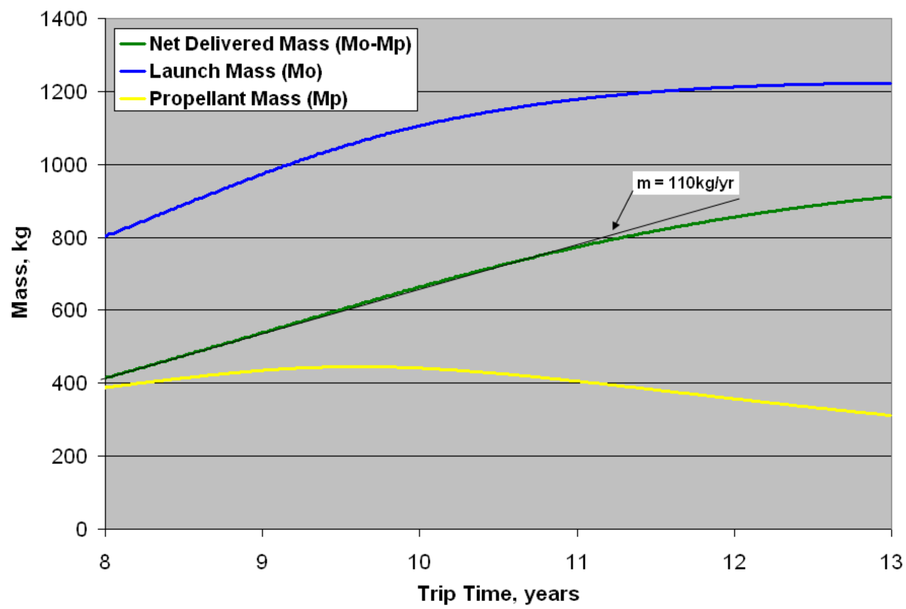

Figure D.3.—Launch, propellant, and net masses to Bienor versus trip time for 750 W ASRG.

Figure D.4 shows the mission sensitivities (mass, thruster operation time, and coast period) to I_{sp} of the Electric thrusters. This trade will help to decide which I_{sp} to operate the electric thrusters at.

Figure D.5 is the baseline trajectory from Earth to Bienor1 for this mission, at an I_{sp} of 2306 s and power of 750 W. The thruster I_{sp} choice will be 1950 s, which will change all of the performance analysis in the previous four figures. The erosion of the thrusters at higher I_{sp} are high, and the stability issues of the current at higher power caused the thruster I_{sp} selection change for the mission.

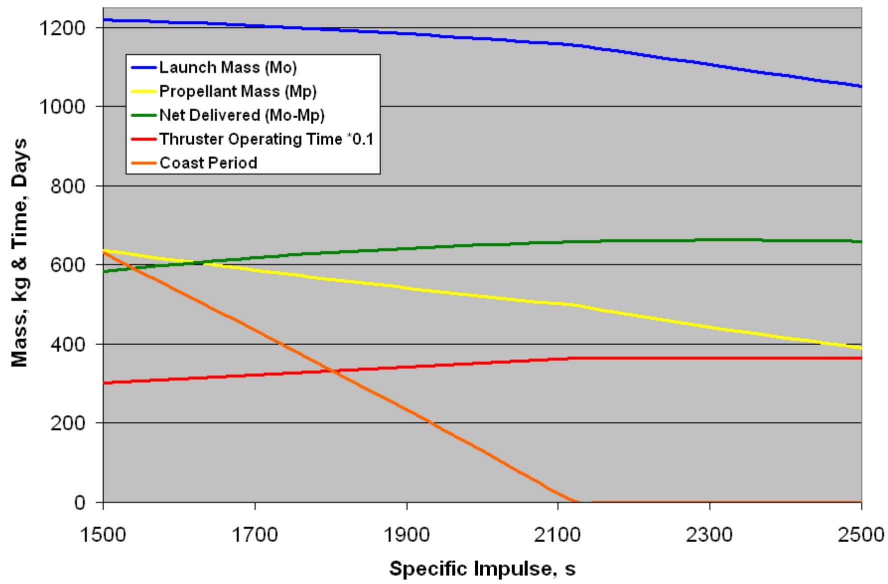

Figure D.4.—Mission sensitivities versus I_{sp}.

