NASA/TM—2009-215825 CD–2008–28

COMPASS Final Report: Near Earth Asteroids Rendezvous and Sample Earth Returns (NEARER)

Steven R. Oleson and Melissa L. McGuire
Glenn Research Center, Cleveland, Ohio

September 2009

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Cleveland, Ohio 44135

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1.0 EXECUTIVE SUMMARY

In this study, the Collaborative Modeling for Parametric Assessment of Space Systems (COMPASS) team completed a design for a multi-asteroid (Nereus and 1996 FG3) sample return capable spacecraft for the NASA In-Space Propulsion Office. The objective of the study was to support technology development and assess the relative benefits of different electric propulsion systems on asteroid sample return design. The design uses a single, heritage Orion solar array (SA) (~6.5 kW at 1 AU) to power a single NASA Evolutionary Xenon Thruster ((NEXT) a spare NEXT is carried) to propel a lander to two near Earth asteroids. After landing and gathering science samples, the Solar Electric Propulsion (SEP) vehicle spirals back to Earth where it drops off the first sample's return capsule and performs an Earth flyby to assist the craft in rendezvousing with a second asteroid, which is then sampled. The second sample is returned in a similar fashion. The vehicle, dubbed Near Earth Asteroids Rendezvous and Sample Earth Returns (NEARER), easily fits in an Atlas 401 launcher and its cost estimates put the mission in the New Frontier's (NF's) class mission.

Table 1.1 collects the details of the subsystems at a top level in the baseline design (case 1).

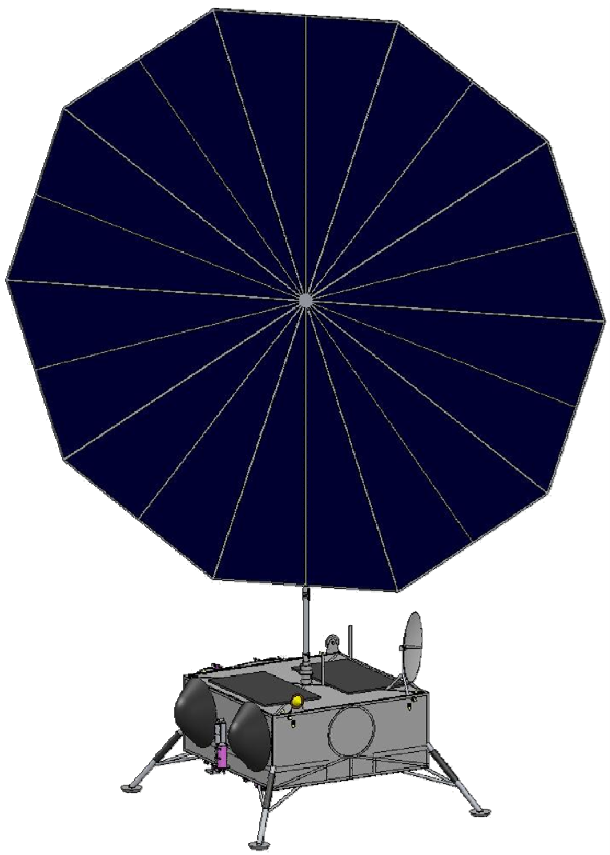


Figure 1.1—NEARER Case 1a Concept Vehicle Design

Table 1.1—Mission and Spacecraft (S/C) Summary—Baseline Case 1

Subsystem area	Details	Total mass with growth
Top level system	SEP enables sample returns from both Nereus and 1996 FG3 Asteroids	1352 kg (wet with growth)
Mission, operations	7 yr mission: December 6, 2014 launch; First science (Nereus landing) June 16, 2016; First sample return February 6, 2018, Second landing (FG3) May 30, 2020; Second sample return October 28, 2021	
Attitude Control System (ACS)	Off-the-shelf (OTS) inertial measurement unit (IMU), Star-trackers, Wheels, hydrazine thrusters LIDAR assisted precision landing system (landing gear for contingency, up to 1m/s landing velocity) Solar pressure torque from off-set solar easily countered by canting electric thrusters <1°	54 kg
Launch	Atlas 401 Launch, C_3 = 38.07 km^2/sec^2 New Frontiers (NF) Expendable Launch Vehicle (ELV), performance to C_3 of 1528 kg (1375 kg after 10% launch margin).	
Science	Science/collection arm with camera, Two, six-bay sample capsules Extensive in-situ science powered by SA Wide/narrow field imager, infrared (IR) spectrometer, laser altimeter, IR, gamma-ray, neutron spectrometer, Alpha Particle X-ray Spectrometer (APXS), LAMS, thermal conductivity, electrical dissipation, ground penetrating radar return samples in two capsules.	29.3 kg payload, (61.4 kg empty capsules)
Power	Single Orion derived Ultra-Flex SA (built for high-g Orion loads), Li-ion batteries for eclipse stays	134 kg
Propulsion	+1 NEXT Ion thrusters (7 m SA), OTS Xe feed and storage system, hydrazine Reaction Control System (RCS), 500 kg Xe for NEXT Cold gas Xe 'landing' system to minimize surface contamination	209 kg
Structures and mechanisms	Thrust tube and tubular space frame propellant tanks mount directly to thrust tube	98 kg
Communications	0.7 m antenna, two axis gimbal hemispheric coverage, 3 to 10 kbps, three omni antennas,	31 kg
Command & data handling (C&DH)	Two RAD750 processors for fault tolerance, 48 Gbit data storage	37 kg
Thermal	Heat-pipe radiators for cooling electronic components Heaters for propellant systems, MLI for S/C	47 kg

2.0 STUDY BACKGROUND AND ASSUMPTIONS

2.1 Introduction

The focus of this COMPASS study was to design a S/C and mission which samples at least two near Earth asteroids and a returns multiple samples from each to the Earth. This study focused on using SEP to enable sample returns from the Asteroids Nereus and 1996 FG3. Additional science mapping and in-situ science was also sought as a science objective. The design parameters (ELV choice, launch mass, incl. cost) were all designed to fit within a NF Class of mission. The trajectory will utilize an Earth flyby to both return first sample capsule and boost the S/C to the second asteroid target.

A number of Trades to be looked at during the course of the design study were:

- Trade primary SEP systems
  - NEXT
  - BPT4000
  - HiVHAC
- Trade level of in-situ science
- Trade number of asteroids sampled (1 or 2)

2.2 Assumptions

Summary of study assumptions and requirements are shown in Table 2.1.

Table 2.1—Study Assumptions

Subsystem area	Assumptions and study requirements	Critical trades
Top-level	Sampler to orbit and land on two separate Near Earth Asteroids (NEA) Return rock samples and soils Figures of Merit (FOMs): Returned sample mass, # of samples, variety, science data, mission success probability, cost well below NF	All- SEP, all-chemical, chem/SEP split
System	OTS equipment where possible, Technology Readiness Level 6(TRL 6) cutoff 2010, 2014 launch year, Single fault tolerant Mass growth per ANSI/AIAA R-020A-1999 (add growth to make system level 30%)	
Mission summary	Integrated SEP system, Earth-asteroid-Earth-asteroid-Earth, visit/orbit/land/ sample two asteroids, sample one, Earth flyby to return first payload, then sample the second and Earth flyby return, 6.5 yr SEP thrusting, 7 yr round trip Atlas 401 launch, C_3 = 38.07 km^2/sec^2 7 yr mission: December 6, 2014 launch; First science (Nereus landing) June 16, 2016; First sample return February 6, 2018, Second landing (FG3) May 30, 2020; Second sample return October 28, 2021 Return of first sample before second landing lowers risks	
GN&C	Closed loop 'docking' system, wheels for stability, gimbaled EP, 250 m/s secondary, 13500 m/s primary (SEP), option for extended missions after sample return OTS IMU, Star-trackers, Wheels, hydrazine thrusters LIDAR assisted precision landing system (landing gear for contingency, up to 1 m/s landing velocity) Solar pressure torque from off-set solar easily countered by canting electric thrusters <1°	SEP or chemical trajectories, landing or docking or hovering, sample collection scheme, cold gas for proximity ops
Launch Vehicle	Atlas 401 C_3 38.07 km^2/sec^2, 1528 kg Adapter: 4 m LPF Launch loads: Axial 11± 1 g, Lateral .4 ± 1.6g	Atlas V, trade adaptors
Propulsion	Primary: 1+1 NEXT (7 m SA), 2+1 4.5 kW BPT-4000, four off the shelf Xe tanks Secondary: blow-down hydrazine RCS system, 1 lbf thrusters Terminal landing: 500 kg Xe cold gas (reduce contamination)	Trade: 1+1 7 kW ion, 2+2 4.5 kW BPT-4000, serial PPUs or cross-strapped
Power	Single Orion derived Ultra-flex SA (built for high-g Orion loads), Li-ion batteries for eclipse stays. 6500 W power to propulsion system (with 400 W housekeeping) Batteries for Asteroid and Moons eclipse, Sampling landing (> 9 hr)	Array type, dual gimbals, cell type, battery options, use of SA to allow long stay times on moons
Avionics/ Communications	Science run from central controller (and one spare 0.7 m antenna, two axis gimbal hemispheric coverage, 3 to 10 kbps, three omni antennas, two RAD750 processors for fault tolerance, 48 Gbit data storage	Computer type, X band or Ka band
Thermal & environment	Body mounted radiator (main loads 350 Wth (PPUs), 100 W (transmitters)). Heat-pipe radiators for cooling electronic components Heaters for propellant systems, MLI for S/C Tank heaters, 0.6 to 1.7 AU thermal environment Deep space radiation level at 1.7 AU	
Mechanisms	Science arm/camera/sampler, two-axis 0.3 m antenna, thruster gimbals ± 12 °, docking legs, sample capsule (2.9 km/s entry velocity capability), parachute impact suppression	Landing legs, sample capsule, sampler arm, foam only impact suppression, harpoons
Structures	Primary: Rectangular, 3- by 3-m, truss, Al-Li; Secondary: 4% of stage components; Thrust tube and tubular space frame propellant tanks mount directly to thrust tube	Developing model, need launch loads
Science	Science/collection arm with camera. Two, six-bay Sample Capsules (for total of 12 samples) Extensive in-situ science powered by SA Wide/narrow field imager, IR spectrometer, laser altimeter, IR, gamma-ray, neutron spectrometer, APXS, LAMS, thermal conductivity, electrical dissipation, ground penetrating radar	

2.3 Growth, Contingency and Margin Policy

Mass Growth: The COMPASS team uses the ANSI/AIAA R-020A-1999, *Recommended Practice for Mass Properties Control for Satellites, Missiles, and Launch Vehicles* (ref. 1). Table 2.2 shows the Percent Mass Growth separated into a matrix specified by level of design maturity and specific subsystem. Mass Growth Allowance (MGA) is defined as the predicted change to the basic mass of an item based on an assessment of the design maturity and fabrication status of the item, and an estimate of the in-scope design changes that may still occur.

The percent growth factors are applied to each subsystem, after which the total system growth of the design is calculated. An additional growth is carried at the system level in order to add up to a total system growth of 30% of the dry mass of the system. Note that growth in propellant is either carried in the propellant calculation itself or in the ΔV used to calculate the propellant required to fly a mission.

Power Growth: The COMPASS team uses a 30% margin on the bottoms up power requirements in modeling the power system. See Sections 3.1.2 and 5.4 for the power system assumptions.

Table 2.2—Percent Mass Growth Allowance

Code	Design Maturity (Basis for Mass Determination)	Percent Mass Growth Allowance									
		Electrical/Electronic Components			Structure	Thermal Control	Propulsion	Batteries	Wire Harnesses	Mechanisms	Instrumentation
		0-5 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	**Pre-Release Drawings** (or minor modification of existing hardware)	20	15	10	8	8	8	10	25	8	25
C	**Released Drawings** (calculated values)	10	5	5	4	4	4	5	5	4	5
X	**Existing Hardware** (actual mass from another program)	3	3	3	2	2	2	3	3	2	3
A	**Actual Mass** (measured flight hardware)	0	0	0	0	0	0	0	0	0	0
CFE	**Customer Furnished Equipment**	0	0	0	0	0	0	0	0	0	0

2.4 Redundancy Assumptions

- Single fault tolerant where possible in the design of the subsystems.
- Exceptions
  - SA
  - Propellant tanks
  - Radiators (design can be modified at ~11 kg penalty)
  - Sampling arm

2.5 Mission Description

This mission returned samples from two asteroids to the surface of the Earth. The asteroids 4660 Nereus and 1996 FG3 were chosen as scientifically desirable asteroids for this mission.

2.5.1 Mission Analysis Assumptions

Earth asteroids named after the first of their type discovered in 1862 (Apollo). Their heliocentric orbital semi major axes are greater than that of Earth, and their perihelion distance is greater than 1 AU (i.e., the distance the Earth is from the sun). The orbits of the Apollo asteroids cross the orbits of both that of Asteroid and Earth. Because Nereus' orbit frequently comes very close to Earth, it is very accessible from Earth as well as a potential threat to the Earth. Due to its small size (approximately 1 km diameter) and hence smaller mass, its ΔV for rendezvous is smaller than the ΔV for rendezvous with our Moon. Nereus has a roughly 15 hr rotation. The asteroid Nereus trajectory details, orbital elements and assumptions are shown in Figure 2.1.

4660 Nereus (1982 DB)

Classification: Apollo [NEO, PHA] **SPK-ID:** 2004660

[Ephemeris | Orbit Diagram | Orbital Elements | Physical Parameters | Discovery Circumstances]

[show orbit diagram]

Orbital Elements at Epoch 2454800.5 (2008-Nov-30.0) TDB
Reference: JPL 49 (heliocentric ecliptic J2000)

Element	Value	Uncertainty (1-sigma)	Units
e	.36019244401616	4.7069e-09	
a	1.488583000071747	4.7973e-09	AU
q	.9524066511549965	6.1256e-09	AU
i	1.43274203863557	3.5926e-06	deg
node	314.5039316193447	6.8718e-05	deg
peri	157.9778666089343	7.4339e-05	deg
M	282.9388465725897	7.7262e-06	deg
t_p	2454942.500872761549 (2009-Apr-21.00087277)	1.4858e-05	JED
period	663.3733330024928	3.2068e-06	d
	1.82	8.78e-09	yr
n	.5426808436368774	2.6234e-09	deg/d
Q	2.024759348988497	6.5253e-09	AU

Orbit Determination Parameters

# obs. used (total)	395
# delay obs. used	11
# Doppler obs. used	2
data-arc span	8290 days (22.70 yr)
first obs. used	1981-09-30
last obs. used	2004-06-11
planetary ephem.	DE405
SB-pert. ephem.	SB405-CPV-2
quality code	0
fit RMS	.37974
data source	ORB
producer	OSOD/JPL
solution date	2008-Mar-04 11:12:41

Additional Information

Earth MOID = .00331794 AU
T_jup = 4.493

[show covariance matrix]

[Ephemeris | Orbit Diagram | Orbital Elements | Physical Parameters | Discovery Circumstances]

Physical Parameter Table

Parameter	Symbol	Value	Units	Sigma	Reference	Notes
absolute magnitude	H	18.2	mag	1.0846	49	autocmod
rotation period	rot_per	15.1	h	n/a	IAU Lightcurve Data (A.W. Harris, E-mail Mar. 2006)	**Publishe** [Ishibashi, [Ishibashi, [Delbo, M.
SMASSII spectral type	spec_B	Xe		n/a	Binzel et al. (2004) Planet. Space Sci. 52, 291-296	

Figure 2.1—4660 Nereus Body Orbital Details

175706 (1996 FG3)

Classification: Apollo [NEO, PHA] **SPK-ID:** 2175706

[Ephemeris | Orbit Diagram | Orbital Elements | Physical Parameters | Discovery Circumstances]

[show orbit diagram]

Orbital Elements at Epoch 2454800.5 (2008-Nov-30.0) TDB
Reference: JPL 13 (heliocentric ecliptic J2000)

Element	Value	Uncertainty (1-sigma)	Units
e	.3498722316799864	1.7249e-07	
a	1.054271178623885	2.571e-08	AU
q	.685410968562857	1.6656e-07	AU
i	1.990349699461728	4.1869e-06	deg
node	299.8810582732225	0.00013428	deg
peri	23.93035757621093	0.00014157	deg
M	150.9185875210075	0.0001535	deg
t_p	2454634.744801032776 (2008-Jun-17.24480104)	0.00016264	JED
period	395.3911351038489	1.4463e-05	d
	1.08	3.96e-08	yr
n	.9104908229807593	3.3306e-08	deg/d
Q	1.423131388684914	3.4705e-08	AU

Orbit Determination Parameters

# obs. used (total)	364
data-arc span	4319 days (11.82 yr)
first obs. used	1996-03-24
last obs. used	2008-01-20
planetary ephem.	DE405
SB-pert. ephem.	SB405-CPV-2
quality code	1
fit RMS	.63184
data source	ORB
producer	Otto Matic
solution date	2008-Feb-22 13:55:02

Additional Information

Earth MOID = .0283384 AU
T_jup = 5.778

[show covariance matrix]

[Ephemeris | Orbit Diagram | Orbital Elements | Physical Parameters | Discovery Circumstances]

Physical Parameter Table

Parameter	Symbol	Value	Units	Sigma	Reference	Notes
absolute magnitude	H	18.211	mag	.88714	13	autocmod 2.4a
rotation period	rot_per	3.5942	h	n/a	IAU Lightcurve Data (A.W. Harris, E-mail Mar. 2006)	Suspected binary object, **Published Reference Li** [Mottola, S., Lahulla, F.: [Pravec, P., Sarounova, [Pravec, P., plus 56 co-a
SMASSII spectral type	spec_B	C		n/a	EAR-A-5-DDR-TAXONOMY-V4.0	based on a high-resolutio

Figure 2.2—1996 FG3 Body Orbital Details

The second asteroid chosen, 1996 FG3, is another Apollo, near Earth asteroid. Initial readings show that this asteroid is the dominant part of a binary asteroid system. 1996 FG3's period of rotation has been determined to be about 3.5 hr and the orbital period of its satellite has been determined at 16 hr. The average bulk density of the asteroid has been estimated at 1.4 g cm^3 and that the surface has a rubble pile structure. 1996 FG3 trajectory details, orbital elements and assumptions are shown in Figure 2.2.

2.5.2 Mission Analysis Analytic Methods

The trajectory design for this mission was optimized using the Mission Analysis Low-thrust Trajectory Optimization (MALTO) tool. The baseline mission launches to a C_3 of 38.07 km^2/s^2 and performs a rendezvous with Nereus in June of 2016, stays at the target for two months for sample collection operations and then departs for Earth to drop the first sample return capsule (SRC) with a constrained entry velocity and position. After the sample is released, the S/C completes the Earth flyby and arrives at the second target, 1996 FG3, in May of 2020. Following the two months at 1996 FG3, the S/C then departs and targets Earth with constrained entry conditions with an arrival V_∞ of 6.8 km/s.

Mission analysis was performed in an iterative fashion. An initial trajectory to the target was performed using MALTO to get the electric propulsion system propellant loading for the missions. With this propellant, the bottoms-up estimation of the vehicle mass was completed by the team. Once this bottoms-up mass was calculated, the trajectory was rerun in order to provide performance for at least that calculated total wet mass. The mission was iterated until the amount of mass pushed by the EP system was greater than or equal to the total wet mass of the vehicle.

2.5.3 Mission Analysis Event Timeline

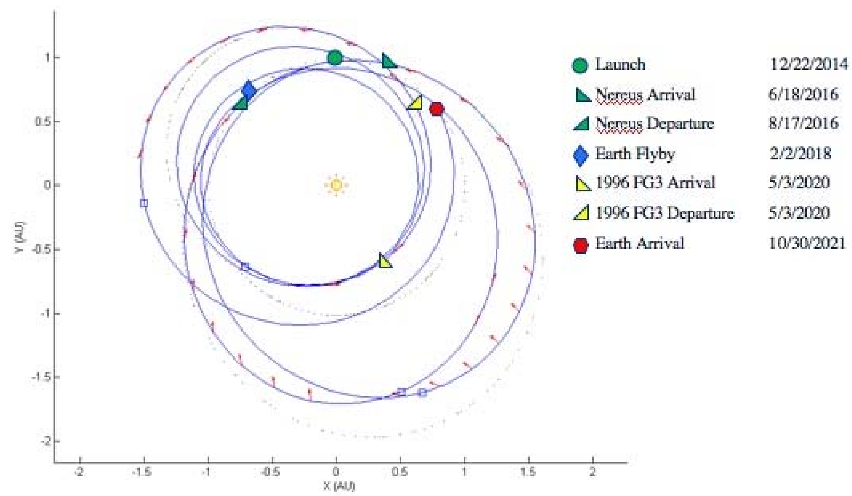


Figure 2.3—Trajectory Main Mission Details

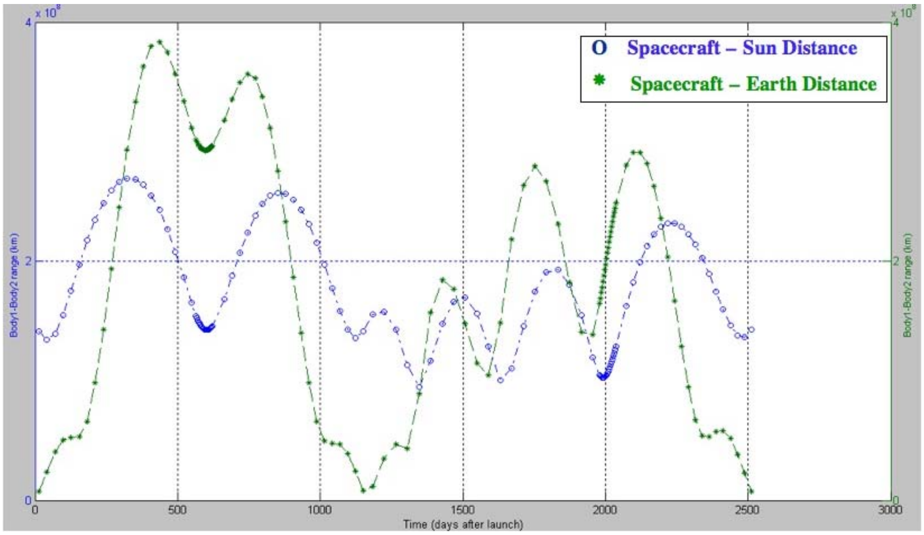


Figure 2.4—S/C Distance From the Sun and Earth Over Mission Time

2.5.4 Mission Trajectory Details

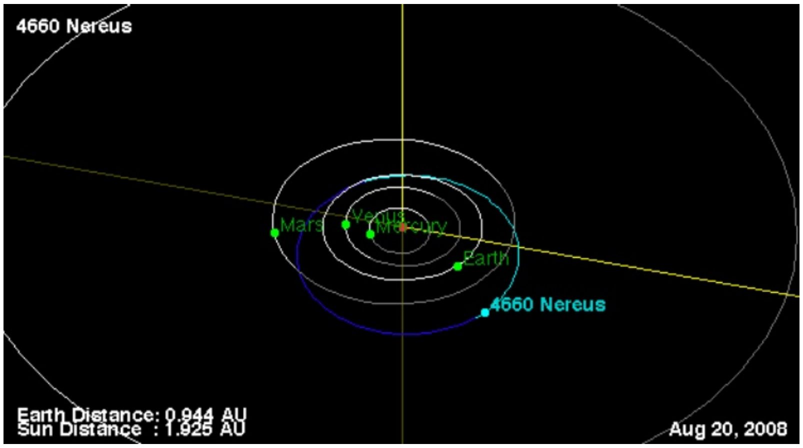


Figure 2.5—4660 Nereus Trajectory Plot

2.6 Concept of Operations (CONOPS)

Remote Sensing (30 days)