Figure D.5.—Baseline Earth to Bienor.

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REPORT DOCUMENTATION PAGE			Form Approved OMB No. 0704-0188

1. REPORT DATE *(DD-MM-YYYY)*
01-02-2011

2. REPORT TYPE
Technical Memorandum

3. DATES COVERED *(From - To)*

4. TITLE AND SUBTITLE
COMPASS Final Report: Radioisotope Electric Propulsion (REP) Centaur Orbiter New Frontiers Mission

5a. CONTRACT NUMBER

5b. GRANT NUMBER

5c. PROGRAM ELEMENT NUMBER

6. AUTHOR(S)
Oleson, Steven, R.; McGuire, Melissa, L.

5d. PROJECT NUMBER

5e. TASK NUMBER

5f. WORK UNIT NUMBER
WBS 346620.02.01.02.03.01

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
John H. Glenn Research Center at Lewis Field
Cleveland, Ohio 44135-3191

8. PERFORMING ORGANIZATION REPORT NUMBER
E-17599

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)
National Aeronautics and Space Administration
Washington, DC 20546-0001

10. SPONSORING/MONITOR'S ACRONYM(S)
NASA

11. SPONSORING/MONITORING REPORT NUMBER
NASA/TM-2011-216971

12. DISTRIBUTION/AVAILABILITY STATEMENT
Unclassified-Unlimited
Subject Categories: 18 and 13
Available electronically at http://www.sti.nasa.gov
This publication is available from the NASA Center for AeroSpace Information, 443-757-5802

13. SUPPLEMENTARY NOTES

14. ABSTRACT
Radioisotope Electric Propulsion (REP) has been shown in past studies to enable missions to outer planetary bodies including the orbiting of Centaur asteroids. Key to the feasibility for REP missions are long life, low power electric propulsion (EP) devices, low mass Radioisotope Power System (RPS) and light spacecraft (S/C) components. In order to determine the key parameters for EP devices to perform these REP missions a design study was completed to design an REP S/C to orbit a Centaur in a New Frontiers (NF) cost cap. The design shows that an orbiter using several long lived (~200 kg xenon (Xe) throughput), low power (~700 W) Hall thrusters teamed with six (150 W each) Advanced Stirling Radioisotope Generators (ASRG) can deliver 60 kg of science instruments to a Centaur in 10 yr within the NF cost cap. Optimal specific impulses (Isp) for the Hall thrusters were found to be around 2000 s with thruster efficiencies over 40 percent. Not only can the REP S/C enable orbiting a Centaur (when compared to an all chemical mission only capable of flybys) but the additional power from the REP system can be used to enhance science and simplify communications. The mission design detailed in this report is a Radioisotope Power System (RPS) powered EP science orbiter to the Centaur Thereus with arrival 10 yr after launch, ending in a 1 yr science mapping mission. Along the trajectory, approximately 1.5 yr into the mission, the REP S/C does a flyby of the Trojan asteroid Tlepolemus. The total ΔV of the trajectory is 8.9 km/s. The REP S/C is delivered to orbit on an Atlas 551 class launch vehicle with a Star 48 B solid rocket stage.

15. SUBJECT TERMS
Centaur launch vehicle; Electric propulsion; Costs; Hall thrusters; Trajectories; Deep Space Network; Planets; Asteroids

16. SECURITY CLASSIFICATION OF:			17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES	19a. NAME OF RESPONSIBLE PERSON
a. REPORT	**b. ABSTRACT**	**c. THIS PAGE**			STI Help Desk (email:help@sti.nasa.gov)
U	U	U	UU	79	**19b. TELEPHONE NUMBER** *(include area code)* 443-757-5802

Standard Form 298 (Rev. 8-98)
Prescribed by ANSI Std. Z39-18