- CHAllenging Minisatellite Payload (CHAMP)imaging, radio science to map gravity, LIDAR, neutron detector (find hydrogen)
- High orbit spiral down to
- Low orbit (5 km)
- Asteroids mapped to sufficiency for landing near rock outcropping

Landing (~100 min)

- One burn descent from 0.5 to 1 km starting altitude
- Autonomous landing using maps generated from remote sensing phase, LIDAR and CHAMP imager
  - Impact landing speeds of 25 cm/s
- 10° incline (max), nearby ***large*** boulder (10s to 100s of meters)
- Seeking erosion of large boulder for sampling

Landed Science (Minimum 3.5 hr, Desired 15 hr)

- CHAMP imaging (panoramic and microscopy), APXS, Neutron detector
- Sample acquisition arm/CHAMP imaging of sample area for context of sample
  - Images sent to Earth for review and sample selection
  - Sample acquisition program sent from Earth
  - Samples collected and stored (with confirmation from CHAMP)

Samples Collected

- Rocks eroded from ejecta blocks (from asteroid core)
- Soil samples

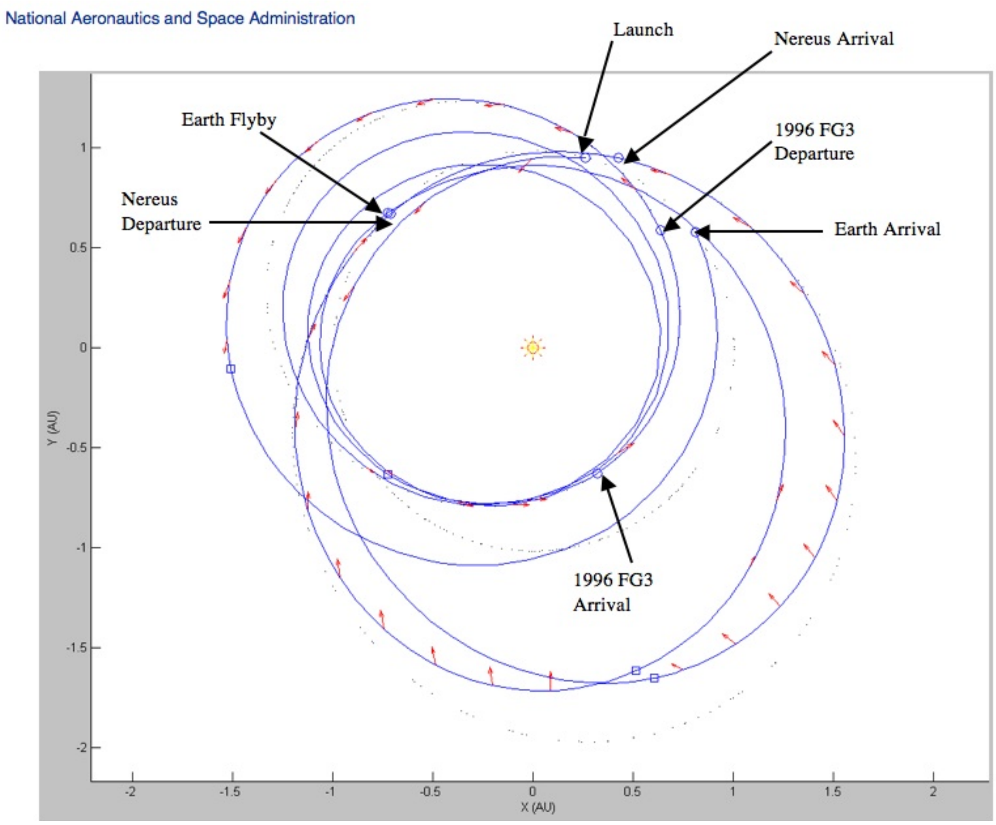


Figure 2.6—Mission Main Events Trajectory Graphic

2.7 Launch Vehicle Details

2.7.1 Launch Vehicle Trade-Space Relative Performance

For this mission, several lower performing launch vehicles were looked at as options. Figure 2.7 shows the relative performance as a function of C_3, of the Delta II, Delta II H and Falcon 9.

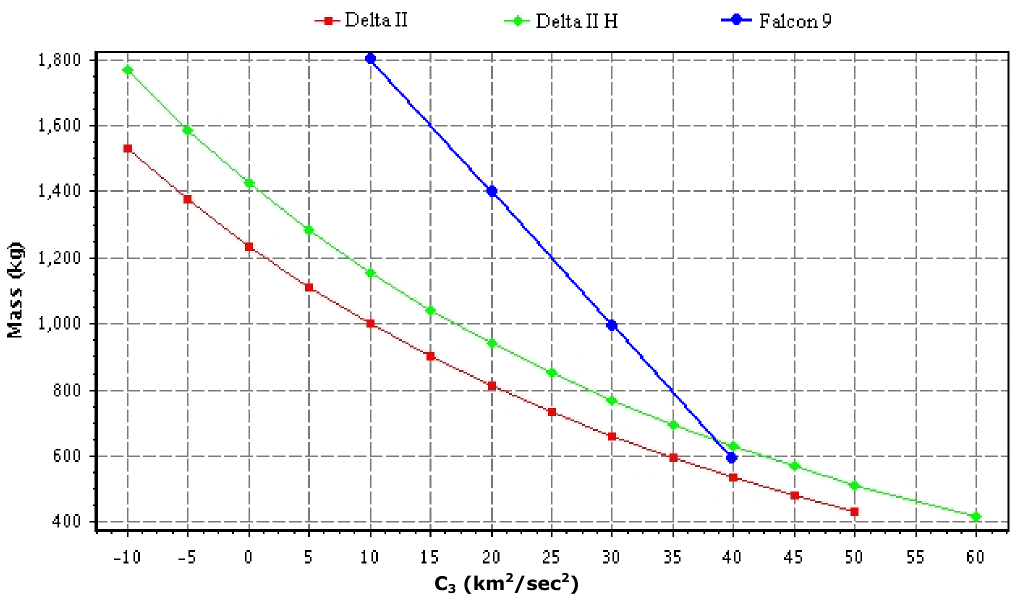


Figure 2.7—Performance Curves for Launch Vehicles of Interest

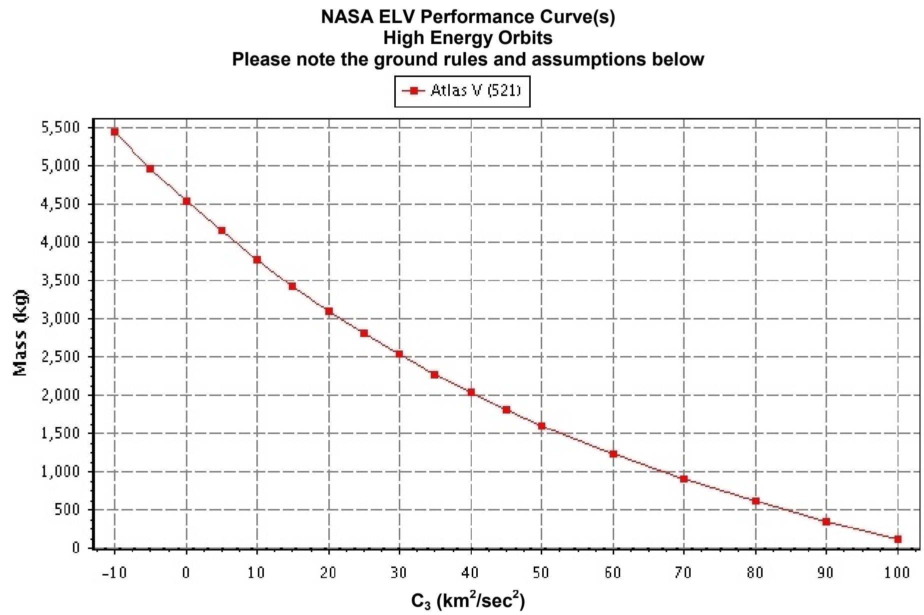
2.7.2 Atlas 401 (4 m Fairing) Performance

From the website astronautix.com, the Atlas V family of launch vehicles offers the performance in Table 2.3. In order to clear up confusion, note that the Atlas 401 is the Atlas V launch vehicle with a 4 m diameter fairing.

Table 2.3—Atlas V family Performance

Configuration	LEO 28°	LEO polar	Geosynch transfer	Geosynch
Atlas V 401	12,500	10,750	5,000	N/A
Atlas V 501	10,300	9,050	4,100	1,500
Atlas V 511	12,050	10,200	4,900	1,750
Atlas V 521	13,950	11,800	6,000	2,200
Atlas V 531	17,250	14,600	6,900	3,000
Atlas V 541	18,750	15,850	7,600	3,400
Atlas V 551	20,050	17,000	8,200	3,750

The Atlas V launch vehicle system is based on the 3.8-m (12.5-ft) diameter Common Core Booster (CCB) powered by a single RD-180 engine. A three-digit naming convention was developed for the Atlas V launch vehicle system to identify it's multiple configuration possibilities, and is indicated as follows: the first digit identifies the diameter class (in meters) of the payload fairing (4 or 5 m); the second digit indicates the number of solid rocket motors used (zero for Atlas V 400 and zero to five for Atlas V 501); the third digit represents the number of Centaur engines. Figure 2.8 shows the performance of the Atlas V (521) versus C_3 from the NASA Kennedy Space Center (KSC) launch performance website. Use of the Atlas 401 for NF missions allows for a raise of the cost cap by $40M.



Assumptions:
Atlas V (521)

This performance does not include the effects of orbital debris compliance, which must be evaluated on a mission-specific basis. This could result in a significant performance impact for mission in which launch vehicle hardware remains in Earth orbit.

3-sigma mission required margin, plus additional reserves as determined by the LSP.

Launch from SLC–41 at Cape Canaveral Air Force Station (CCAFS).

Performance values assume harness, logo, reradiating antenna, three payload fairing doors.

Payload mass greater than 9000 kg (19,841 lb) may require mission unique accommodations. Type B2 payload adapter plus type C2 spacer.

5-m Short Payload Fairing

185 km (100 n mi) minimum park orbit perigee altitude.

185 km (100 n mi) minimum escape orbit perigee altitude.

Performance shown is applicable to declinations between 28.5° and –28.5°.

Figure 2.8—Atlas V 421 Performance Curve.

2.7.3 Atlas 401 Payload Fairing Details

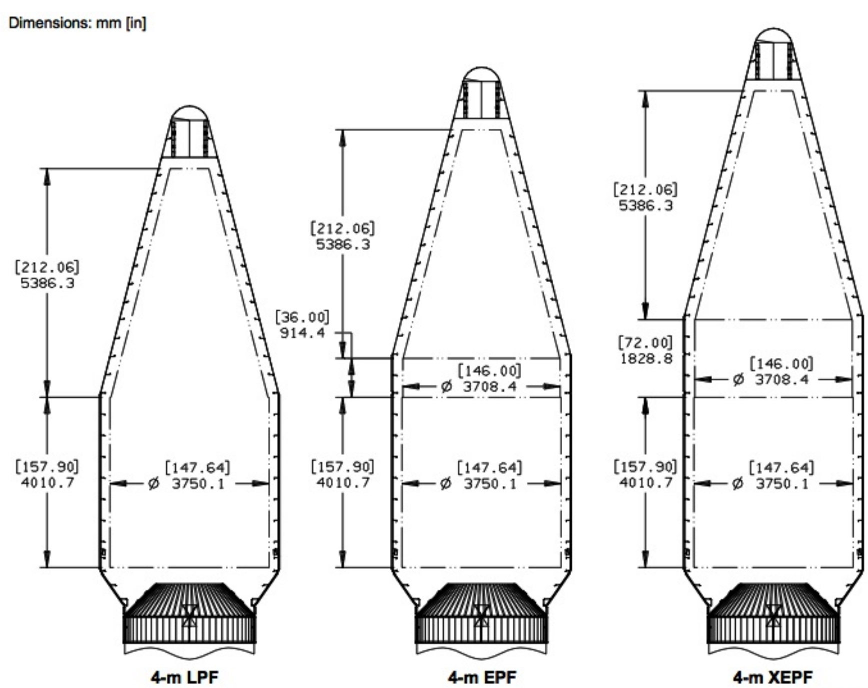


Figure 2.9—Atlas 401 4 m LPF Payload Fairing ELV

2.8 Launch Vehicle Packaging

Using a side-launch configuration allows for the following packaging and concept design benefits.

- Use of smaller, 4-m shroud
- Fixed landing legs
- Lower center-of-mass for more stable asteroid landings
- Eases stowage of large SA
- Long thrust tube attachment of propellant tanks

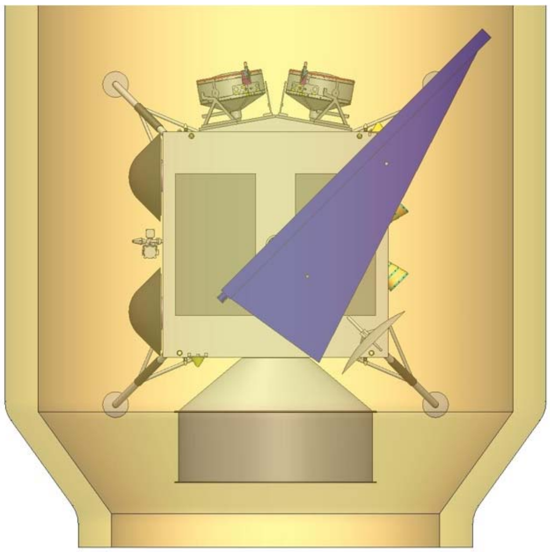


Figure 2.10—NEA Sample Return S/C—Atlas 401 Fairing Packaging Close View

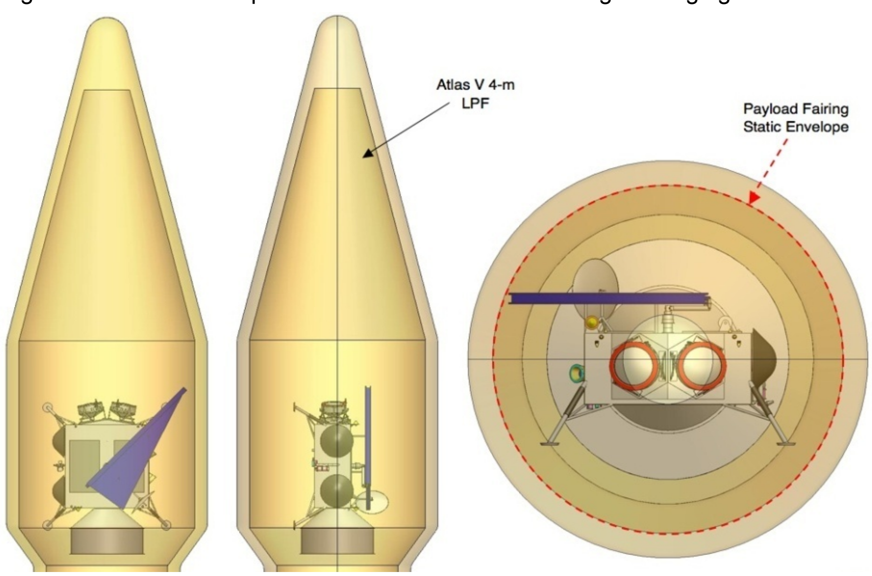


Figure 2.11—NEA Sample Return S/C—Atlas 401 Fairing Envelop and Packaging View

2.9 Sample Return Capsule (SRC) System Level Summary

2.9.1 Sample Collection Requirements

Science return of at least nine total viable samples and one spare. Each sample collection container has enough room to hold six samples, for a total of 12 samples returned (six from each asteroid). Given a total of 12 chances to get a sample worth studying, the following requirements were levied on the sample collection containers.

- Desired sample traits—nine total (100 g each) + one spare
  - Density ~1.7 g/cm^3
  - Portion of 'large blocks'
  - ~4.5 cm diameter, 'golf ball size'
- Mechanisms to collect the sample
  - Four degrees-of-freedom (DOF) collection arm (example in Figure 2.12)
  - 1 m reach
  - Scoop type bifurcated shovel
  - Motorized joints
  - Cable end effecter actuation
  - Sample capsule loading/sealing/separation
  - Swing type carousel

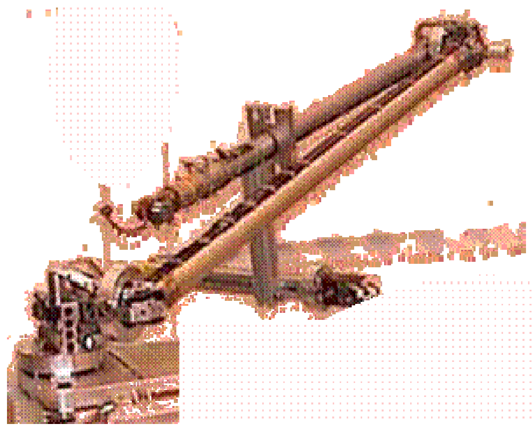


Figure 2.12—Example Science Sampler Collection Arm

2.10 Basic Science Payload Description

Table 2.4 is the MEL of the baseline science payload used in all cases but case 1a in the trade space examined in this study. An additional few elements are carried in a super science package included in case 1 and detailed in the trade studies Section 8.0.

Table 2.4—Science Package Portion of the MEL

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			1138.27	9.9%	113.05	1251.32
06.1	**Science Payload**			25.20	16.1%	4.06	29.26
06.1.1	**Arm Mounted Science Instruments**			1.40	21.4%	0.30	1.70
06.1.1.a	Panoramic / microscopic color imager (JPLÕs CHAMP)	1	1.00	1.00	20.0%	0.20	1.20
06.1.1.b	Alpha Particle X-ray Spectrometer (U. GuelphÕs APXS)	1	0.4	0.40	25.0%	0.10	0.50
06.1.1.c	Misc #3	0	0.0	0.00	0.0%	0.00	0.00
06.1.1.d	Misc #4	0	0.0	0.00	0.0%	0.00	0.00
06.1.1.e	Misc #5	0	0.0	0.00	0.0%	0.00	0.00
06.1.1.f	Misc #6	0	0.0	0.00	0.0%	0.00	0.00
06.1.2	**Body Mounted Science Instruments**			23.80	15.8%	3.76	27.56
06.1.2.a	Approach/Hazard Avoidance/Landing Lidar (OptechÕs C/	1	20.00	20.00	15.0%	3.00	23.00
06.1.2.b	Neutron Detector/Gamma Ray Spect. (IKIÕs HEND)	1	3.80	3.80	20.0%	0.76	4.56
06.1.2.c	Misc #3	0	0.00	0.00	0.0%	0.00	0.00
06.1.2.d	Misc #4	0	0.00	0.00	0.0%	0.00	0.00

Below in the following two sections are short, bulleted description of the science instruments used in the baseline payload.

2.10.1 Arm Mounted (~1 m) Instruments

The following instruments were mounted on the sample collection arm.

- Panoramic/microscopic color imager (NASA Jet Propulsion Laboratory's (JPLs) CHAMP, used on orbit/surface)
  - 1 kg, 30.1- by 13.1- by 9.5-cm, $Power_{peak}$ = 7 W, 3 µm/pixel, 0.4 mrad from orbit, 120 Mb/day, 10 GB internal storage
- APXS (U. Guelph, used on surface)
  - 0.2 kg sensor head (on arm) 7- by 5-cm diameter, 0.2 kg electronics (on S/C) (20- by 10- by 1-cm), $Power_{peak}$ 2.5 W (30 V), 32 kB/s

2.10.2 Body Mounted Instruments

The following instruments are mounted on the body of the S/C.

- Approach/hazard avoidance/landing LIDAR (Optech's Canadian Asteroid Exploration LIDAR for Orbital Topometry–2 (CAMELOT–2) Canadian Space Agency (CSA) contribution)
  - 20 kg, 0.0225 m^3, $Power_{peak}$ = 140 W, Data $Rate_{peak}$ = 500 kbps

Lastly, a potential contribution to the list of science instruments is the neutron detector/gamma ray spectrometer below. Note that it does not appear in the science payload Some of the science instruments were book-kept in other subsystems due to their dual use. Specifically, the laser altimeter sensor listed in the Remote Sensing section of Table 2.5 lists the science instruments as chosen by the science team at the APL. Table 2.5 lists how those instruments were grouped by the science payload planners in a bottoms-up science instrument MEL. Color-coding of blue, yellow and aqua group the science elements into like sensing instruments categories. These instruments were then regrouped in the baseline science MEL in Table 2.4.

Table 2.4 is book-kept in the Guidance, Navigation and Control subsystem MEL. The items in light blue are grouped together on the arm. Items in light green are grouped together in the spectrometers line item in the MEL. The shared DPU is book-kept in the C&DH system MEL Table 2.5 or the MEL in Table 2.4

- Neutron Detector (ND)/Gamma Ray Spectrometer (IKI's HEND) (possible contribution—used in orbit)
  - Mass = 3.8 kg, dimensions: 25 by 15 by 15 cm, $Power_{peak}$ =8 W, Data rate: 1 kB/frame

2.11 Super Science Package Description

Table 2.5 lists the science instruments as chosen by the science team at the APL. Table 2.5 lists how those instruments were grouped by the science payload planners in a bottoms-up science instrument MEL. Color-coding of blue, yellow and aqua group the science elements into like sensing instruments categories. These instruments were then regrouped in the baseline science MEL in Table 2.4.

Table 2.5—APL Science Payload

	Instrument	Heritage/Analog	Mass (kg)	Power (W)
Remote Sensing	Wide/Narrow Field Imager	MESSENGER (MDIS)	3.5	4.2
	IR Spectrometer	MESSENGER (MASCS?)	3.1	6.7
	Laser altimeter	NEAR	5	15
	Neutron Spectrometer	MESSENGER (GRNS)	4	5
	Gamma-Ray Spectrometer	MESSENGER (GRNS)	8	16
Surface Sensing	APXS	MER	2	2
	LAMS		4	6
	Microscopic Imager		1	0.6

	Instrument	Heritage/Analog	Mass (kg)	Power (W)
	Thermal Conductivity	Rosetta et al.	0.5	0.5
	Electrical Dissipation		1.5	3
	Ground Penetrating Radar		8	30
		Total:	40.6 kg	89.0 W
Support System	Robotic Arm		6	8
	Shared DPU	MESSENGER	3.6	6
		Total:	50.2 kg	103.0 W

Some of the science instruments were book-kept in other subsystems due to their dual use. Specifically, the laser altimeter sensor listed in the Remote Sensing section of Table 2.5 lists the science instruments as chosen by the science team at the APL. Table 2.5 lists how those instruments were grouped by the science payload planners in a bottoms-up science instrument MEL. Color-coding of blue, yellow and aqua group the science elements into like sensing instruments categories. These instruments were then regrouped in the baseline science MEL in Table 2.4.

Table 2.4 is book-kept in the Guidance, Navigation and Control subsystem MEL. The items in light blue are grouped together on the arm. Items in light green are grouped together in the spectrometers line item in the MEL. The shared DPU is book-kept in the C&DH system MEL Table 2.5 or the MEL in Table 2.4

Table 2.5—NEARER Baseline Science Payload Details
(Adapted From Applied Physics Laboratory (APL) Science Payload)

NEARER Science Payload	Heritge/Analog	Mass (kg)	Power (W)
Arm Mounted Science Instruments		**5.4**	
Panoramic / microscopic color imager (JPL's CHAMP)	CHAMP	1	
Alpha Particle X-ray Spectrometer (U. Guelph's APXS)	APXS	0.4	
LAMS		4	
Misc #4			
Misc #5			
Misc #6			
Body Mounted Science Instruments		**48.6**	
Approach/Hazard Avoidance/Landing Lidar (Optech's CAMELOT-2)	CAMELOT-2	**20**	
Spectrometers: IR, Neutron, Gamma-Ray (fm Trojan Lander)	MESSENGER	**15.1**	
IR Spectrometer	MESSENGER	3.1	6.7
Neutron Spectrometer	MESSENGER	4	5
Gamma-Ray Spectrometer	MESSENGER	8	16
Wide/Narrow Field Imager	MESSENGER	**3.5**	4.2
Thermal Conductivity (Rosetta), Electrical Dissipation, Ground Penetrating Radar		**10**	
Thermal Conductivity	Rosetta et al.	0.5	0.5
Electrical Dissipation		1.5	3
Ground Penetrating Radar		8	30
	Total	**54**	

Table 2.7 is the MEL used in the super science case 1a described in more detail in the trade space section of this report.

Table 2.6—Super Science Package Portion of the MEL

WBS Number	Description NEA Sampler (Sept. 2008) - Case 1a	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			**1281.47**	**9.5%**	**122.06**	**1403.53**
06.1	**Science Payload**			**54.00**	**11.7%**	**6.32**	**60.32**
06.1.1	**Arm Mounted Science Instruments**			**5.40**	**5.6%**	**0.30**	**5.70**
06.1.1.a	Panoramic / microscopic color imager (JPL's CHAMP)	1	1.00	1.00	20.0%	0.20	1.20
06.1.1.b	Alpha Particle X-ray Spectrometer (U. Guelph's APXS)	1	0.4	0.40	25.0%	0.10	0.50
06.1.1.c	LAMS	1	4.0	4.00	0.0%	0.00	4.00
06.1.1.d	Misc #4	0	0.0	0.00	0.0%	0.00	0.00
06.1.1.e	Misc #5	0	0.0	0.00	0.0%	0.00	0.00
06.1.1.f	Misc #6	0	0.0	0.00	0.0%	0.00	0.00
06.1.2	**Body Mounted Science Instruments**			48.60	12.4%	6.02	54.62
06.1.2.a	Approach/Hazard Avoidance/Landing Lidar (Optech's CAMELOT-2)	1	20.00	20.00	15.0%	3.00	23.00
06.1.2.b	Spectrometers: IR, Neutron, Gamma-Ray (fm Trojan Lander)	1	15.10	15.10	20.0%	3.02	18.12
06.1.2.c	Wide/Narrow Field Imager	1	3.50	3.50	0.0%	0.00	3.50
06.1.2.d	Thermal Conductivity (Rosetta), Electrical Dissipation, Ground Penetrating Radar	1	10.00	10.00	0.0%	0.00	10.00

3.0 BASELINE DESIGN—CASE 1

3.1 Top Level Design (MEL and PEL)

3.1.1 Master Equipment List (MEL)

Table 3.1 lists the MEL of the design for only the top level masses. The total growth on the dry mass of the S/C is then rolled up to find a total growth mass and growth percentage. The Growth column is where each subsystem lists the recommended growth factor on each line items following the AIAA WGA schedule outlined in Table 2.4 in Section 2.4. The MEL takes all of the items and racks them up into totals and calculates a total CBE mass, a Total mass and a total Growth Mass.

Table 3.1—Master Equipment List—Case 1

WBS	Description	QTY	Unit Mass	CBE Mass	Growth	Growth	Total Mass
Number	NEA Sampler (August 2008) - Case 1		(kg)	(kg)	(%)	(kg)	(kg)
06	NEA Sampler Spacecraft			1138.27	9.9%	113.05	1251.32
06.1	Science Payload			25.20	16.1%	4.06	29.26
06.2	Lander Spacecraft			1019.77	9.0%	92.07	1111.85
06.2.1	Attitude Determination and Control			44.28	21.1%	9.36	53.64
06.2.2	Command and Data Handling			29.30	27.2%	7.96	37.26
06.2.3	Communications and Tracking			24.20	27.1%	6.55	30.75
06.2.4	Electrical Power Subsystem			115.50	15.7%	18.19	133.69
06.2.5	Thermal Control (Non-Propellant)			40.11	18.0%	7.22	47.33
06.2.6	Propulsion			192.10	8.3%	16.00	208.09
06.2.7	Propellant			425.41	0.0%	0.00	425.41
06.2.8	Structures and Mechanisms			148.88	18.0%	26.80	175.68
06.3	Sample Return Craft			93.30	18.1%	16.91	110.21
06.3.1	Electrical Power Subsystem			6.00	20.0%	1.20	7.20
06.3.2	Thermal Control (Non-Propellant)			4.60	18.0%	0.83	5.43
06.3.3	Structures and Mechanisms			82.70	18.0%	14.89	97.59

The MEL (Table 3.1) captures the bottoms up estimation of CBE and growth percentage line item by item for each subsystem. Table 3.3 wraps up those total masses, CBE and total mass after applied growth percentage. In order to meet the total of 30% at the system level, an allocation is necessary for system level growth. This additional system level mass is assumed as part of the inert mass that is flown along the required trajectory. Therefore, the additional system level growth mass impacts the total propellant loading for the mission design.

Using the low thrust trajectory tool MALTO, and a starting guess of delivered target mass set to the bottoms up mass of the Case 1, the performance to the C_3 of 38.07 km/s is calculated from a look up table to be 1580 kg (see 2.5.2 for mission analysis assumptions, and 2.7.1 for Launch Vehicle assumptions). A 10% margin is taken off of this performance, leaving 1375 kg available launch performance. This mass of 1375 is what the SEP system flies as the wet mass of the S/C, and the xenon propellant loading is sized to push the total mass of 1375 kg. Launch vehicle margin is the mass that is not flown with the spacecraft.

Performance and margin on the launch vehicle are calculated as follows and shown on Table 3.2.

$$\text{ELV Perfomance to } C_3 = A = 1528\,\text{kg}$$
$$\text{Margin} = A * 10\% = 152\,\text{kg}$$
$$\text{Available ELV performance} = A - (A * 10\%) = 1375\,\text{kg}$$

The total bottoms-up wet mass as shown in Table 3.1 of the system before the additional system mass is carried is 1251 kg (alsosee the row marked Estimated S/C Wet Mass and look in the column for Total Mass in Table 3.2). The total bottoms-up growth in this mass is 113 kg, or 16% of the total dry mass. In order to meet the 30% dry mass growth requirement at the system level, an additional 101 kg system mass is carried as shown in Table 3.2. This brings the total wet mass with growth to 1352 kg. An additional 23 kg is available in this bottom's up modeling over the 10% launch vehicle mass and can be added to the 1352 wet mass to bring the system level growth up from 30%. Each SRC's mass with 30% growth applied is 61 kg. Each SRC can return up to 1kg worth of sample material.

Table 3.2—System Integration Summary—Case 1

Spacecraft Master Equipment List Rack-up (Mass)				COMPASS S/C Design	
WBS	**Main Subsystems**	**CBE Mass (kg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
01	**Asteroids Sampler Spacecraft**	**1138.3**	**113.0**	**1251.3**	
06.1	Science Payload	25.2	4.1	**29.3**	16%
06.2	*Asteroids Sampler Lander*	***1019.8***	***92.1***	***1111.8***	
06.2.1	Attitude Determination and Control	44.3	18.7	**53.6**	42%
06.2.2	Command and Data Handling	29.3	8.0	**37.3**	27%
06.2.3	Communications and Tracking	24.2	6.5	**30.7**	27%
06.2.4	Electric Power	115.5	18.2	**133.7**	16%
06.2.5	Thermal Control	40.1	7.2	**47.3**	18%
06.2.6	Propulsion	192.1	16.0	**208.1**	8%
06.2.7	Propellant	425.1			
06.2.8	Structures and Mechanisms	148.9	26.8	**175.7**	18%
06.3	*Sample Return Craft (total, empty)*	***93.3***	***16.9***	***110.2***	18%
06.3.1	Electrical Power Subsystem	6.0	1.2	**7.2**	20%
06.3.2	Thermal Control (Non-Propellant)	4.6	0.8	**5.4**	18%
06.3.3	Structures and Mechanisms	82.7	14.9	**97.6**	18%
	Estimated Spacecraft Dry Mass	**713**	113	**826.2**	**16%**
	Estimated Spacecraft Wet Mass	1138	113	1251.3	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	713	214	**927.1**	**30%**
	Additional Growth (carried at system level)		101		**14%**
	Total Wet Mass with Growth	1138	214	**1352.2**	
	Available Launch Performance to C3 (kg)			1375.4	
	Launch margin available (kg)			**23.2**	
	Estimated Spacecraft Inert Mass (for traj.)	808	214	**1021.5**	
	Sample Return Craft Total Mass	CBE Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
	Estimated Sample Return Craft Mass	***93.3***	***16.9***	***110.2***	18%
	Total with System Level Growth	93	28	**121.3**	**30%**
	Number of Sample Return Craft	**2**			
	Total Mass per Sample Return Craft (empty)	**60.6**	kg		
	Total Mass, Sample Returned	1	kg		
	Total Mass, Sample Return Capsule (Full)	**61.6**	kg		

3.1.2 Power Equipment List (PEL)

Power mission operations modes assumed: Peak, nominal, and standby. The power system data was provided by science, GN&C, avionics, communications, and thermal subsystems. The power required by the power subsystem (conversion boxes, line losses) was kept internally to the power design and is not shown in the PEL. Peak power was assumed for propulsion subsystem during SEP Thrusting and the asteroid science mapping mission events. The panoramic microscopic color imager (JPL's CHAMP) and LIDAR were used during approach and landing phase. The APXS (U. Guelph's) and Neutron Detector/Gamma Ray Spectrometer (IKI's HEND) were used during Landed Science Phase. Reuse of the SEP solar power for science and operations was assumed. Battery size was based on asteroid eclipse times: ~8 hr maximum. Growth of 30% was assumed for power needs (except for Electric Propulsion).

The power required for nominal loads, based on mission modes of operation, is shown in Table 3.3. The waste heat used to size the thermal system is shown in the bottom of Table 3.3.

Table 3.3—Power Estimations and Waste Heat Over Mission Modes

	Propulsion, W	Avionics, W	Comm., W	Thermal, W	GN&C, W	Power, W	Science, W	CBE Total, W	30 % Margin, W	Total, W
Launch	0	24	0	33	31	0	0	88	26.37	114
S/C separation	16	24	383	33	31	0	0	487	141.27	628
S/C checkout	16	24	403	33	126	0	221	823	242.13	1065
SEP Thrusting	6350	24	0	33	88	0	2	6497	44.16	6541
SEP Coast	16	24	403	33	98	0	2	576	168.06	744
Communications	16	24	383	33	98	0	2	556	162.06	718
Nereus Targeting	6350	24	0	33	88	0	2	6497	44.16	6541
Nereus Science Mapping	16	24	0	33	98	0	221	392	112.83	505
Nereus Mapping Communications	16	24	383	33	98	0	221	775	227.73	1003
Nereus Approach and Landing	0	24	383	33	98	0	331	869	260.73	1130
Nereus Landed Science	16	24	383	33	98	0	221	775	227.73	1003
Nereus Landed Communications	16	24	403	33	98	0	2	576	168.06	744
Nereus Take-off	16	24	383	33	98	0	2	556	162.06	718
Earth Sample Dropoff/Flyby	16	24	383	33	98	0	2	556	162.06	718
1996FG3 Body Targeting, etc.	16	24	383	33	98	0	2	556	162.06	718
Waste Heat	Propulsion, W	Avionics, W	Comm., W	Thermal, W	GN&C, W	Power, W	Science, W	CBE Total, W	30 % Margin, W	Total, W
Launch	0.0	1.2	0.0	1.7	1.6	0.0	0.0	4.4	1.3	5.7
S/C separation	0.8	1.2	19.2	1.7	1.6	0.0	0.0	24.3	7.3	31.6
S/C checkout	0.8	1.2	20.2	1.7	6.3	0.0	11.0	41.2	12.3	53.5
SEP Thrusting	443.4	1.2	0.0	1.7	4.4	0.0	0.1	450.7	135.2	586.0
SEP Coast	0.8	1.2	20.2	1.7	4.9	0.0	0.1	28.8	8.6	37.5
Communications	8.0	12.0	191.5	16.5	49.2	0.0	1.0	278.1	83.4	361.5
Nereus Targeting	443.4	1.2	0.0	1.7	4.4	0.0	0.1	450.7	135.2	586.0
Nereus Science Mapping	0.8	1.2	0.0	1.7	4.9	0.0	11.0	19.6	5.9	25.5
Nereus Mapping Communications	0.8	1.2	19.2	1.7	4.9	0.0	11.0	38.8	11.6	50.4
Nereus Approach and Landing	0.0	1.2	19.2	1.7	4.9	0.0	16.5	43.5	13.0	56.5
Nereus Landed Science	0.8	1.2	19.2	1.7	4.9	0.0	11.0	38.8	11.6	50.4
Nereus Landed Communications	0.8	1.2	20.2	1.7	4.9	0.0	0.1	28.8	8.6	37.5
Nereus Take-off	0.8	1.2	19.2	1.7	4.9	0.0	0.1	27.8	8.3	36.2
Earth Sample Dropoff/Flyby	0.8	1.2	19.2	1.7	4.9	0.0	0.1	27.8	8.3	36.2
1996FG3 Body Targeting, etc.	0.8	1.2	19.2	1.7	4.9	0.0	0.1	27.8	8.3	36.2

3.2 Baseline System Level Summary

- Low center of mass configuration for landing stability
  - 500 kg of Xe in COPV tanks: Four cylindrical OTS COPVs
  - SA (> 5 kW Orion heritage) deployed after launch—used for landed power
  - 5 kW Hall or 7 kW Ion propulsion systems on side of S/C
  - Radiators on top deck
- Minimize deployables/mechanisms (only science, power, and communications)
  - Science/collection arm (1m with 0.5 m telescoping extension)
  - Sample capsule loading/sealing/separation
  - Single axis gimbal SA
  - Two-axis communications antenna

- Power
  - Single Orion derived Ultra-Flex SA (built for high-g Orion loads)
  - Li-ion batteries for eclipse stays
- Propulsion
  - 1+1 NEXT Ion thrusters (7 m SA) , OTS Xe feed and storage system, OTS hydrazine landing system
  - 950 kg Xe BPT-4000 Hall or 500 kg for NEXT
  - Cold Gas Xe ‘landing’ system to minimize surface contamination and hold S/C fixed to surface during sample arm operations (as needed)
- GN&C
  - OTS IMU, Star-trackers, wheels, hydrazine thrusters
  - LIDAR assisted precision landing system
  - Solar pressure torque from off-set solar easily countered by canting electric thrusters <1°
- Avionics/Communications
  - One 0.7 m antenna, two axis gimbal hemispheric coverage, 3 to 10 kbps (Kilobits per second) , three omni antennas
  - Two RAD 750 processors for fault tolerance
- Thermal
  - Heat-pipe radiators for cooling electronic components
  - Heaters for propellant systems
  - MLI for S/C

3.3 Baseline Design Concept Drawing and Description

In order to maintain stability while landed, it is desirable to keep the center of gravity (CG) of the landed configuration as close to the surface as possible while maximizing the diameter of the landed “footprint”. These goals were accomplished by putting the two cylindrical Xe tanks in horizontally with respect to the main bus shelf. Cylindrical tanks generally are not designed to handle large loads in the radial direction very well, thus it was decided to launch the lander on its side (relative to the landed configuration) as shown previously in Figure 2.10, to ensure the launch loads are incurred while the tanks are in their axial direction. This orientation allowed the use of a thrust tube type structural design to handle the high launch loads, encapsulate the spherical hydrazine tanks for the RCS system, and provided a structurally sound mounting point for the Xe tanks.

Two small decks were placed next to the Xe tanks to allow mounting of all the internal science, power, guidance, avionics, and propulsion components. Those guidance and science instruments needed during landing or while on the surface were place on the bottom of the lower deck to allow a clear view of the surface, while all other internal components were mounted on the top of the decks. The space frame structure was used to mount all components external to the S/C including the thrusters, antennas, ultraflex array, radiators, and sample collection and return components, as well as provide good mounting points for the stowed array during launch. All components included in the design, except for the SA, are shown in Figure 3.1 and Figure 3.2. Note that the top of the lander is used to mount the SA and radiators, allowing the array to track the sun while keeping the radiators perpendicular to the sun. The dish and omni antennas are also mounted on the top surface. The bottom is dedicated to surface science and landing guidance. One side is dedicated to the star trackers, another side is dedicated to the thrusters, the third side is dedicated to sample collection and return, while the final side is dedicated to interfacing to the launch vehicle. The NEXT thrusters are mounted to a structure that allows for minimal gimbaling by orienting the thrust vector of each thruster through the CG of the vehicle. Figure 3.3 and Figure 3.4 show the deployed and stowed configurations of the lander while the overall dimensions of the lander are shown in Section 3.4.

The vehicle design was based around a Single Ultraflex array. This design concept has the following benefits:

- Saves mass
- Keeps array away from asteroid/dust
- Can be kept deployed to allow long term landings (hours-days)
- Orion Ultraflex design capable of resisting large forces while deployed

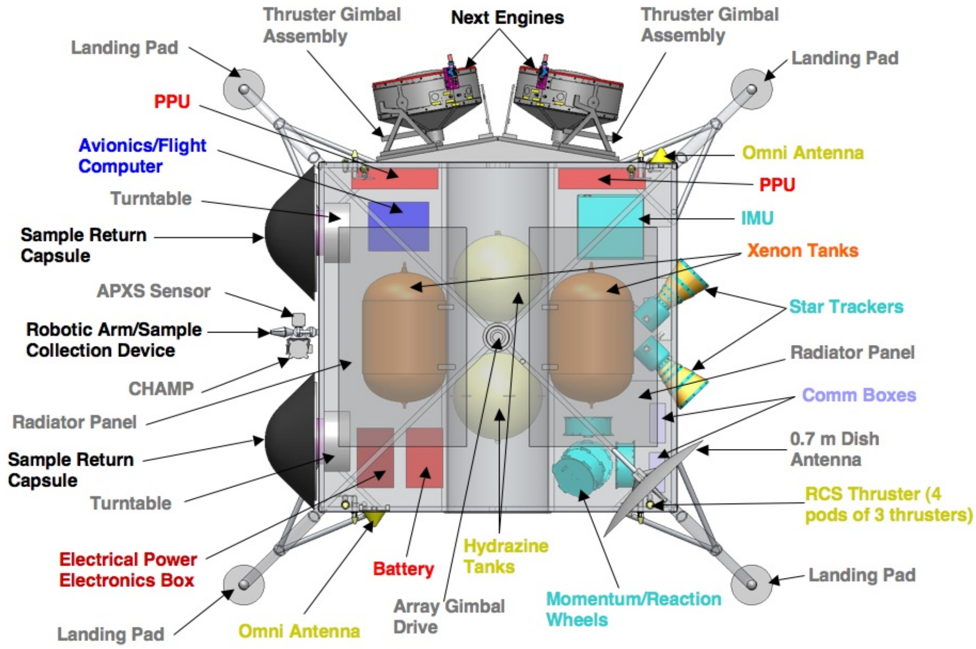


Figure 3.1—NEA Sample Return S/C—Top Components View

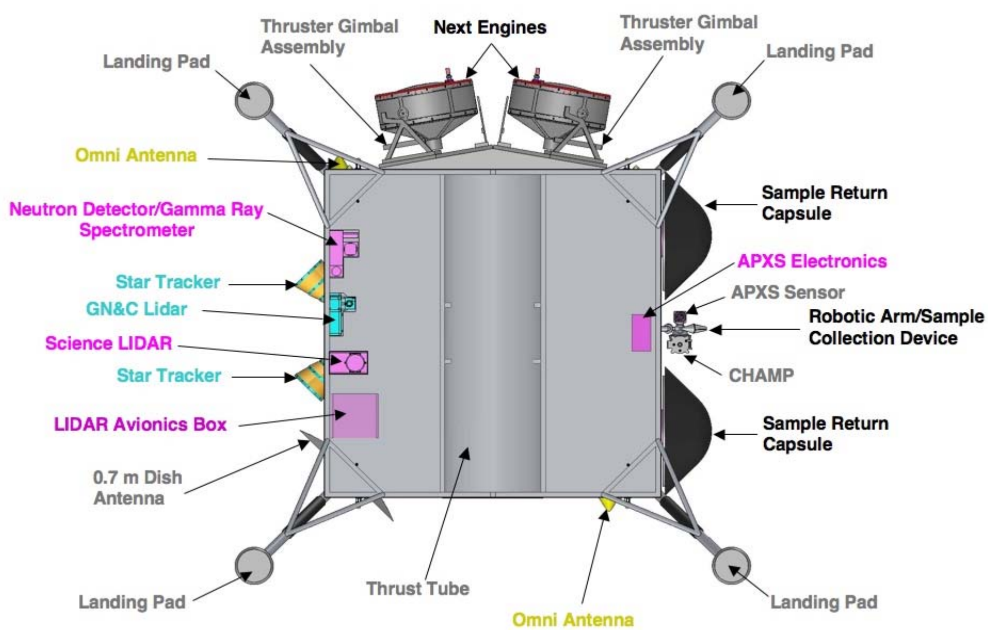


Figure 3.2—NEA Sample Return S/C—Bottom Components View

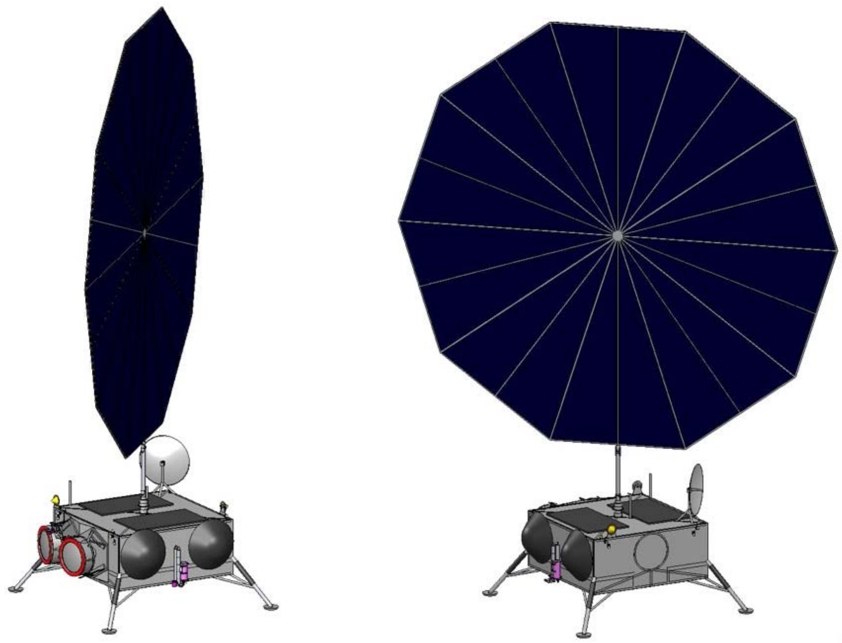


Figure 3.3—NEA Sample Return S/C—Deployed View

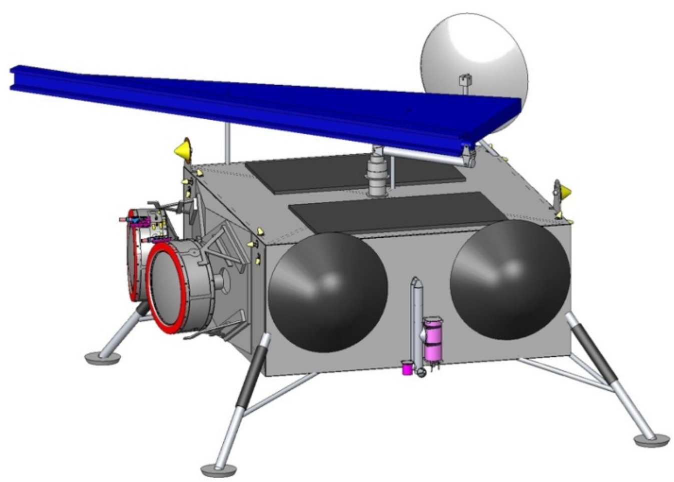


Figure 3.4—NEA Sample Return S/C—Stowed View

3.4 Baseline Design Concept Dimensions

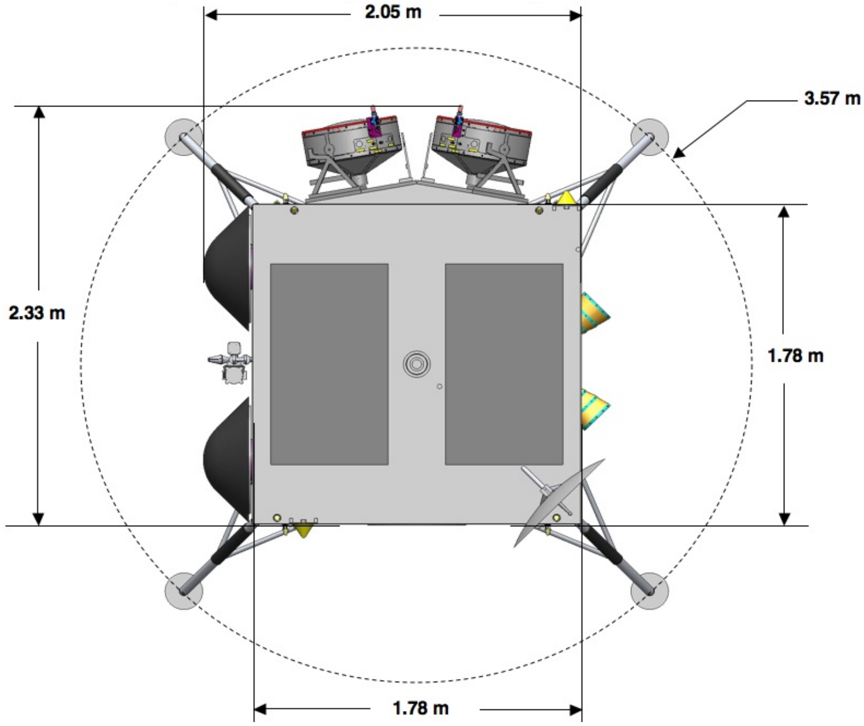


Figure 3.5—NEA Sample Return S/C—Footprint Dimensions

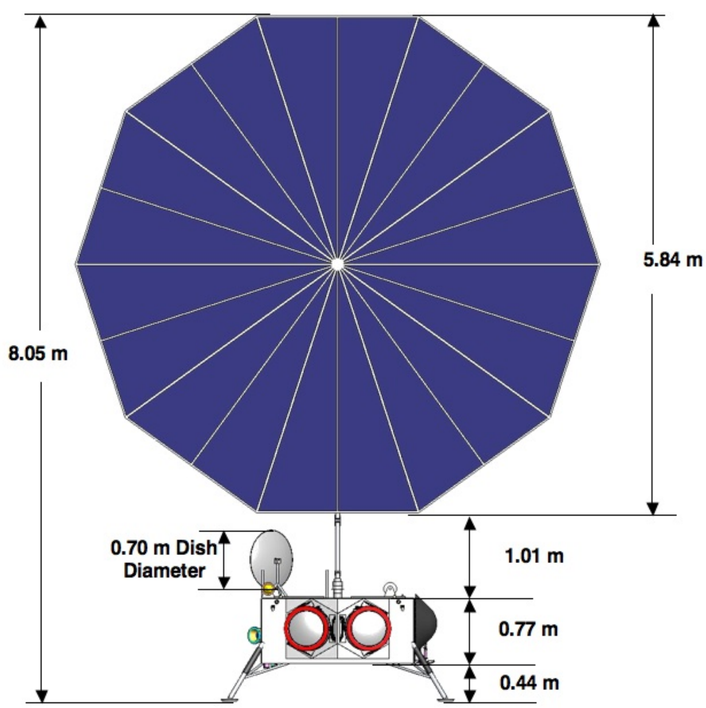


Figure 3.6—NEA Sample Return S/C—Deployed Dimensions

4.0 CHALLENGES, LESSONS LEARNED, AREAS FOR FUTURE STUDY

4.1 Future Work

- Better definition of major challenges
  - Precision landing
  - Securing and unsecuring S/C to surface to allow sampling
  - Hopping for more samples
  - Sample acquisition and storage
  - Asteroid environment (low gravity, dust)
- Trades to reduce costs
  - Simplify science and collection strategy
  - Utlize more OTS systems

4.2 Lessons Learned

Electric Propulsion

- Allows sample returns from two disparate near Earth asteroids
- The large CEV derived SA provides sufficient power to allow for long-term landings for science collection
- Reuse of Xe propellant for terminal landing and contingency 'hopping' avoids hydrazine contamination of surface samples
- Bringing back samples should save costs of in-situ science instruments and operations

5.0 SUBSYSTEM BREAKDOWN

5.1 Communications

This section describes the telecommunications subsystem of NEARER, dealing specifically with communications equipment on board the lander. Major telecommunications subsystem components have been chosen for NEARER (Near Earth Asteroids Rendezvous and sample Earth Returns) in response to the science mission requirements and design considerations such as anticipated maximum distances, desired data rates, on-board power and mass limitations.

5.1.1 Communications Requirements

The high level general requirements on the telecommunications subsystem are to provide the best signal possible in terms of available on-board electrical power, accuracy, reliability, and quality assurance, with constraints on mass, size and costs. Table 5.1 provides some of the important communications subsystem requirements.

Table 5.1—Communications System Requirements

Requirements	Description
Data rates	10 bps to 3 kbps for command (1 kbps typical) 40 bps to 10 kbps for health and status telemetry 10 kbps or higher for mission/science
Daily data volume	Daily data volume shall be at least 120 Mbits/day
Data storage	A minimum of 10 GB (8×10^{10} bits) internal storage shall be required
Frequency	Shall support Asteroid Moons Sampler at X-Band and use 8.44/7.75 GHz for uplink and downlink
Housekeeping and overhead	Housekeeping and any overhead shall include in stated data rates in this Table
Available on-board power	Shall consider different options such as using larger antennas/higher efficiency amplifiers and data requirements.
Equivalent Isotropically Radiated Power (EIRP)	EIRPs shall be as required in order to achieve 10 BER (Bit Error Rate) for given data rates

The assumed requirement: 7 kbits/sec during 8 hr daily communications. The electronics were assumed to be single fault tolerant.

5.1.2 Communications Assumptions

Data transmitted back to Earth will go through the DSN 70-m dishes. The DSN consists of facilities in California's Mojave Desert; near Madrid, Spain; and near Canberra, Australia. These stations are spaced about 120° apart on the globe—making sure any S/C can be observed constantly as Earth rotates. The data sent to the 70 m DSN dishes are transferred to some science ground station.

The highest data rates will come during the mission phases when the Sampler return craft is on the surface of the moons. The sampler S/C will be on the surface of each of the asteroids for a maximum of 8 hr.

The onboard processing and data buffering capabilities of the avionics system will handle data taken by the LIDAR during landing and departure to and from the asteroids.

Orbital Downlink

- Diameter 40 km (5×10^8 m^2)
- m/pixel = 1
- Image overlap 1.5
- Views/site 2
- Effective bits/pix 8
- Colors 5
- Total orbital downlink 10^{11} bits
  - Assumes 60% is imaging data
- Approximately 60 days of downlink

Landed Downlink

- Dominated by microscopic imager
- Estimated total data ~ 8×10^9 bits
- Some realtime data needed for validating surface science operations

5.1.3 Communications Design and MEL

The key components of the telecommunications subsystem include a 0.7 m high gain antenna (HGA) providing two-axis gimbaled hemispheric coverage, two omni-antennas, and two 85-W radio frequency (RF) Traveling Wave Tube Amplifiers (TWTA). The 0.7 m HGA is designed to support data rates from 3 to 10 kbps and the Omni-antennas for emergency at data rates from 10 to 100 bps. The communications MEL is provided in Table 5.2.

Table 5.2—Communications MEL for Baseline (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			1138.27	9.9%	113.05	1251.32
06.1	**Science Payload**			25.20	16.1%	4.06	29.26
06.2	**Lander Spacecraft**			1019.77	9.0%	92.07	1111.85
06.2.1	**Attitude Determination and Control**			44.28	21.1%	9.36	53.64
06.2.2	**Command and Data Handling**			29.30	27.2%	7.96	37.26
06.2.3	**Communications and Tracking**			24.20	27.1%	6.55	30.75
06.2.3.a	**X High Gain Antenna**			23.40	27.5%	6.43	29.83
06.2.3.a.a	Transmitter/Receiver	2	2.90	5.80	30.0%	1.74	7.54
06.2.3.a.b	Power Amp	2	2.60	5.20	30.0%	1.56	6.76
06.2.3.a.c	Switch Unit	1	4.40	4.40	15.0%	0.66	5.06
06.2.3.a.d	Antenna	1	1.50	1.50	30.0%	0.45	1.95
06.2.3.a.e	Band Pass Filter	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.a.f	Band Reject Filter	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.a.g	Sensor	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.a.h	Cabling	1	2.00	2.00	50.0%	1.00	3.00
06.2.3.a.i	Diplexer	2	0.40	0.80	15.0%	0.12	0.92
06.2.3.a.j	Coupler	1	0.40	0.40	15.0%	0.06	0.46
06.2.3.a.k	Gimbal	1	2.30	2.30	30.0%	0.69	2.99
06.2.3.a.l	Misc#2	1	1.00	1.00	15.0%	0.15	1.15
06.2.3.b	**Omni Antenna**			0.60	3.0%	0.02	0.62
06.2.3.b.a	Transponder	0	4.00	0.00	10.0%	0.00	0.00
06.2.3.b.b	RF Assembly	0	0.20	0.00	3.0%	0.00	0.00
06.2.3.b.c	Processing Module	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.b.d	Antenna	2	0.30	0.60	3.0%	0.02	0.62
06.2.3.b.e	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.b.f	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.b.g	Misc#3	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.b.h	Misc#4	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.b.i	Misc#5	0	0.00	0.00	0.0%	0.00	0.00
06.2.3.c	**Communications Instrumentation**			0.20	50.0%	0.10	0.30
06.2.3.c.a	Coaxial Cable	2	0.10	0.20	50.0%	0.10	0.30

The communications system consisted of a 200 W X-Band Transponder, a single, dual axis 0.7 m HGA, and two Omni-antennas as shown in Figure 5.1.

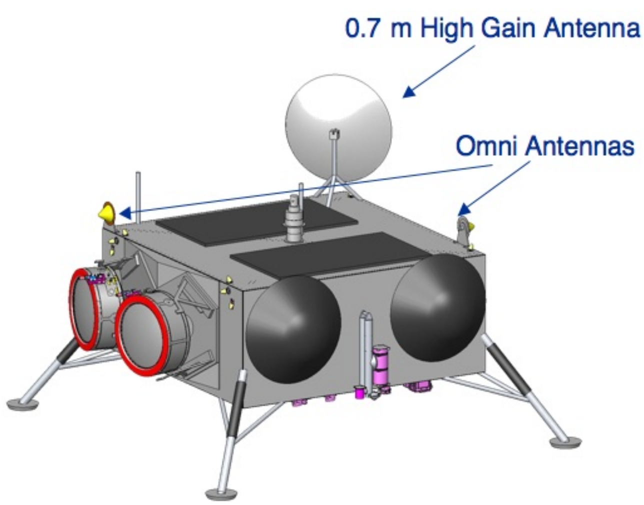


Figure 5.1—NEA Sample Return S/C—Communications Instruments

5.1.4 Communications Trades

None

5.1.5 Communications Analytical Methods

The following section contains the calculations for five different ways to quantify link budget analysis for this mission. The link budgets for NEARER provide values of RF transmit power of 85 and 200 W and antenna gains for X-Band. The first link budget calculation is the worst case with a distance between the Earth and Nereus of 2.4 AU. The next four are between Earth at apogee and Nereus at apogee assuming the apogees are collinear with the center of the Sun. Table 5.3 summarizes the findings. Note: All analysis is done assuming 8 Gbits of data to be transmitted.

Table 5.3—Link Budget Calculations for Communications Scenarios

Case	Power, W	Earth antenna size, m	Data rates, kbps	Transmission, hr
Worst	200	70	5.9	376.7
Case 1	200	70	212.0	10.5
Case 2	200	34	48.0	46.3
Case 3	85	70	90.0	24.7
Case 4	85	34	21.0	105.9

The number hours on Earth between the beginning of the transmission to the end of transmission will depend on the percentage of time the S/C are attached to the asteroid, can see Earth and the probability the Earth stations used to receive the transmission is available. The product of these numbers when divided into the hours of transmission will give one the total time necessary to transmit the data to Earth.

The link budgets for NEARER provide values of RF transmit power at most 200 W and antenna gains for X-band. Worst case is for 2.4 AU Earth - Nereus distance (an overestimate - should be 2.0 AU), provides nearly the 7 kbps rate needed. Further trades of DSN time, antenna sizes (ground and S/C), and transmitter power need to be made.

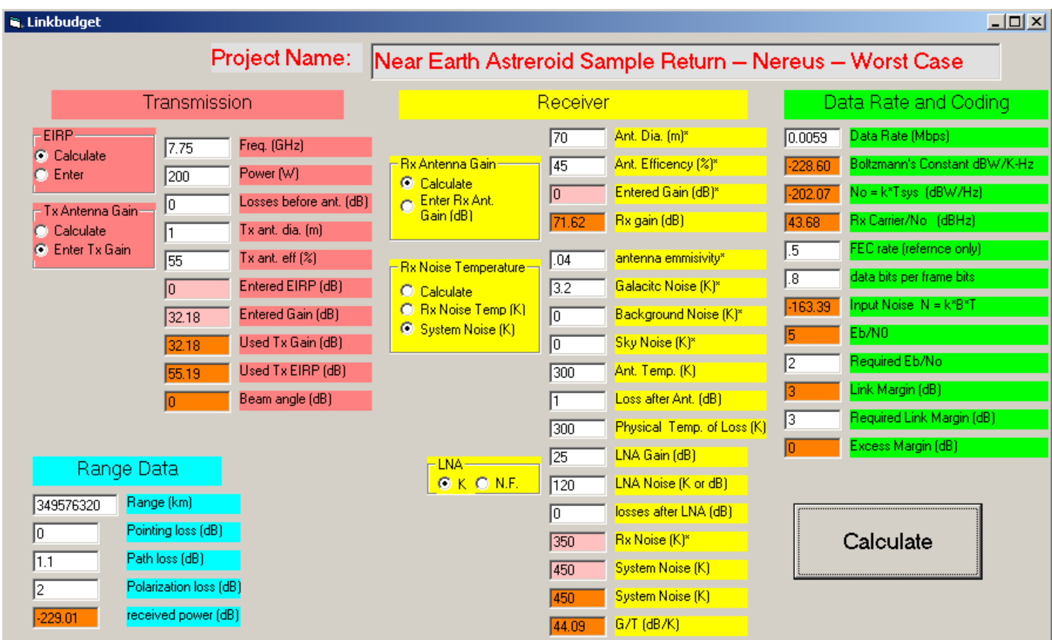


Figure 5.2—Worst Case Nereus Link Budget Analysis

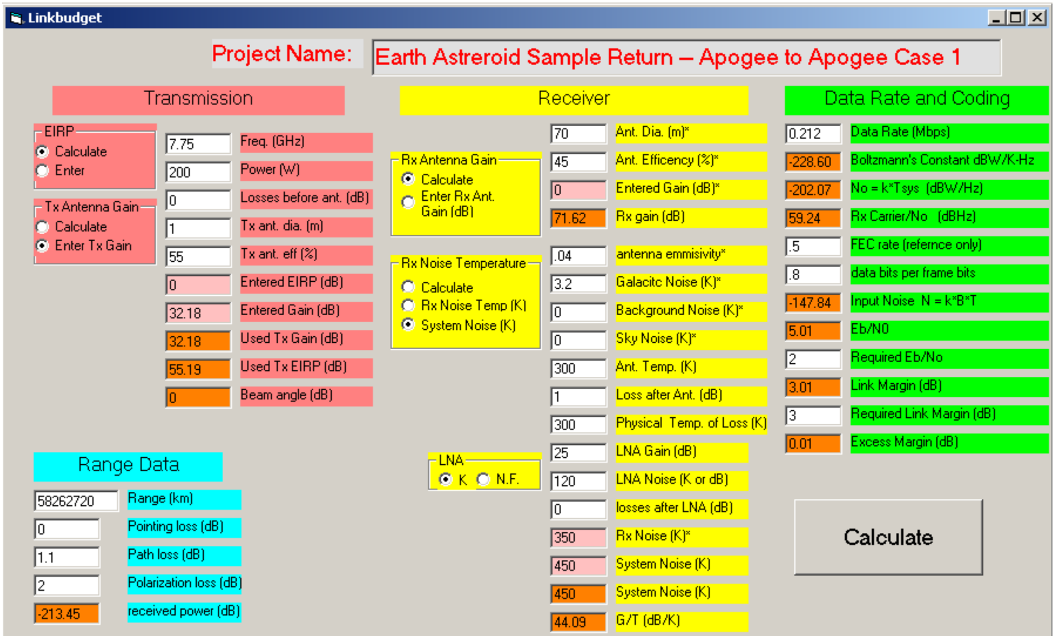


Figure 5.3—Case1 Nereus Link Budget Analysis

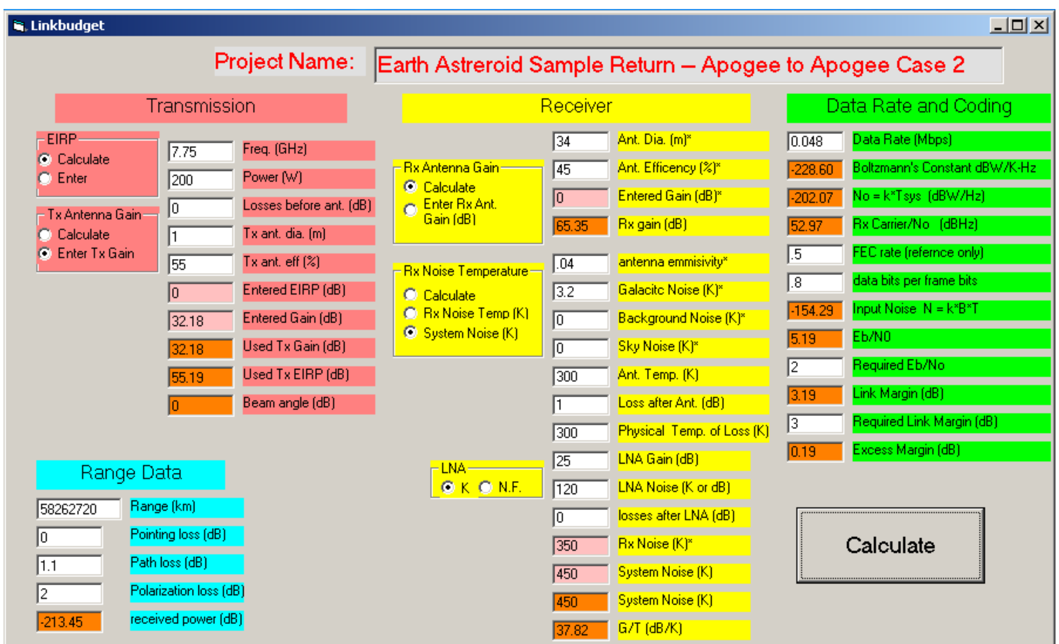


Figure 5.4—Case 2 Nereus Link Budget Analysis

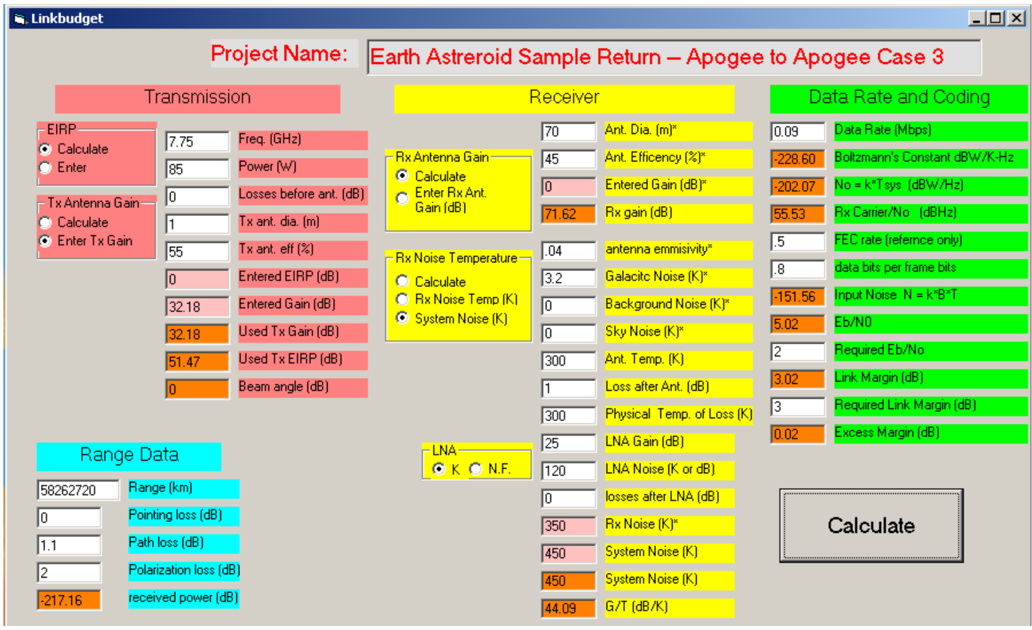


Figure 5.5—Case 3 Nereus Link Budget Analysis

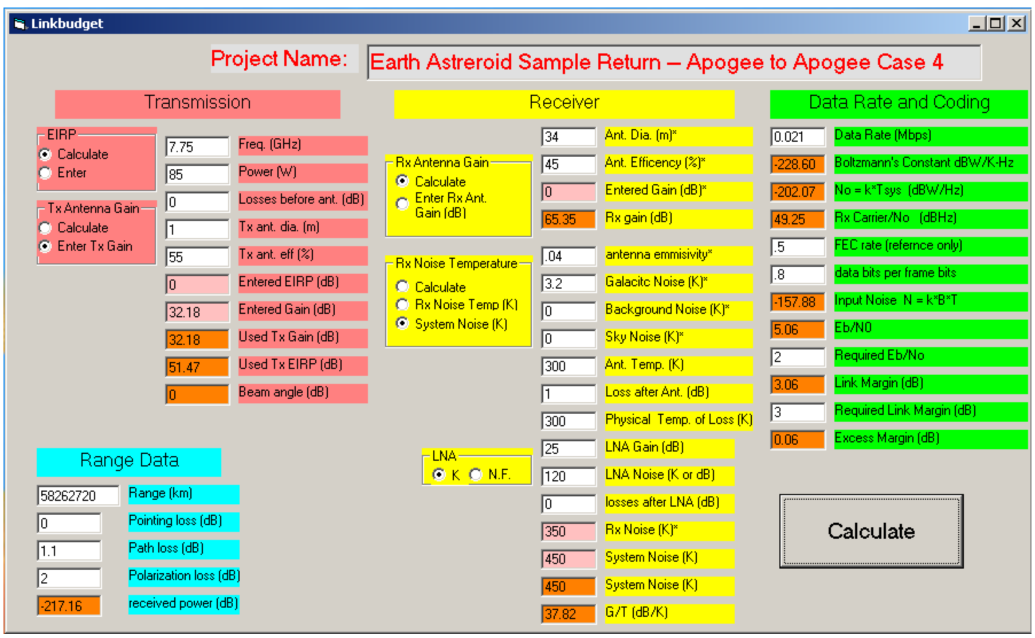


Figure 5.6—Case 4 Nereus Link Budget Analysis

5.1.6 Communications Risk Inputs

None submitted

5.1.7 Communications Recommendation

See design

5.2 Guidance, Navigation and Control (GN&C)

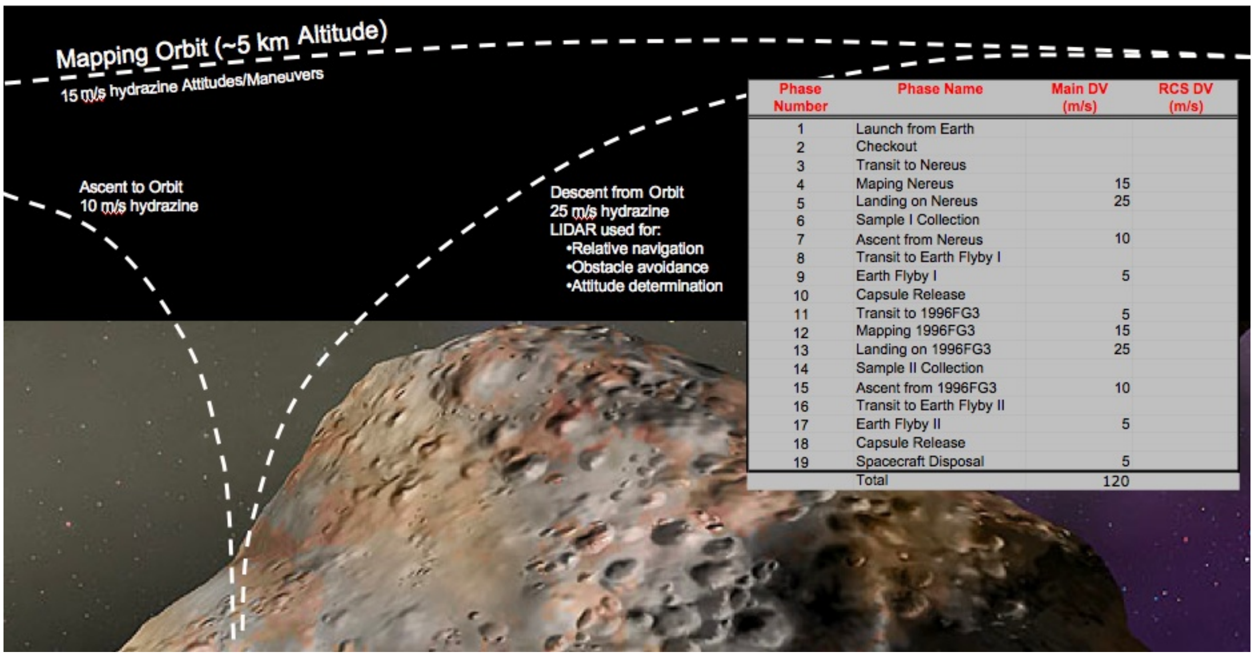
5.2.1 GN&C Requirements

The GN&C subsystem shall provide full 6-DOF control of the vehicle from launch through end of mission. This includes stabilization of the vehicle after launch vehicle separation, attitude control throughout the cruise, commanding and controlling all slews, and performing all automated landings, hops, and ascent maneuvers.

5.2.2 GN&C Assumptions

Much of the work in modeling the GN&C system for the Near Earth Asteroid Sample Return mission was built upon the system designed to perform this function in the Asteroid Sample Return mission completed by COMPASS.

The ΔV values and Xe propellant allocations used are summarized in the table in Figure 5.7.



Phase Number	Phase Name	Main DV (m/s)	RCS DV (m/s)
1	Launch from Earth		
2	Checkout		
3	Transit to Nereus		
4	Maping Nereus	15	
5	Landing on Nereus	25	
6	Sample I Collection		
7	Ascent from Nereus	10	
8	Transit to Earth Flyby I		
9	Earth Flyby I	5	
10	Capsule Release		
11	Transit to 1996FG3	5	
12	Mapping 1996FG3	15	
13	Landing on 1996FG3	25	
14	Sample II Collection		
15	Ascent from 1996FG3	10	
16	Transit to Earth Flyby II		
17	Earth Flyby II	5	
18	Capsule Release		
19	Spacecraft Disposal	5	
	Total	120	

Figure 5.7—Mapping Orbit Delta V Assumptions

5.2.3 GN&C Design and MEL

The GN&C subsystem hardware is made up of:

- Four reaction wheels (Valley Forge VF MR 14.0, 14 Nms reaction wheel, http://www.vfct.com/aerospace/wheels/small-wheels)
- Two Star Trackers (Adcole)
- One internally redundant IMU (Northrop Grumman HRG)
- Sun sensors to aide in Earth acquisition (Adcole Sun Sensors, New Horizons (NH) Heritage, two electronics boxes and three sensor heads, each)
- GN&C software run on main C&DH computers
- Utilizes LIDAR in science instrument subsystem for precision landing

The detailed mass accounting for the GN&C subsystem can be seen in Table 5.4. The block diagram can be seen in Figure 5.8.

Table 5.4—GN&C MEL (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			**1138.27**	**9.9%**	**113.05**	**1251.32**
06.1	**Science Payload**			**25.20**	**16.1%**	**4.06**	**29.26**
06.2	**Lander Spacecraft**			**1019.77**	**9.0%**	**92.07**	**1111.85**
06.2.1	**Attitude Determination and Control**			**44.28**	**21.1%**	**9.36**	**53.64**
06.2.1.a	**Guidance, Navigation, & Control**			**44.28**	**21.1%**	**9.36**	**53.64**
06.2.1.a.a	Sun Sensors	6	1.00	6.00	20.0%	1.20	7.20
06.2.1.a.b	Reaction Wheels	4	5.00	20.00	20.0%	4.00	24.00
06.2.1.a.c	Star Trackers	2	3.19	6.38	20.0%	1.28	7.66
06.2.1.a.d	IMU	1	6.90	6.90	20.0%	1.38	8.28
06.2.1.a.e	Laser Altimeter (from Science Payload)	1	5.00	5.00	30.0%	1.50	6.50
06.2.1.a.f	LIDAR	0	0.00	0.00	0.0%	0.00	0.00
06.2.1.a.g	Misc#3	0	0.00	0.00	0.0%	0.00	0.00

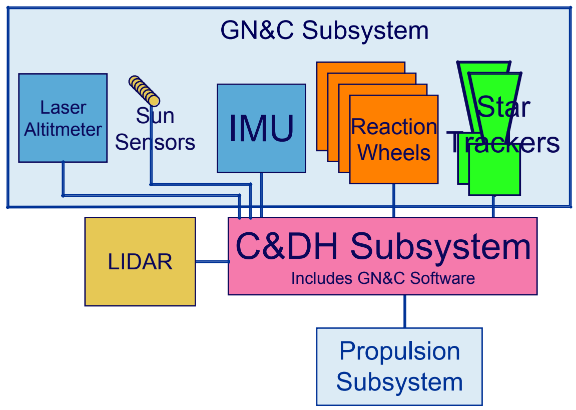


Figure 5.8—GN&C Subsystem Block Diagram

5.2.4 GN&C Trades

No specific trades were completed. A feasible design was constructed and minor changes were made to help close the mission.

5.2.5 GN&C Analytical Methods

The GN&C subsystem mainly utilized the Rocket Equation to calculate propellant masses for the mapping, landing, hop, and ascent burns. Again, because of the short duration of the study, no in-depth dynamic analyses were completed.

5.2.5.1 Solar Pressure Torques

- Solar pressure torques are significant because of single array configuration
- 3.7 m offset between centers of mass and pressure (Figure 5.9)
- 8.5×10^{-4} Nm of torque at 1 AU
- Assuming 15 Nm of momentum storage per axis, wheels would saturate approximately every 5 hr, with decreasing frequency as distance to sun increases
  - Despinning the wheels with the EP system with a 1° gimbal angle would take between 1.4 and 10.4 hr depending on current thrust level
- Preferably, off-pointing of the thruster by between 0.28° and 2.2° can counter the solar pressure torque with minimal thrust loss
  - Thrust loss of less than 0.1%

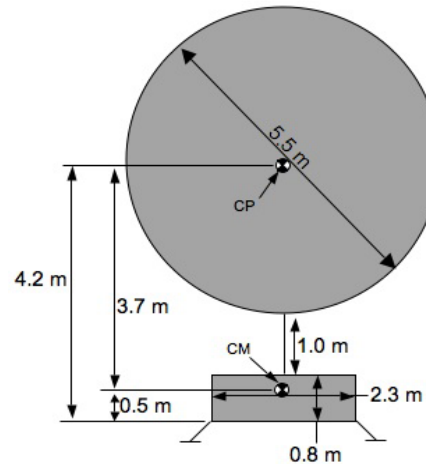


Figure 5.9—Solar Pressure Torque Geometry

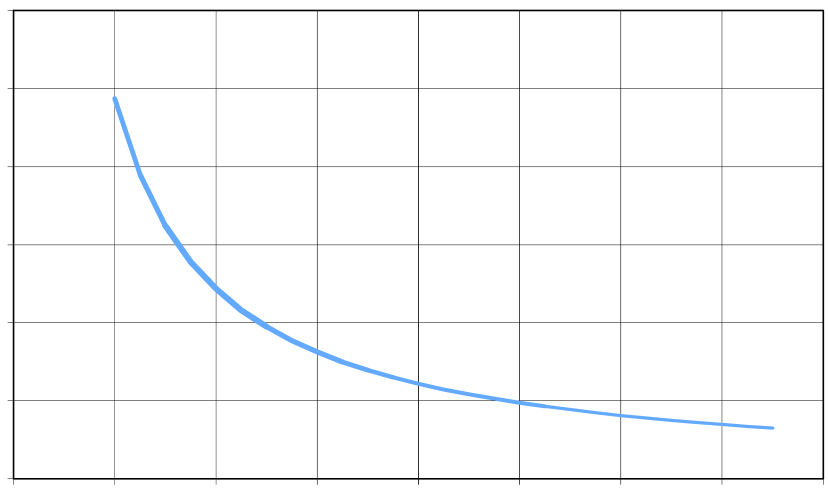


Figure 5.10—Thrust as a Function of Gimbal Angle

5.2.6 GN&C Risk Inputs

None submitted

5.3 Command and Data Handling (C&DH)

5.3.1 C&DH Requirements

- Storage: Be able to store all 60 days of an asteroid encounter: >12 Gbit
- Transmission: 8 hr per day
- ~7 kbits/sec needed

5.3.2 C&DH Assumptions

Assumed ~200 Mbits/day of data for all science instruments/housekeeping from combined science packages.

5.3.3 C&DH Design and MEL

Table 5.5—C&DH MEL for Baseline (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			**1138.27**	**9.9%**	**113.05**	**1251.32**
06.1	**Science Payload**			**25.20**	**16.1%**	**4.06**	**29.26**
06.2	**Lander Spacecraft**			**1019.77**	**9.0%**	**92.07**	**1111.85**
06.2.1	**Attitude Determination and Control**			**44.28**	**21.1%**	**9.36**	**53.64**
06.2.2	**Command and Data Handling**			**29.30**	**27.2%**	**7.96**	**37.26**
06.2.2.a	**Command & Data Handling**			**21.00**	**18.9%**	**3.96**	**24.96**
06.2.2.a.a	Flight Computer	2	2.00	4.00	20.0%	0.80	4.80
06.2.2.a.b	Command and Telemetry Computer	0	0.00	0.00	0.0%	0.00	0.00
06.2.2.a.c	Data Interface Unit	2	1.00	2.00	30.0%	0.60	2.60
06.2.2.a.d	Data Bus Operations Amplifier	0	0.00	0.00	0.0%	0.00	0.00
06.2.2.a.e	Operations Recorder	2	1.10	2.20	30.0%	0.66	2.86
06.2.2.a.f	Command and Control Harness (data)	1	4.00	4.00	30.0%	1.20	5.20
06.2.2.a.g	Shared DPU (From APL Science Instruments)	0	0.00	0.00	0.0%	0.00	0.00
06.2.2.a.h	Avionics enclosure	1	8.80	8.80	8.0%	0.70	9.50
06.2.2.a.i	Misc #3	0	0.00	0.00	0.0%	0.00	0.00
06.2.2.b	**Instrumentation & Wiring**			**8.30**	**48.2%**	**4.00**	**12.30**
06.2.2.b.a	Operational Instrumentation, sensors	1	0.30	0.30	0.0%	0.00	0.30
06.2.2.b.b	Data Cabling	1	8.00	8.00	50.0%	4.00	12.00
06.2.2.b.c	Misc #1	0	0.00	0.00	0.0%	0.00	0.00
06.2.2.b.d	Misc #2	0	0.00	0.00	0.0%	0.00	0.00

5.3.4 C&DH Trades

None.

5.3.5 C&DH Analytical Methods

Design approach will be based on the New Horizons avionics systems, but using RAD750 processor instead of RAD6000. GN&C and C&DH processors are to be combined into one. Avionics will be similar to this, but greatly modified for RF communications and flight and command & telemetry computers.

5.3.6 C&DH Risk Inputs

None submitted

5.3.7 C&DH Recommendation

See Design

5.4 Electrical Power System

5.4.1 Power Requirements

The power system shall provide sufficient power for the S/C system throughout the mission including Earth orbit, transfer to Nereus orbits, Nereus orbit and surface operations, transfer to 1996 FG3, 1996 FG3 orbit and surface operations, and sample return to Earth. In addition, the power system must be sized to supply sufficient power to the EP system throughout the mission.

5.4.2 Power Assumptions

Two SA sizes are included in different design iterations of the mission: one completely based on the Orion SA being developed under Constellation, and a second array with a similar design but larger diameter. An example of such an array is shown in Figure 5.11. The former is assumed to be OTS hardware without significant development costs. The larger diameter SA would incur some additional design and development costs but would provide more capability to the S/C.

Figure 5.11—Orion-Derived UltraFlex SA Design

The Orion SA is assumed to be about 5.5 m diameter and capable of producing 5 kWe of net power at 30 Vdc to the S/C at beginning of life (BOL) at 1 AU distance from the sun. The net power includes losses from the array wiring and electronics, the gimbal assembly, and the main electronics unit. The mass of the SA wing is assumed to be 50 kg.

The larger diameter SA net power and mass values are scaled based on a constant power and mass per area from the Orion array above.

The dust environment on the Apollo asteroids is assumed to not have significant affect on the SA due the lack of an atmosphere. However, the array is expected to have some degraded performance after each landing. Consequently, the array is oversized.

The batteries and electronics are also based on Orion technology. The battery chemistry is Li-ion type technology with a battery system-level specific energy of 120 W-hr/kg. An older example is shown in Figure 5.12.

Figure 5.12—Example Li-Ion Battery, Built for the Mars Phoenix Lander
(Similar Batteries are Planned for Orion)

The power system electronics are based on the Main Bus Switching Unit (MBSU) approach of Orion where a single box contains electronics cards for SA regulation and control, battery charge control, and overall bus current handling. An overall charge efficiency of 80% was assumed to account for the losses in recharging the battery from the SA.

Power Assumptions

- 5 kW BOL, 1 AU
- 2 kW BOL, 2.5 AU
- 320 W_{net} during eclipse on Moons' surface
- 500 W_{net} in daylight on Moons' surface

Figure 5.13 shows the change of available power from the SAs available over mission time as the S/C travels farther from the sun, and then orbits with the NEA.

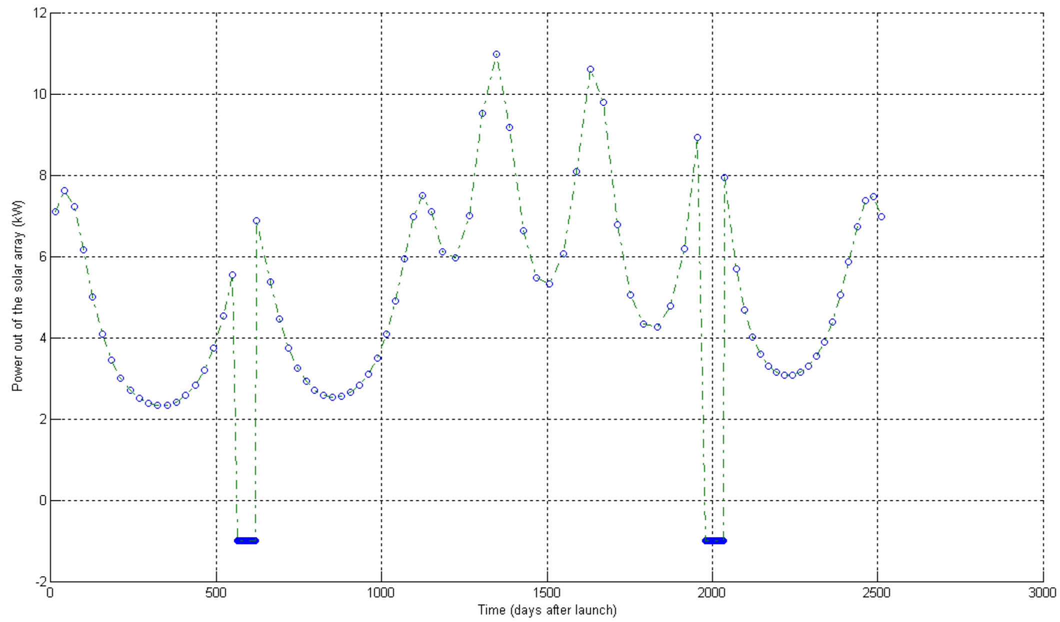


Figure 5.13—Power Available From SA Over Mission Time

5.4.3 Power Design and MEL

The circular UltraFlex SA was chosen for two primary reasons. First, it provides OTS array technology that is lightweight and close to the required net power level for the mission. Second, it is more structurally sound when deployed, which may be necessary for the multiple landings on the moons' surfaces. Leaving the array deployed during the landings seems to be less risky than retracting the array for each landing. Better understanding of the dust environment may affect this decision in the future.

The solar distances of the overall mission made a larger SA attractive, so a second UltraFlex design at 7 m diameter was also included in another iteration. This size array should have similar characteristics to the 5.5 m array with possibly some lower amount of allowable acceleration or higher structural requirement or both.

The Orion-based array has not been finalized and there is a chance of the diameter changing. An increase in diameter would most likely be beneficial and welcome, whereas a decrease would most likely require a redesigned array for this mission due to the 5.5 m array seeming to provide what is the lowest possible power level acceptable for this mission.

Since the projected timeline of this mission is similar to that of Orion, the solar cells were assumed to be in the same range of efficiency as Orion, 29% at the cell-level. If the launch date is later, beyond 2020, more advanced cells may be more realistic, with efficiencies around 30 to 32% and proportional power increases.

Traditional rectangular folding SAs were also evaluated but did not make sense due to their lower power per mass and lower structural integrity when deployed. Their cost may be lower than the UltraFlex SA, but the other factors appear to be more important here.

The SA power requirements are sized primarily from the EP needs. EP requires an array that is oversized for the remainder of the mission, with potential exception for the surface operations. Due to the dust environment and Sun-pointing challenges on the moons' surfaces, excess array power for both 'daytime' power and recharging the batteries for 'nighttime' operations is probably worthwhile.

The challenging sun-pointing requirements mentioned above are also why a two-axis gimbal is specified. This additional DOF may allow more potential landing sites that might have significant slopes or challenging lines-of-sight to the sun.

The sample return S/C includes a small mass of batteries and power electronics as well. The same specific energy assumption was made here, though a primary battery may be a good choice since it might reduce the mass slightly, but no recharge capability is available.

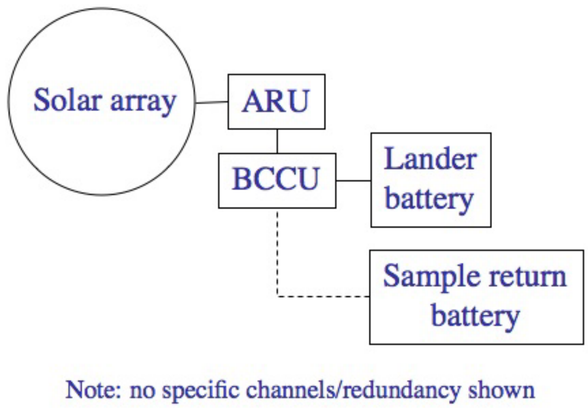


Figure 5.14—Power System Schematic

Table 5.6—Electrical Power System MEL for Baseline (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			**1138.27**	9.9%	**113.05**	**1251.32**
06.1	**Science Payload**			**25.20**	16.1%	**4.06**	**29.26**
06.2	**Lander Spacecraft**			**1019.77**	9.0%	**92.07**	**1111.85**
06.2.1	**Attitude Determination and Control**			**44.28**	**21.1%**	**9.36**	**53.64**
06.2.2	**Command and Data Handling**			**29.30**	**27.2%**	**7.96**	**37.26**
06.2.3	**Communications and Tracking**			**24.20**	**27.1%**	**6.55**	**30.75**
06.2.4	**Electrical Power Subsystem**			**115.50**	**15.7%**	**18.19**	**133.69**
06.2.4.a	**Solar Arrays**			78.00	10.8%	8.44	86.44
06.2.4.a.a	Solar Array Mass (cells and structure only)	1	70.00	70.00	10.0%	7.00	77.00
06.2.4.a.b	Solar Array Gimbal Assembly	1	8.00	8.00	18.0%	1.44	9.44
06.2.4.a.c	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.a.d	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.b	**Power Management & Distribution**			15.00	15.0%	2.25	17.25
06.2.4.b.a	Main Bus Switching Unit	1	15.00	15.00	15.0%	2.25	17.25
06.2.4.b.b	Battery Charge Control Unit	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.b.c	DC Switchgear/Shunt Regulator	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.b.d	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.c	**Power Cable and Harness Subsystem (C and HS)**			10.00	50.0%	5.00	15.00
06.2.4.c.a	Spacecraft Bus Harness	1	10.00	10.00	50.0%	5.00	15.00
06.2.4.c.b	PMAD Harness	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.c.c	Electric Propulsion Harnes	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.c.d	RPS to Spacecraft Harness	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.c.e	Power Cabling	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.c.f	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.c.g	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.4.d	**Battery System**			12.50	20.0%	2.50	15.00
06.2.4.d.a	Battery Assembly-Primary	1	12.50	12.50	20.0%	2.50	15.00
06.2.4.d.b	Secondary Battery Subsystem	0	0.00	0.00	0.0%	0.00	0.00

5.4.4 Power Trades

None performed.

5.4.5 Power Analytical Methods

The power system analysis is based on current approaches and assumptions from the Orion S/C system.

5.4.6 Power Risk Inputs

None submitted

5.4.7 Power Recommendation

The recommended power system based on current understanding is a single UltraFlex SA, sized to meet the overall mission requirements; and Li-ion batteries for both the main S/C and the sample return S/C.

5.5 Structures and Mechanisms

5.5.1 Structures and Mechanisms Requirements

The intent is to provide the necessary hardware for the science research instrumentation, avionics, communications, propulsion and power. Structure must be able to withstand applied loads from launch vehicle and provide minimum deflections, sufficient stiffness, and vibration damping to perform the mission and survive the round trip trajectory. The design of the structure will strive to minimize weight (mass) in order to optimize performance of the S/C and fit on the launch vehicle. Physically, the structure of the S/C must allow is to fit within confines of launch vehicle and its payload fairing. The structure must be stiff and strong enough to accommodate landing and takeoff from low g terrestrial bodies.

5.5.2 Structures and Mechanisms Assumptions

The structural design used the following baseline assumptions in the design of the main bus and SRC.

- Material: Aluminum
- Space frame with tubular members
- Composite sandwich structure shelf
- Welded and threaded fastener assembly

5.5.3 Structures and Mechanisms Design and MEL

Design Description

- Thrust tube and tubular space frame in square configuration
- Shelf of composite sandwich architecture with honeycomb core to mount hardware
- Thin sheets to enclose structure and provide shear stiffness
- Spring struts to support landing hardware

5.5.4 Main S/C Bus Design

The main S/C bus is modeled as a tubular space frame in polygonal configuration shown in Figure 5.15. The material is a shelf of composite sandwich architecture with honeycomb core to mount hardware. Thin sheets were used to enclose structure. Struts were added to support landing hardware.

Table 5.7 shows both the lander S/C and the sample return craft as they were reported in the master S/C MEL. Note that there are two SRCs in this MEL, with quantity = 2. The installation calculation is done using 4% of the dry CVE mass of each subsystem.

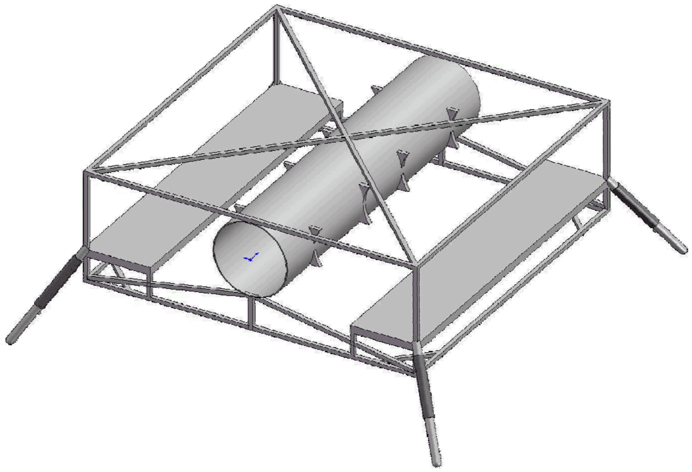


Figure 5.15—Main S/C Bus Analytical Design

Table 5.7—Structures and Mechanical Systems MEL for Baseline (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			1138.27	9.9%	113.05	1251.32
06.1	**Science Payload**			25.20	16.1%	4.06	29.26
06.2	**Lander Spacecraft**			1019.77	9.0%	92.07	1111.85
06.2.1	**Attitude Determination and Control**			44.28	21.1%	9.36	53.64
06.2.2	**Command and Data Handling**			29.30	27.2%	7.96	37.26
06.2.3	**Communications and Tracking**			24.20	27.1%	6.55	30.75
06.2.4	**Electrical Power Subsystem**			115.50	15.7%	18.19	133.69
06.2.5	**Thermal Control (Non-Propellant)**			40.11	18.0%	7.22	47.33
06.2.6	**Propulsion**			192.10	8.3%	16.00	208.09
06.2.7	**Propellant**			425.41	0.0%	0.00	425.41
06.2.8	**Structures and Mechanisms**			148.88	18.0%	26.80	175.68
06.2.8.a	**Structures**			121.52	18.0%	21.87	143.40
06.2.8.a.a	Primary Structures			102.07	18.0%	18.37	120.45
06.2.8.a.a.a	Main Bus Structure	1	102.07	102.07	18.0%	18.37	120.45
06.2.8.a.a.b	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.8.a.a.c	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.8.a.b	Secondary Structures			19.45	18.0%	3.50	22.95
06.2.8.a.b.a	Balance Mass	1	0.00	0.00	18.0%	0.00	0.00
06.2.8.a.b.b	Tank Supports and Bracketry	1	9.09	9.09	18.0%	1.64	10.73
06.2.8.a.b.c	Landing Gear Structure	1	8.15	8.15	18.0%	1.47	9.62
06.2.8.a.b.d	Solar Array Boom	1	0.61	0.61	18.0%	0.11	0.72
06.2.8.a.b.e	Misc#3	1	0.70	0.70	18.0%	0.13	0.83
06.2.8.a.b.f	Misc#4	1	0.89	0.89	18.0%	0.16	1.05
06.2.8.b	**Mechanisms**			27.35	18.0%	4.92	32.28
06.2.8.b.a	Solar Array Mechanisms			0.00	0.0%	0.00	0.00
06.2.8.b.b	Thruster Mechanisms			0.00	0.0%	0.00	0.00
06.2.8.b.c	Communications Mechanisms			0.00	0.0%	0.00	0.00
06.2.8.b.d	Thermal Mechanisms			0.00	0.0%	0.00	0.00
06.2.8.b.e	Adaptors and Separation			0.00	0.0%	0.00	0.00
06.2.8.b.f	Additional Mechanisms			9.84	18.0%	1.77	11.61
06.2.8.b.f.a	Sample Arm & Mechanism	1	7.50	7.50	18.0%	1.35	8.85
06.2.8.b.f.b	Landing Gear Displacement Mech.	1	2.34	2.34	18.0%	0.42	2.76
06.2.8.b.g	Installations			17.51	18.0%	3.15	20.67
06.2.8.b.g.a	Science Payload Installation	1	1.17	1.17	18.0%	0.21	1.38
06.2.8.b.g.b	C&DH Installation	1	1.17	1.17	18.0%	0.21	1.38
06.2.8.b.g.c	Communications and Tracking Installation	1	0.97	0.97	18.0%	0.17	1.14
06.2.8.b.g.d	GN&C Installation	1	0.37	0.37	18.0%	0.07	0.44
06.2.8.b.g.e	Electrical Power Installation	1	4.62	4.62	18.0%	0.83	5.45
06.2.8.b.g.f	Therrmal Control Installation	1	1.60	1.60	18.0%	0.29	1.89
06.2.8.b.g.g	Electric Propulsion Installation	1	7.61	7.61	18.0%	1.37	8.97
06.2.8.b.g.h	Misc #1	0	0.00	0.00	0.0%	0.00	0.00

SRC Design

The SRC design (see Figure 5.16) will be constant across the three design cases. The design is loosely based upon the sample return capsule. Each case will use the same SRC to contain the samples from the asteroids for return to Earth. It is designed to withstand the launch and Earth return inertial loading as well as the extreme aerothermal re-entry and deceleration loads. The SRC has an overall diameter of 0.7 m and is 0.4 m tall with a 45° cone heat shield. The heat shield translates to expose a sample containment carousel structure. The heat shield joint is closed by engaging three latching mechanisms equally spaced around the joint circumference. The heat shield is constructed of Phenolic Impregnated Carbon Ablator (PICA) to withstand temperatures of up to 3600 °C. PICA is a modern Thermal protection systems (TPS) material and has the advantages of low density (much lighter than carbon phenolic) coupled with efficient ablative capability at high heat flux.

The SRC is spin stabilized at 14 rpm for control of orientation during the aerodynamic deceleration with an estimated re-entry velocity approaching 2.9 km/s. The deceleration chutes are activated by a 3g accelerometer switch for the drogue and main chute deployment with timer delays. A battery powered UHF beacon transponder is also carried to assist in the recovery operation.

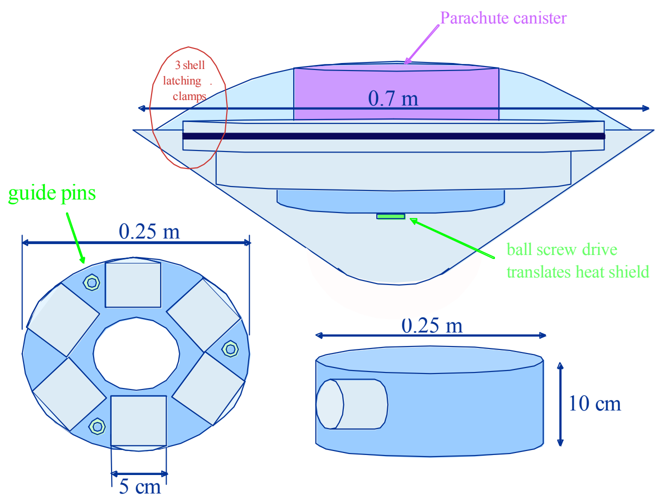


Figure 5.16—SRC concept design

The SRC is designed for the return of a total of 1 kg of samples in two separate six-sample carousel (for a total of 12 between the two carousels), with each container 5 cm deep with a 5 cm diameter. The entire sample carousel is 10 cm tall and 0.25 m in diameter. The temperature design limits are –4 to 40 °C for samples with a landing recovery shock mitigated to limit it to 4 g's of impact by incorporating crushable honeycomb foam beneath the heat shield.

The sample retrieval mechanism is a four DOF collection arm with a 1 m reach and a 0.5 m telescoping extension. It incorporates a scoop type bi-furcated shovel and uses motorized joints. Other mechanisms include three latching and sealing devices and the SRC spin and separate mechanism with an electrical cable-severing guillotine. A Xenon cold gas system is pulsed to stabilized the lander during retrieval arm operations.

5.5.5 SRC Design

- Overall diameter 1 m, 45° cone angle
- Main SRC 0.4 m tall
- Carousel rotates horizontally for access to the surface and to the science instruments.
- Sealed side door
- Re-entry velocity capability—2.9 km/s entry (below the required 11.5 km/s for this mission)

  - PICA heat shield (3600 °C)
  - Deceleration chutes
  - Beacon for tracking and pickup
  - Battery power
- 12 sample carousels (5 cm diameter by 5 cm deep)
- Sample carousel 0.4 m diameter., 8 cm tall
- Temperature limits of –4 to 40 °C for samples
- Sample return shock mitigation—limit to 4 g's
  - Crushable honeycomb
  - Foam

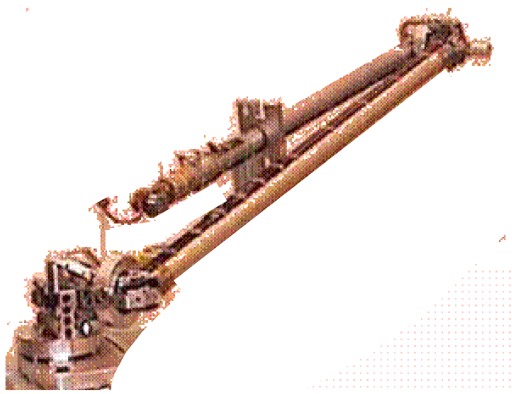


Figure 5.17—Sample Collection Arm

5.5.6 SRC MEL

Table 5.8 racks up the masses and subsystems in the SRC located on the NEARER S/C. Note that in this baseline case (Case 1) in this study, there are two SRCs. In order to enter two carousels, the MEL line elements have quantity (QTY) of 2 in those places where appropriate. For example, there are 2 battery subsystems in power, and 2 sample canisters in structures and mechanisms, etc.

Table 5.8—SRC MEL (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			1138.27	9.9%	113.05	1251.32
06.1	Science Payload			25.20	16.1%	4.06	29.26
06.2	Lander Spacecraft			1019.77	9.0%	92.07	1111.85
06.2.1	Attitude Determination and Control			44.28	21.1%	9.36	53.64
06.2.2	Command and Data Handling			29.30	27.2%	7.96	37.26
06.2.3	Communications and Tracking			24.20	27.1%	6.55	30.75
06.2.4	Electrical Power Subsystem			115.50	15.7%	18.19	133.69
06.2.5	Thermal Control (Non-Propellant)			40.11	18.0%	7.22	47.33
06.2.6	Propulsion			192.10	8.3%	16.00	208.09
06.2.7	Propellant			425.41	0.0%	0.00	425.41
06.2.8	Structures and Mechanisms			148.88	18.0%	26.80	175.68
06.3	Sample Return Craft			93.30	18.1%	16.91	110.21
06.3.1	Electrical Power Subsystem			6.00	20.0%	1.20	7.20
06.3.1.a	Battery Subsystem	2	3	6.00	20.0%	1.20	7.20
06.3.1.b	Misc #2	0	0	0.00	0.0%	0.00	0.00
06.3.1.c	Misc #3	0	0	0.00	0.0%	0.00	0.00
06.3.1.d	Misc #4	0	0	0.00	0.0%	0.00	0.00
06.3.2	Thermal Control (Non-Propellant)			4.60	18.0%	0.83	5.43
06.3.3	Structures and Mechanisms			82.70	18.0%	14.89	97.59
06.3.3.a	Sample Canisters w/Carousel	2	3.35	6.70	18.00%	1.21	7.91
06.3.3.b	Sample Arm & Mechanism	0	7.50	0.00	18.00%	0.00	0.00
06.3.3.c	Primary SRC Structure	2	12.80	25.60	18.00%	4.61	30.21
06.3.3.d	S/C SRC Separation Mechanism	2	3.90	7.80	18.00%	1.40	9.20
06.3.3.e	Re-Entry Aero Heat Shield	2	13.30	26.60	18.00%	4.79	31.39
06.3.3.f	Recovery Parachutes	2	3.50	7.00	18.00%	1.26	8.26
06.3.3.g	Shock Absorbing Material	2	4.50	9.00	18.00%	1.62	10.62

5.5.7 Structures and Mechanisms Trades

Analysis and Trades

- Sizing of space frame to accommodate requirements for
- Antenna
- Instrumentation
- Landing gear
- Fit within confines of launch vehicle

5.5.8 Structures and Mechanisms Analytical Methods

Sizing of the space frame to accommodate requirements for the antenna and instrumentation while fitting within confines of launch vehicle.

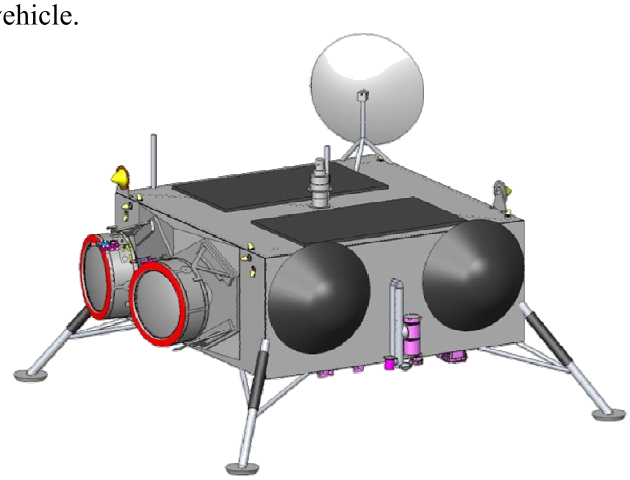


Figure 5.18—Example Graphic of Main Structure

Preliminary Structural Analysis

- Provided conditions
  - Maximum axial acceleration from launch vehicle: 6g
  - Maximum lateral acceleration from launch vehicle: 2g
  - Approach velocity: 10 in./s (0.25 m/s)
  - Maximum allowable pressure on moon surface: 0.5 psi (3.4 kPa)
- Thrust tube wall thickness of 0.125 in. (3.2 mm) with a 20 in. (508 mm) OD
  - Max. bending stress of 52 ksi (357 MPa)
  - Assumed lateral load through CG of lander
- Landing gear displacement
  - Four 8 in. (203 mm) diameter pads
  - ~2200 lb (1000 kg)
  - Constant stiffness springs
  - Needed displacement: 5.5 in. (140 mm)

5.5.9 Structures and Mechanisms Risk Inputs

Potential impact with foreign object or due to nearby operations.

5.5.10 Structures and Mechanisms Recommendation

Finite Element Analysis (FEA) to determine stresses and displacements along with a modal analysis for vibrations.

5.6 Propulsion and Propellant Management

5.6.1 Propulsion and Propellant Management Requirements

The S/C propulsion subsystem was required for three propulsion operations:

1. Electric propulsion: Orbit transfer and insertion
2. Chemical propulsion: Reaction/attitude control
3. Chemical propulsion: Surface landing

Three cases were examined to quantify benefits of alternate electric propulsion technologies.

5.6.2 Propulsion and Propellant Management Assumptions

Because an objective in this study was to determine mission benefits of electric propulsion, the electric propulsion subsystems used were either commercially available or systems currently under advanced development at NASA GRC. The development status of the technology has been accounted for in the Cost Analysis. Electric thruster performance used for mission analysis is based on demonstrated operation.

All chemical thrusters used in the design as well as the propellant management components and propellant tanks for the electric and chemical propulsion propellants were commercially available devices from operating manufactures. The current technology parameters for the various EP thrusters are shown the Table 5.9.

Table 5.9—NEXT Thruster Options Technology Assumptions

		Thruster Options					
		HiVHAC Hall Thruster[*]		NEXT Ion Thruster[†]		BPT-4000 Hall Thruster[∂, f, ß]	
Resource	Subsystem	CBE	Basis	CBE	Basis	CBE	Basis
Mass, kg	Thruster	6.075	?	12.7	PM Actual, w/o harnes	13.3	
	PPU	10.08	?	< 34.5	EM Actual, module sum	12.75	PM Actual
	PMS - HPA	0.59	Development system[Ω]	1.9	EM Actual, w/plate & TSE	1.9	Assumption
	PMS - LPA	0.5	BPT-4000 system	3.1	EM Actual, w/plate & TSE	0.5	PM Actual
	Gimbal	3.0375	?	6	Breadboard Actual	6.15	PM Actual
	Gimbal Drive E	2	?	0	n/a	0	n/a
Envelope, cm	Thruster	n/a		58 dia X 44 length	PM Actual	n/a	
	PPU	n/a		42 X 53 X 14	EM Actual	3 X 40 X 11	PM Actual
	PMS - HPA	n/a		33 X 15 X 6.4	EM Actual	n/a	
	PMS - LPA	n/a		38 X 30.5 X 6.4	EM Actual	n/a	
	Gimbal	n/a		72 cm corner-corner, 61 cm flat-flat	Breadboard Actual	n/a	
Engine Propellant Throughput, kg		300	*	520	Ultimate life/1.5	300	Assumption

* HiVHAC Hall Thruster data provided by D. Manzella & H. Kamhawi, April, 2008

† NEXT thruster provided by S. Benson, June, 2007

∂ BPT-4000 Hall Thruster data provided by D. Manzella & H. Kamhawi, April, 2008

f Fischer, et al, "The development and qualification of a 4.5 kW Hall Thruster propulsion system," AIAA paper 2003-4051, 39th AIAA JPC, July, 2003

ß de Grys, et al, "Multi-mode 4.5 kW BPT-4000 Hall Thruster qualification status," AIAA paper 2003-4552, 39th AIAA JPC, July, 2003

Ω Data provided by J. Dankanich, June, 2007

5.6.3 Propulsion and Propellant Management Design and MEL

Main Electric Propulsion Subsystem

- Two NASA/GRC NEXT ion thrusters—1 operating, 1 spare
- Gimbals on each thruster for thrust vector control
- Two Power Processing Units individually mated to the thrusters (no cross-strapping)
- Two COPV Ti-lined high-pressure cylindrical storage tanks for the Xe propellant (nominal)
- Xe distribution system incorporates VACCO (Vacuum and Air Components Company of America—www.vacco.com)—developed pressure and flow control devices

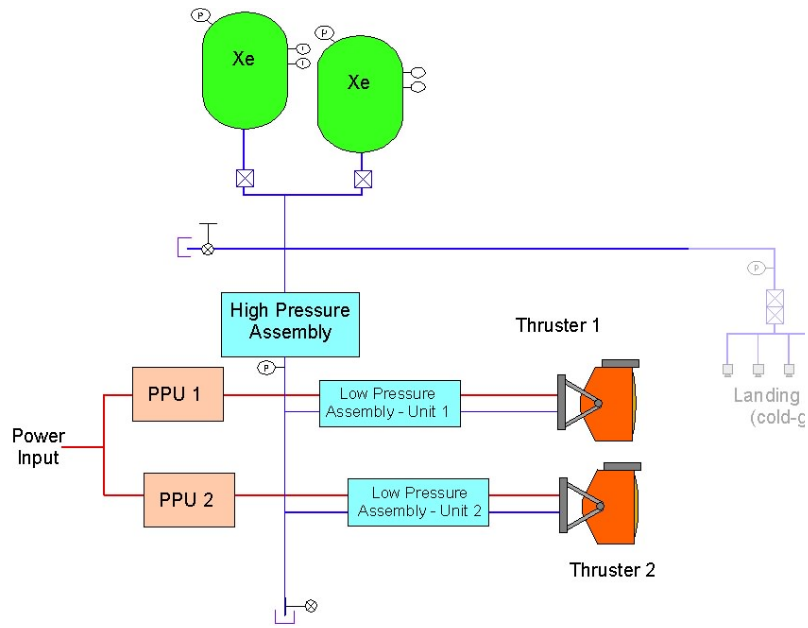


Figure 5.19—Main Propulsion System

Vehicle Landing Propulsion System

- Further revision is required to validate current mass estimates and propellant requirements
- System based on an existing Moog cold-gas thrusters was assembled
  - Used Xe gas from main EP storage tanks

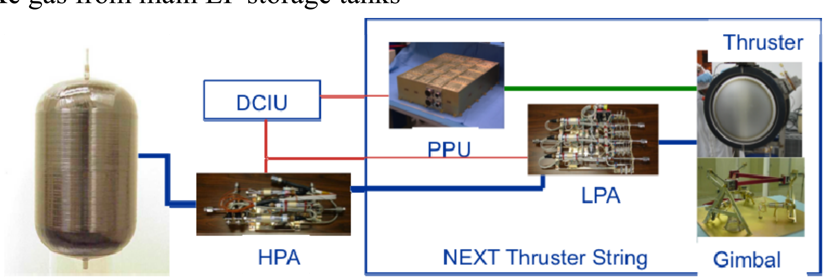


Figure 5.20—Notional Vehicle Landing Propulsion System

Reaction Control Propulsion System

The propulsion subsystem is comprised of

- 16 - 1 lbf mono-prop thrusters placed around S/C body for reaction control
  - Aerojet MR-111 thrusters operating on hydrazine
    - Isp = 229 sec

        - Thrust = 4.4 N
    - Thrusters require power for operation of catalytic bed
- Fuel stored in Ti metallic tank
    - Two spherical tank
    - Blow down pressurization with gHe
    - Propellant distribution system used design similar to systems developed for the Constellation program
        - Including fault tolerance configuration
    - Multiple tank and line heaters are included in mass model to prevent propellant and pressurant from freezing
        - Additionally, insulation included for same elements
    - Instrumentation - nominal suite of temperature and pressure sensors

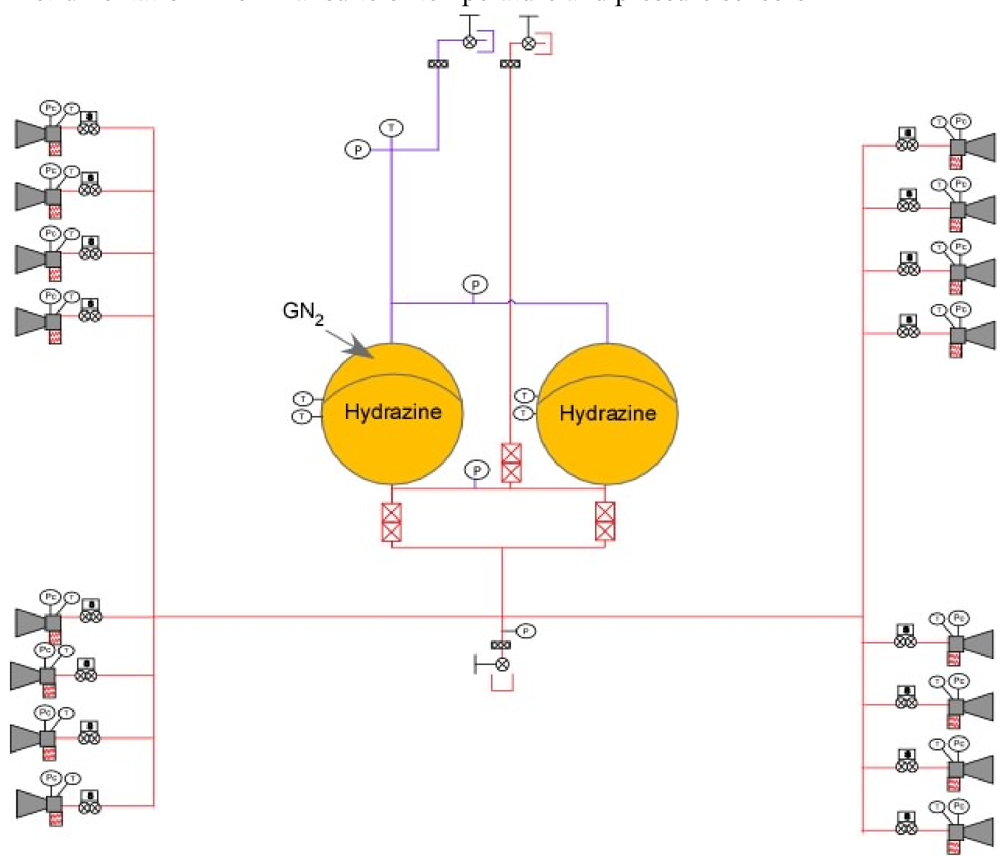


Figure 5.21—RCS Propulsion system schematic

Table 5.10—Propulsion and Propellant System MEL for Baseline (Case 1)

WBS Number	Description NEA Sampler (August 2008) - Case 1	QTY	Unit Mass (kg)	CBE Mass (kg)	Growth (%)	Growth (kg)	Total Mass (kg)
06	**NEA Sampler Spacecraft**			**1138.27**	**9.9%**	**113.05**	**1251.32**
06.1	**Science Payload**			**25.20**	**16.1%**	**4.06**	**29.26**
06.2	**Lander Spacecraft**			**1019.77**	**9.0%**	**92.07**	**1111.85**
06.2.1	**Attitude Determination and Control**			**44.28**	**21.1%**	**9.36**	**53.64**
06.2.2	**Command and Data Handling**			**29.30**	**27.2%**	**7.96**	**37.26**
06.2.3	**Communications and Tracking**			**24.20**	**27.1%**	**6.55**	**30.75**
06.2.4	**Electrical Power Subsystem**			**115.50**	**15.7%**	**18.19**	**133.69**
06.2.5	**Thermal Control (Non-Propellant)**			**40.11**	**18.0%**	**7.22**	**47.33**
06.2.6	**Propulsion**			**192.10**	**8.3%**	**16.00**	**208.09**
06.2.6.a	**Propulsion Hardware (EP)**			**38.20**	**8.0%**	**3.06**	**41.26**
06.2.6.a.a	Primary EP Thrusters	2	13.10	26.20	8.0%	2.10	28.30
06.2.6.a.b	EPS Power Processing and Control	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.a.c	EPS Structure			12.00	8.0%	0.96	12.96
06.2.6.a.c.a	EP Thruster Pod	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.a.c.b	EP Thruster Boom	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.a.c.c	Gimbal	2	6.00	12.00	8.0%	0.96	12.96
06.2.6.a.d	EPS Thermal Control Subsystem			0.00	0.0%	0.00	0.00
06.2.6.a.d.a	EPS Multi-Layer Insulation	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.a.d.b	EPS Heaters and Sensors	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.a.d.c	Misc #1	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b	**Propellant Management (EP)**			45.03	11.7%	5.28	50.31
06.2.6.b.a	Xe propellant tank(s)	2	12.72	25.44	2.0%	0.51	25.95
06.2.6.b.b	High Pressure Feed System	1	15.68	15.68	18.0%	2.82	18.50
06.2.6.b.c	Low Pressure Feed System	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.d	Residual Xe Propellant (non deterministic)	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.e	Temperature sensors	1	3.90	3.90	50.0%	1.95	5.85
06.2.6.b.f	Propulsion Tank heaters	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.g	Propulsion Line heaters	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.h	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.i	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.j	Misc#3	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.b.k	Misc#4	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.c	**Power Processing Unit (PPU)**			69.00	8.0%	5.52	74.52
06.2.6.c.a	PPU Mass	2	34.50	69.00	8.0%	5.52	74.52
06.2.6.c.b	Cabling	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.c.c	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.c.d	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.c.e	Misc#3	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.d	**Propulsion Hardware (Chemical)**			0.52	8.0%	0.04	0.56
06.2.6.d.a	Main Engine			0.52	8.0%	0.04	0.56
06.2.6.d.a.a	Main Engine	4	0.13	0.52	8.0%	0.04	0.56
06.2.6.d.a.b	Main Engine Gimbal	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.d.a.c	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.e	**Propellant Management (Chemical)**			0.00	0.0%	0.00	0.00
06.2.6.f	**Reaction Control System Hardware**			7.37	2.0%	0.15	7.51
06.2.6.f.a	RCS Thruster Subassembly	4	1.84	7.37	2.0%	0.15	7.51
06.2.6.f.b	Misc#1	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.f.c	Misc#2	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.f.d	Misc#3	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.g	**RCS Propellant Management**			31.99	6.1%	1.95	33.93
06.2.6.g.a	Fuel Tanks	2	7.81	15.62	2.0%	0.31	15.93
06.2.6.g.b	Fuel Lines	0	0.00	0.00	0.0%	0.00	0.00
06.2.6.g.c	Pressurization System - tanks, panels, lines	1	2.04	2.04	10.0%	0.20	2.24
06.2.6.g.d	Feed System - regulators, valves, etc	1	14.33	14.33	10.0%	1.43	15.76
06.2.7	**Propellant**			**425.41**	**0.0%**	**0.00**	**425.41**
06.2.7.a	**Propellant (EP)**			359.53	0.0%	0.00	359.53
06.2.7.a.a	Primary EP Propellant Used	1	331.06	331.06	0.0%	0.00	331.06
06.2.7.a.b	Primary EP Propellant Residulals (Unused)	1	28.47	28.47	0.0%	0.00	28.47
06.2.7.a.c	Primary EP Propellant Performance Margin (Unused)	0	0.00	0.00	0.0%	0.00	0.00
06.2.7.b	**Propellant (Chemical)**			0.00	0.0%	0.00	0.00
06.2.7.c	**RCS Propellant**			65.58	0.0%	0.00	65.58
06.2.7.c.a	RCS Used	1	63.98	63.98	0.0%	0.00	63.98
06.2.7.c.b	RCS Residuals	1	1.60	1.60	0.0%	0.00	1.60
06.2.7.d	RCS Pressurant	1	0.31	0.31	0.0%	0.00	0.31
06.2.8	**Structures and Mechanisms**			**148.88**	**18.0%**	**26.80**	**175.68**
06.3	**Sample Return Craft**			**93.30**	**18.1%**	**16.91**	**110.21**

5.6.4 Propulsion and Propellant Management Trades

The primary trades of this study were done on thruster system choice. For each of these thruster types, the impact on thruster choice will be on the total amount of Xe propellant needed to complete the mission. The three types of electric propulsion systems traded out in this study were the following:

- NEXT (1+1)—***Case 1, 1a, 3a***
  - 425 kg Xe for mission
  - 7 kW
  - Larger footprint, more susceptible to contamination
- BPT-4000 (2+1)—***Case 2***
  - 720 kg Xe for mission
  - 6 kW
  - Smaller footprint, less susceptible contamination
- HiVHAC (2+1)—***Case 3***
  - 620 kg Xe for mission
  - 7 kW
  - Smaller footprint, less susceptible contamination

5.6.5 Propulsion and Propellant Management Analytical Methods

Because the propulsion subsystems were assembled from existing components where possible, the analysis performed consisted primarily of maintaining a mass roll up for the various subassemblies. The first propulsion operation was performed with a variety of electric propulsion thruster options. A commercial high power Hall Thruster was used to establish a baseline vehicle for mission analysis and mass assessment. Subsequent trades were performed with two different EP thruster technologies under development at NASA GRC. The performance and physical characteristics of these thrusters were obtained directly from their development groups. Additionally, propellant management systems (PMS) were used based on breadboard systems also currently being developed at GRC through the In Space Program office. Real data for the PMS were used where available. Otherwise, it was obtained from development reports.

The vehicle's attitude and RCS was comprised of technically mature components with flight history. This propulsion system was a mass roll up of physical characteristics obtained from hardware providers.

For the 'delicate' landing operation, a notional cold-gas thruster-based propulsion system was developed. These cold-gas thrusters use the Xe propellant from the electric propulsion subsystem to provide the very small and controllable thrust levels required for landing on an asteroid (see Main Engine line in table 5.11). Commercially available nitrogen-based cold gas thrusters were used for mass estimates while the propellant management subsystem for the landing thruster pod was sized similarly to the other PMS elements.

The primary analysis that was actively performed was to determine the propellant tanks sizes based on propellant conditions over the mission duration. The tank requirements were determined using propellant density and storage pressure through Hoop Stress Analysis. These requirements were then used to select the best match from the PSI and Arde, Inc storage tank catalogs. Thermal control elements (heaters, insulation) were then added based on surface area of tanks and propellant lines

Once the storage tank(s) were selected, the helium pressurization requirements were determined. A conventional He pressurization system configuration was used, based on our experience with previous lander and Orion Service Module studies.

5.6.5.2 The NEXT Thruster Characteristics

- 0.54 to 6.9 kW thruster input power
- Ring-cusp electron bombardment discharge chamber
- 36 cm beam diameter, 2-grid ion optics
- Beam current at 6.9 kW: 3.52 A
- Maximum specific impulse > 4170 sec
- Maximum thrust > 236 mN
- Peak efficiency > 70%
- Xe throughput > 300 kg, (450 kg is the qualification level)
  - Analysis-based capability >450 kg
- Thruster Mass is 12.7 kg (13.5 kg with cable harnesses)

Shown in Figure 5.22 is the NEXT thruster in the Prototype Model Thruster (PM1) in Performance Acceptance Test.

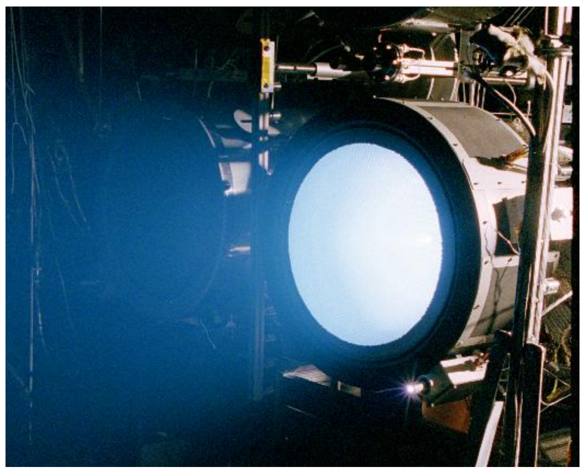


Figure 5.22—PM 1 Performance Acceptance Test

Gimbal Overview

- Breadboard gimbal
  - Designed and fabricated by Swales Aerospace
  - Flight-like design using JPL-approved materials with certifications
- Stepper motors have space-rated option
  - Mass < 6 kg
  - Two-axis range of motion: ±19°, ±17°
- Successful functional testing with PM1 engine
- Gimbal passed two qual-level vibration tests and low-level shock tests with minor issues (fastener backout)
- Good baseline—few if any modifications needed to move into qual program
  - Need to perform torque margin tests with harness and propellant line routing

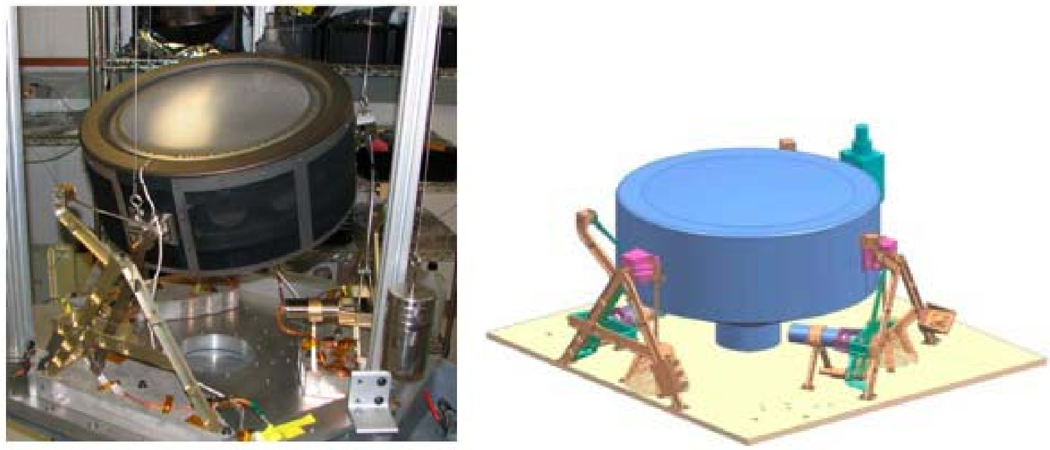


Figure 5.23—Gimbal Performance Test and CAD Illustration

NEXT Power Processing Unit (PPU)

- EM PPU build by L3 Communications Corporation (L3) ETI
- Modular beam supply and improved packaging provides performance and predictability benefits over NSTAR approach
- Digital Control Interface Unit (DCIU) to be integrated in next development phase

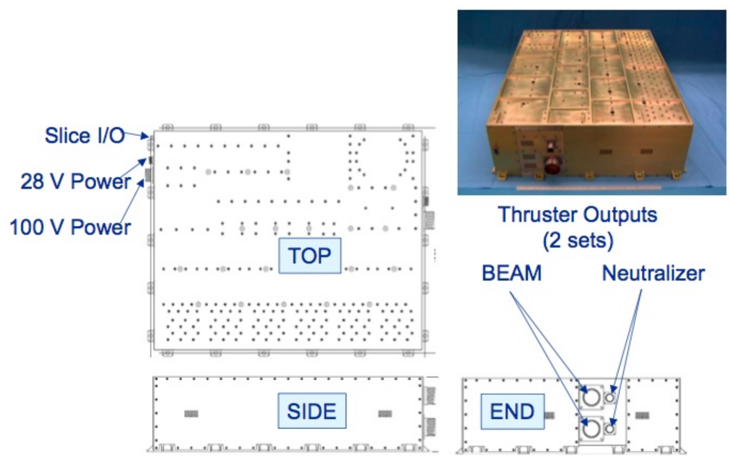


Figure 5.24—NEXT Thruster PPU schematic

5.6.5.3 NEXT Propellant Management System (PMS)

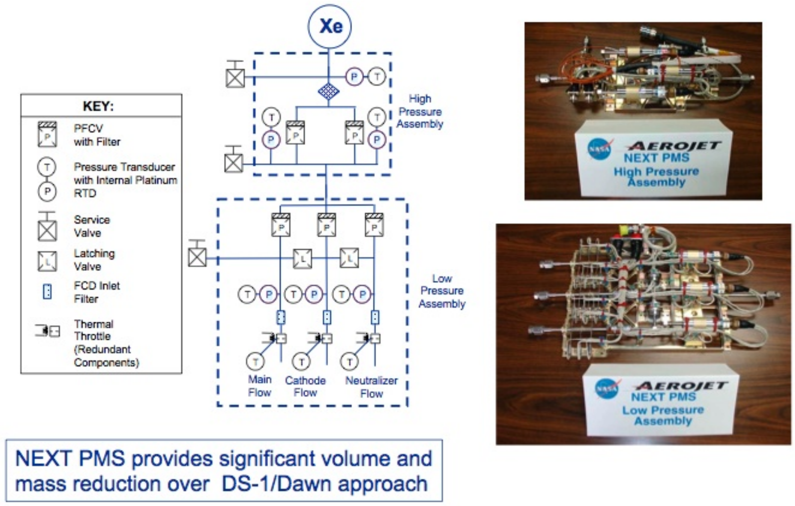


Figure 5.25—NEXT Propellant Management System

- All PMS assemblies are complete
  - Two HPA’s, one Flight-like
  - Three LPA’s, one Flight-like
  - Nonflight assemblies are identical except for use of lower cost equivalent parts
- All assemblies have completed functional tests
- Flight-like LPA and HPA successfully completed qual-level vibration testing and post-vibe functionals
- Qual-level thermal/vacuum testing is pending
- Xe Storage tanks based on COTS unit from ATK-PSI Inc.
  - Carbon Overwrapped tank with Ti liner
  - Derived from model no. 80465-1
    - Size: 0.42 m by 0.75 m L (16.5 in. by 29.6 in. L)
    - Manufacturers expected operating pressure (MEOP)/burst pressure = 198.2/310.3 bar (2875/4500 psig)
  - Minor size changes to match propellant load
- Hydrazine storage tank based on COTS unit design from ATK-PSC Inc.
  - Ti metallic tanks w/polymer diaphragm for blow-down pressurization
    - Size: 0.47 m dia. (18.6 in. diameter)
  - Derived from ATK/PSI Model No. 80439-1

Figure 5.26—Hydrazine Tank (Left), Xe Tank (Right)

5.6.6 Propulsion and Propellant Management Risk Inputs

No Risks gathered in this study.

5.6.7 Propulsion and Propellant Management Recommendation

Design is recommended.

5.7 Thermal Control

Objective: To provide spreadsheet based models capable of estimating the mass and power requirements of the various thermal systems. The thermal modeling provides power and mass estimates for the various aspects of the vehicle thermal control system based on a number of inputs related to the vehicle geometry, flight environment and component size. The system consists of the following elements

- Electric heaters
- MLI
- Thermal paint
- Radiator with louvers
- Thermal Control System (sensors, switches, data acquisition)

5.7.1 Thermal Requirements

The thermal requirements for the mission were to provide a means of cooling and heating of the S/C equipment during transit to and operation on the Asteroid's surface in order to remain within their maximum and minimum temperature requirements.

The maximum heat load to be rejected by the thermal system was 586 W, and the desired operating temperature for the electronics was 300 and 250 K for S/C structure. The S/C was required to survive and operate through any nighttime or shadow periods; therefore a heating system was also required.

5.7.2 Thermal Assumptions

The assumptions utilized in the analysis and sizing of the thermal system were based on the operational environment, both in transit to the asteroids and operation on the asteroid's surface. The following assumptions were utilized to size the thermal system.

- Moon surface operation: Day and night
- Radiator designed to see deep space with minimal view factors to the asteroid surface.

- The maximum angle of the radiator to the Sun was 25°.
- The radiator temperature was 320 K.
- A redundant radiator was used to account for vehicle orientation on the surface and to increase overall reliability.
- MLI was used to insulate the S/C to minimize heat transfer to and from the surroundings.
- Electric heaters and the radiator louvers were used to maintain the desired internal temperature of the S/C

5.7.3 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 throughout shadow or nighttime periods.

Excess heat is collected from a series of aluminum 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 communications box and are integrated to the radiator are protected with a micro meteor shield. The system utilize a louver system on the radiators to regulate the internal temperature and to insulate the radiators during the asteroid nighttime.

Two radiators were used to provide redundancy and margin as well as account for the unknown landing orientation of the S/C. This added margin insures against unforeseen heat loads, degradation of the radiator due to asteroid dust buildup and increased view factor toward any other thermally hot body not accounted for in the analysis.

Table 5.11—Thermal MEL for Baseline Case 1

WBS	Description	QTY	Unit Mass	CBE Mass	Growth	Growth	Total Mass
Number	NEA Sampler (August 2008) - Case 1		(kg)	(kg)	(%)	(kg)	(kg)
06	**NEA Sampler Spacecraft**			1138.27	9.9%	113.05	1251.32
06.1	**Science Payload**			25.20	16.1%	4.06	29.26
06.2	**Lander Spacecraft**			1019.77	9.0%	92.07	1111.85
06.2.1	**Attitude Determination and Control**			44.28	21.1%	9.36	53.64
06.2.2	**Command and Data Handling**			29.30	27.2%	7.96	37.26
06.2.3	**Communications and Tracking**			24.20	27.1%	6.55	30.75
06.2.4	**Electrical Power Subsystem**			115.50	15.7%	18.19	133.69
06.2.5	**Thermal Control (Non-Propellant)**			40.11	18.0%	7.22	47.33
06.2.5.a	**Active Thermal Control**			6.65	18.0%	1.20	7.85
06.2.5.a.a	Heaters	20	0.25	5.00	18.0%	0.90	5.90
06.2.5.a.b	Thermal Control/Heaters Circuit	2	0.20	0.40	18.0%	0.07	0.47
06.2.5.a.c	Data Acquisition	1	1.00	1.00	18.0%	0.18	1.18
06.2.5.a.d	Thermocouples	25	0.01	0.25	18.0%	0.05	0.30
06.2.5.a.e	Misc#1	1	0.00	0.00	18.0%	0.00	0.00
06.2.5.a.f	Misc#2	1	0.00	0.00	18.0%	0.00	0.00
06.2.5.b	**Passive Thermal Control**			26.87	18.0%	4.84	31.71
06.2.5.b.a	Heat Sinks	2	3.46	6.93	18.0%	1.25	8.17
06.2.5.b.b	Heat Pipes	1	2.93	2.93	18.0%	0.53	3.46
06.2.5.b.c	Radiators	1	10.06	10.06	18.0%	1.81	11.87
06.2.5.b.d	MLI (Multi Layer Insulation)	1	3.84	3.84	18.0%	0.69	4.53
06.2.5.b.e	Temperature sensors	50	0.01	0.50	18.0%	0.09	0.59
06.2.5.b.f	Phase Change Devices	1	0.00	0.00	18.0%	0.00	0.00
06.2.5.b.g	Thermal Coatings/Paint	1	0.95	0.95	18.0%	0.17	1.12
06.2.5.b.h	Micro Meteor shielding	1	0.00	0.00	18.0%	0.00	0.00
06.2.5.b.i	Spacecraft RTG MLI	1	0.00	0.00	18.0%	0.00	0.00
06.2.5.b.j	Spacecraft Engine MLI	1	1.66	1.66	18.0%	0.30	1.96
06.2.5.c	**Semi-Passive Thermal Control**			6.59	18.0%	1.19	7.78
06.2.5.c.a	Louvers	1	5.79	5.79	18.0%	1.04	6.84
06.2.5.c.b	Thermal Switches	4	0.20	0.80	18.0%	0.14	0.94
06.2.5.c.c	Misc#1	0	0.00	0.00	18.0%	0.00	0.00
06.2.5.c.d	Misc#2	0	0.00	0.00	18.0%	0.00	0.00
06.2.5.c.e	Misc#3	0	0.00	0.00	18.0%	0.00	0.00
06.2.6	**Propulsion**			192.10	8.3%	16.00	208.09
06.2.7	**Propellant**			425.41	0.0%	0.00	425.41
06.2.8	**Structures and Mechanisms**			148.88	18.0%	26.80	175.68
06.3	**Sample Return Craft**			93.30	18.1%	16.91	110.21

5.7.4 Thermal Trades

No significant design trades were made between components of the thermal control system.

5.7.5 Thermal Analytical Methods

The analysis performed to size the thermal system is based on first principle heat transfer from the S/C to the surroundings. This analysis takes into account the design and layout of the thermal system and the thermal environment to which heat is being rejected to or insulated from.

Environmental Models

Solar Intensity Based on S/C Location components were sized for worst case operating conditions, Heat Rejection: Near Earth, Minimum Temperature: near Earth Asteroid Orbital Location

Systems Modeled

- Micro meteor shielding on radiator
- Radiator panels
- Thermal control of propellant lines and tanks
- S/C insulation
- Avionics, and Power Management and Distribution (PMAD) cooling

Table 5.12—Thermal System Data Exchange

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

Radiator Sizing

The radiator panel area has been modeled along with an estimate of its mass. The model was based on first principles analysis of the area needed to reject the identified heat load to space. From the area, a series of scaling equations were used to determine the mass of the radiator within the asteroid environment. Asteroid orbit 1 AU thermal environment was used to size the radiator.

Table 5.13—Thermal System Radiator Sizing Assumptions

Variable	Value
Radiator solar absorptivity	0.14
Radiator emissivity	0.84
Radiator Sun angle	70°
Radiator operating temperature	320 k
Total radiator dissipation power	656.5 W
View Factor to SA	0.10
View Factor to Earth	0.10

Louvers are active or passive devices that regulate the amount of heat rejected by the radiator. Active controlled louvers use temperature sensors and actuators to control the louver position. Passive controlled louvers commonly use a bimetallic spring that opens and closes the louver based on temperature. The louver specific mass is 4.5 kg/m^2

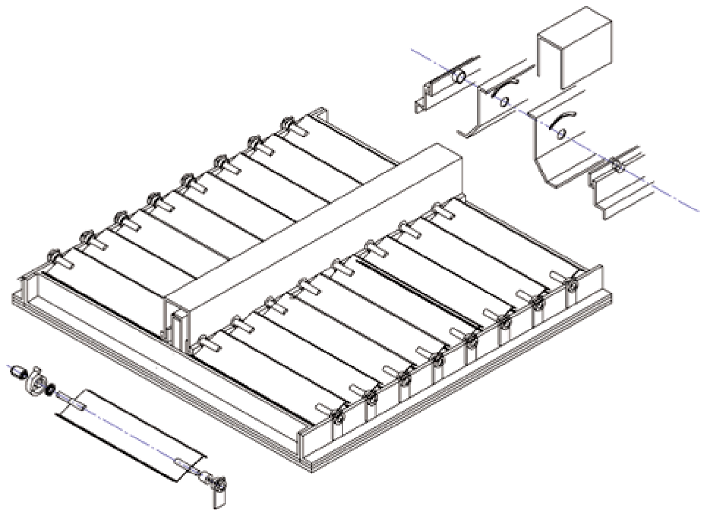


Figure 5.27—Schematic of the Louver Prototype

Thermal Analysis Propellant Lines and Tanks

Power requirements and mass have been modeled. This modeling included propellant tank MLI and heaters and propellant line insulation and heaters.

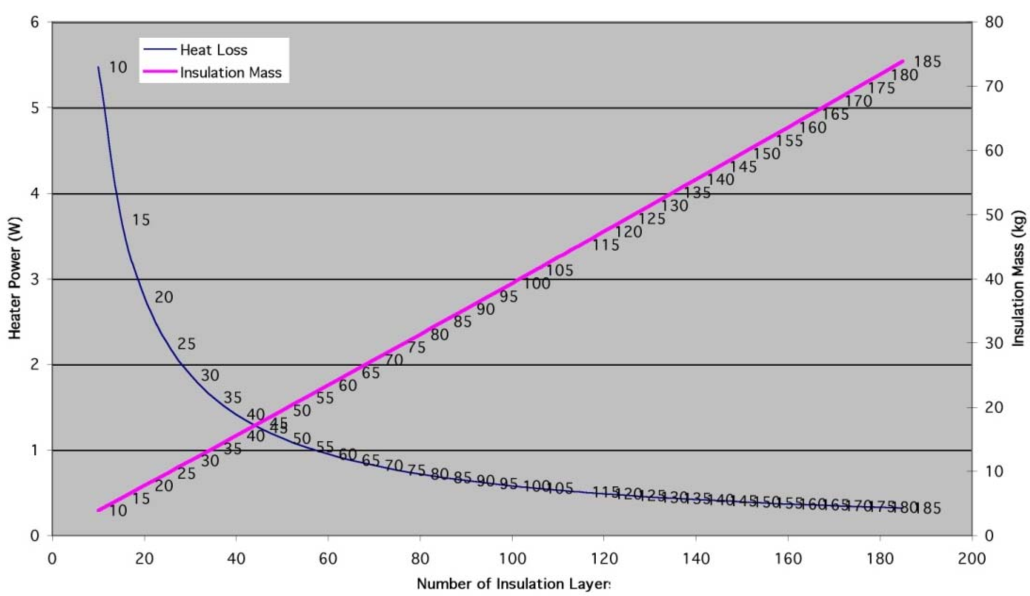


Figure 5.28—MLI

The model was based on a first principles analysis of the radiative heat transfer from the tanks and propellant lines through the S/C structure to space. The heat loss through the insulation set the power requirement for the tank and line heaters. The 1 AU thermal environment was used to calculate the heat loss. Assumptions used:

Table 5.14—Thermal System Tank Insulation Sizing Assumptions

Variable	Value
Tank surface emissivity (ε_t)	0.1
MLI emissivity (ε_i)	0.07
MLI material	Al
MLI material density (ρ_i)	2,770 kg/m^3
Internal tank temperature (T_i)	300 K
MLI layer thickness (t_i)	0.025 mm
Number of insulation layers (n_i)	10
MLI layer spacing (d_i)	1.0 mm
Tank immersion heater mass & power level	1.02 kg @ 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 & power	0.143 kg/m @ up to 39 W/m
Line foam insulation density	56 kg/m^3

Thermal Analysis—S/C Insulation

The mass of the S/C MLI insulation was modeled to determine the mass of the insulation and heat loss. The model was based on a first principles analysis of the heat transfer from the S/C through the insulation to space. Nighttime asteroid surface thermal environment was used to size the insulation. Two types of heaters were considered, Radioisotope Heater Unit (RHU), and electrical heaters. Assumptions used:

Table 5.15—Thermal System Tank Insulation Sizing Assumptions

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

Thermal Analysis—PMAD Cooling

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

Table 5.16—Thermal System PMAD Cooling Sizing Assumptions

Variable	Value
Cooling plate & lines material	Al
Cooling plate & lines material density	2,770 kg/m^3
Number of cooling plates	4
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

5.7.6 Thermal Risk Inputs

None

5.7.7 Thermal Recommendation

See Design

6.0 SOFTWARE COST ESTIMATION

6.1 Objectives

6.2 Assumptions

6.2.1 SW Sizing Assumptions

- Programming Language is Ada
- Two RAD750 processors
- Complex autonomy
- RTOS on RAD750 compact PCI processor board
- Ground software not included

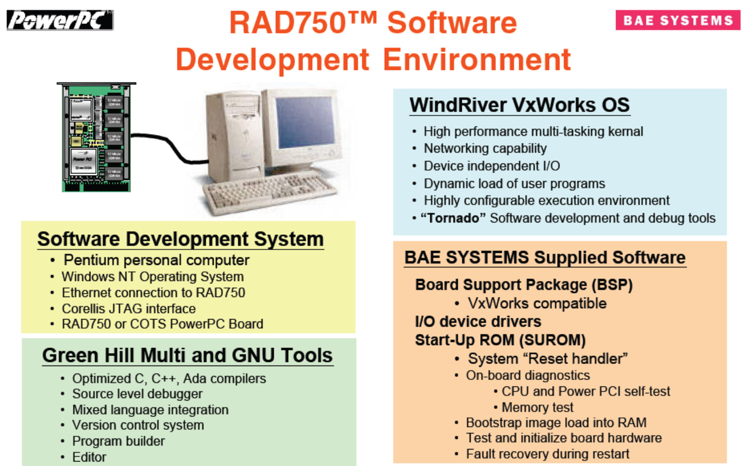


Figure 6.1—Typical RAD750 Processor and Software Environment

6.2.2 S/C Bus Functions

- Attitude Determination & Control (ACS): 5500 Single Line of Codes (SLOC)
  - This estimate includes: Sun sensor, Star Tracker, Rate Gyros, Complex Ephemeris, Kinematic Integration, Error Determination, Thruster Control, Reaction Wheel Control, Orbit Propagation
  - C&DH: 750 SLOCs
    - Command and telemetry processing
  - On-board autonomy (complex assumed): 4100 SLOCs
  - Fault detection (on-board systems monitoring and correction): 1650 SLOCs
  - Power Management and Thermal Control: 500 SLOCs

6.2.3 Payload Functions

- Sensor Processing: 670 SLOCs
- Data Reduction and Transmission: 200 SLOCs

6.3 Software Cost

Software Development

- Estimated 13370 SLOCs for on-board software
- Estimate at least 32,000 SLOCs for ground support software development, including

- Avionics simulation software
- Data collection and reduction software
- Ground station interface and simulation software

Support Equipment

- Estimate 4 engineering units (RAD750)
- RAD750 Compiler and run-time kernel
- Estimate 10 PCs/Workstations
- GUI builder, software development tools
- I/O cards, card cage, and drivers

Table 6.1—Software Costing estimate

S/C bus functions	Code size, K words	Code size by 4, K words	SLOC
C&DH (C&T processing)	2.8	11.0	750
AD&C (Sun sensor, Star Tracker, rate gyros, complex ephemeris, kinematic integration, error determination, thruster control, reaction wheel control, orbit propagation)	20.2	80.8	5500
Onboard autonomy ▪ Complex autonomy	15.0	60.0	4100
fault detection			
▪ Onboard system monitoring	4.0	16.0	1100
▪ Fault correction	2.0	8.0	550
Power management and thermal control ▪ Residing in onboard computers assumed	1.8	7.2	500
Subtotal	45.8	183.2	12500

Total (including payload functions): 13370 SLOCs.

7.0 COST, RISK AND RELIABILITY

7.1 Costing

The following section contains a ***draft*** Cost Estimate (all cost in FY09$M). The S/C cost estimates represents prime contractor cost. Assumes a proto-flight development. Flight spares are included where appropriate. Mission operations costs include 2 yr data analysis post sample return per NF 2009 AO (5.1.5.2—Curation of Returned Samples). Launch Services were not included per NF 2009 AO.

Table 7.1—NEXT 1+1, Added Science, Case 1a, LCC Cost (FY09 $M)

Near Earth Asteroid Sample Return NF Mission		
All costs in FY09 $M		
NASA Project Office/Technical Oversight	25	5% of all other costs
Phase A	21	5% of S/C cost and fee
S/C (without science instruments)	183	50 percentile estimate (DD and FH only)
Science instruments and SRC	98	Expanded science package (DD and FH only)
S/C systems integration and wraps	119	Includes integrating S/C and science instruments
S/C Prime Contractor Fee (10%)	30	Not applied to science instruments costs.
Mission operations	54	Based on DAWN (includes data analysis)
Life Cycle Cost Estimate	**530**	*Does not include Launch Services per 2009 NF AO
NF 2009 Cost Cap	**650**	NF 2009 AO ('09 NFAO)
Launch services: 4 m fairing/med performance	40	Per 2009 NFAO (5.9.2—Launch Services)
Propulsion system: NEXT	15	Per 2009 NFAO (5.9.3—Propulsion Technology Infusion)
Adjusted cost cap	**705**	
Reserves based on LCC estimate	**175**	**33% Reserve**

7.1.1 Detailed Cost Breakdown by WBS

WBS element	Element name	DDT&E total by $M	Flight hardware by $M	Mfg/DDT&E total by $M
06.1.1	**Science Payload**	**$48.5**	**$23.9**	**$72.4**
06.1.1	Arm Mounted Science Instruments			
06.1.1.a	Panoramic / microscopic color imager	$12.4	$5.3	$17.7
06.1.1.b	APXS	$3.2	$1.4	$4.6
06.1.1.c	LAMS	$11.1	$4.7	$15.8
06.1.2	Body Mounted Science Instruments			
06.1.2.a	Approach/Hazard Avoidance/Landing Lidar	$7.2	$6.2	$13.4
06.1.2.b	Neutron Detector/Gamma Ray Spect.	$1.4	$0.6	$2.1
06.1.2.c	wide narrow field imager	$6.0	$2.6	$8.5
06.1.2.d	Ground penetrating radar	$7.2	$3.1	$10.3
06.2.1	**Attitude Determination and Control**	**$17.1**	**$13.1**	**$30.2**
06.2.1.a.a	Sun Sensors	$1.2	$2.9	$4.2
06.2.1.a.b	Reaction Wheels	$2.2	$2.4	$4.5
06.2.1.a.c	Star Trackers	$1.4	$1.7	$3.1
06.2.1.a.d	IMU	$5.6	$2.8	$8.4
06.2.1.a.e	Laser Altimeter (from Science Payload)	$6.7	$3.2	$9.9
06.2.2	**Command & Data Handling**	**$16.6**	**$4.6**	**$21.2**
06.2.2.a.a	Flight Computer	$2.3	$2.4	$4.7
06.2.2.a.c	Data Interface Unit	$0.5	$0.3	$0.8
06.2.2.a.e	Operations Recorder	$0.1	$0.1	$0.2
06.2.2.a.f	Command and Control Harness (data)	$6.2	$0.6	$6.8
06.2.2.a.g	Shared DPU (From APL Science Instruments)	$2.3	$1.2	$3.5
06.2.2.b	Instrumentation & Wiring	$0.0	$0.0	$0.0
	Flight Software/Firmware	**$5.1**	**$0.0**	**$5.1**
06.2.3	**Communications and Tracking**	**$17.2**	**$7.6**	**$24.8**
06.2.3.a	X/Ka High Gain Antenna			
06.2.3.a.a	Transmitter/Receiver	$2.1	$1.2	$3.3
06.2.3.a.b	Power Amp	$2.6	$1.7	$4.3
06.2.3.a.c	Switch Unit	$2.5	$1.0	$3.5
06.2.3.a.d	Antenna	$0.9	$0.4	$1.2
06.2.3.a.h	Cabling	$1.3	$0.2	$1.5
06.2.3.a.i	Diplexer	$0.5	$0.4	$1.0
06.2.3.a.j	Coupler	$1.0	$0.5	$1.5
06.2.3.a.k	Misc#1	$1.9	$0.7	$2.7
06.2.3.a.l	Misc#2	$0.3	$0.0	$0.3
06.2.3.b	Ka-band Antenna			
06.2.3.b.a	Transponder	$2.4	$0.0	$2.4
06.2.3.b.b	RF Assembly	$0.1	$0.0	$0.1
06.2.3.b.d	Antenna	$0.9	$1.1	$2.0
06.2.3.c.a	Coaxial Cable	$0.7	$0.3	$1.0
06.2.4	**Electrical Power Subsystem**	**$18.3**	**$9.9**	**$28.2**
06.2.4.a	Solar Arrays	$0.0	$0.0	$0.0
06.2.4.a.a	Solar Array Mass (cells and structure only)	$10.1	$6.4	$16.6
06.2.4.a.b	Solar Array Gimbals	$1.5	$0.8	$2.2
06.2.4.b.a	Power management/control electronics	$3.5	$1.5	$5.0
06.2.4.c	Power Cable and Harness Subsystem	$1.4	$1.0	$2.3
06.2.4.d	Battery System	$1.8	$0.3	$2.1
06.2.5	**Thermal Control (Non-Propellant)**	**$6.4**	**$1.2**	**$7.6**
06.2.5.a	Active Thermal Control	$0.5	$0.4	$0.9
06.2.5.b	Passive Thermal Control	$5.9	$0.7	$6.6
06.2.6	**Propulsion**	**$36.8**	**$15.3**	**$52.1**
06.2.6.a.a	Primary EP Thrusters	$8.5	$4.3	$12.8
06.2.6.a.c.c	Gimbal	$2.6	$1.7	$4.2
06.2.6.b.a	Xe propellant tank(s)	$0.3	$0.3	$0.7
06.2.6.b.b	High Pressure Feed System	$5.4	$1.4	$6.8
06.2.6.b.e	Temperature sensors	$0.1	$0.0	$0.1
06.2.6.c	Power Processing Unit (PPU)			
06.2.6.c.a	PPU Mass	$13.0	$5.5	$18.5
06.2.6.d.a.a	Main Engine	$0.0	$0.0	$0.1
06.2.6.f	Reaction Control System Hardware			
06.2.6.f.a	RCS Thruster Subassembly	$0.2	$0.1	$0.3
06.2.6.g	RCS Propellant Management			
06.2.6.g.a	Fuel Tanks	$0.3	$0.3	$0.5
06.2.6.g.c	Feed and Pressurizations Systems	$6.4	$1.7	$8.1
06.2.7	**Propellant**	**$0.0**	**$0.0**	**$0.0**

WBS element	Element name	DDT&E total by $M	Flight hardware by $M	Mfg/DDT&E total by $M
06.2.8	**Structures and Mechanisms**	**$32.3**	**$11.6**	**$43.9**
06.2.8.a	Structures	$10.6	$5.1	$15.7
06.2.8.b.f.a	Arm	$7.9	$3.4	$11.3
06.2.8.b.f.b	Mechanisms	$0.9	$0.5	$1.4
06.3	**Sample Return Craft**	**$12.8**	**$2.7**	**$15.5**
Subtotal		**$193.3**	**$87.1**	**$280.4**
SYSTEMS INTEGRATION		**$98.0**	**$20.9**	**$118.9**
1.6.2.2	IACO	$8.2	$3.4	$11.6
1.6.2.2	STO	$9.6	$0.0	$9.6
1.6.2.2	GSE Hardware	$17.1	$0.0	$17.1
1.6.2.1	SE&I	$29.8	$12.2	$42.0
1.6.2.1	PM	$18.0	$5.3	$23.2
1.6.2.2	LOOS	$15.3	$0.0	$15.3
TOTAL PRIME COST		$291.3	$108.0	$399.3

7.1.2 System Integration Wraps Defined

The **Integration, Assembly and Checkout (IACO)** element contains all labor and material required to physically integrate (assemble) the various subsystems into a total system. Final assembly, including attachment, and the design and manufacture of installation hardware, final factory acceptance operations, packaging/crating, and shipment are included. IACO charged to DDT&E represents those costs incurred for the integration, assembly, and checkout of major test articles. IACO charged to the flight unit includes those same functions applied to the actual flight unit.

This item excludes the engineering effort required to establish the integration, assembly, and checkout procedures necessary for this effort. These engineering efforts are covered under systems engineering and integration.

The **System Test Operations (STO)** element includes development testing and the test effort and test materials required for qualification and physical integration of all test and qualification units. Also included is the design and fabrication of test fixtures.

Specifically included are tests on all STO to determine operational characteristics and compatibility with the overall system and its intended operational parameters. Such tests include operational tests, design verification tests, and reliability tests. Also included are the tests on systems and integrated systems to verify acceptability for required mission performance. These tests are conducted on hardware that has been produced, inspected, and assembled by established methods meeting all final design requirements. Further, system compatibility tests are included, as well as, functions associated with test planning and scheduling, data reduction, and report preparation.

Functional elements associated with **Ground Support Equipment (GSE)** include the labor and materials required to design, develop, manufacture, procure, assemble, test, checkout, and deliver the equipment necessary for system level final assembly and checkout. Specifically, the equipment utilized for integrated and/or electrical checkout, handling and protection, transportation, and calibration, and items such as component conversion kits, work stands, equipment racks, trailers, staging cryogenic equipment, and many other miscellaneous types of equipment are included.

Specifically excluded is the equipment designed to support only the mission operational phase.

The functions included in the **Systems Engineering and Integration (SE&I)** element encompass: (1) the system engineering effort to transform an operational need into a description of system requirements and/or a preferred system configuration; (2) the logistics engineering effort to define, optimize, and integrate logistics support considerations to ensure the development and production of a supportable and cost effective system; and (3) the planning, monitoring, measuring, evaluating, and directing of the overall technical program. Specific functions include those for control and direction of engineering activities, cost/performance trade-offs, engineering change support and planning studies, technology utilization, and the engineering required for safety, reliability, and quality control and assurance. Also included is the effort for system optimization, configuration requirements analyses, and the submittal and maintenance of Interface Control Documents (ICDs).

Excluded from the SE&I element are those functions which are identifiable to subsystem SE&I.

Elements included in the **Program Management (PM)** function consist of the effort and material required for the fundamental management direction and decision-making to ensure that a product is developed, produced, and delivered per requirements

Specifically included are direct charges for program administration, planning and control, scheduling and budgeting, contracts administration, and the management functions associated with engineering, manufacturing, support, quality assurance, configuration and project control, and documentation.

The PM element sums all of the effort required for planning, organizing, directing, coordinating, and controlling the project to help ensure that overall objectives are accomplished. This element also includes the effort required to coordinate, gather, and disseminate information.

Excluded from the PM element are those functions commonly charged to subsystem level activities.

7.2 Risk Analysis and Reduction

7.2.1 Assumptions

7.2.2 Risk List

7.2.3 Risk Summary

Risks

- Dust and debris impact on
- Flight system
- Instruments
- Gridded Ion thrusters
- SA

Mitigation

- Single Fault Tolerant Design for most flight systems
- Mission Scenario
  - Spiral down and Map Asteroid 1
  - Sample Asteroid 1
  - Perform Asteroid 1 sample return at Earth
  - Repeat for second asteroid
  - Ensures first asteroid sample not at risk during second asteroid sampling
- SA away from surface

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

7.3 Reliability

None performed for this study.

8.0 TRADE SPACE ITERATIONS

Two different trade options were explored in the five cases run on the multiple small body sample return study. The first option looked at using different thruster technologies. Cases 1, 2, and 3 used the three different thruster technologies NEXT, BPT-4000 and HiVAC respectively. A secondary set of trades was run on the science payload and the SRC to explore the ceiling and basement of science capabilities of this mission. In cases 1a and 3a, a slightly larger single SRC was returned to the Earth from the Near Earth Asteroids, rather than two SRC. Both cases used the NEXT thruster but case 1a delivered a larger science payload while case 3a delivered the standard baseline science package used in cases 1, 2 and 3.

The baselined case 1a and case 3a returned a single sample capsule of a slightly larger size than cases 1, 2 and 3.

Table 8.1—Spacecraft Case Comparison

	Case 1	Case 1a	Case 2	Case 3	Case 3a
Launch vehicle	Atlas 401	Atlas 401	Atlas 521	Atlas 401	Atlas 401
Thruster	NEXT	NEXT	BPT-4000	HiVAC	NEXT
Science payload	Baseline science package	Super science package	Baseline science package	Baseline Science package	Baseline science package
SRC	2 full size	1 slightly larger size	2 full size	2 full size	1 slightly larger size

Table 8.2—Spacecraft Total Mass Comparison

		Spacecraft Total Mass Comparison				
		Case 1	Case 1a	Case 2	Case 3	Case 3a
WBS	Main Subsystems	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)
01	**Moon Sampler Spacecraft**	**1251.3**	**1262.8**	**1561.8**	**1416.8**	**993.8**
06.1	Science Payload	**29.3**	**60.3**	**29.3**	**29.3**	**29.3**
06.2	*Moon Sampler Lander*	***1111.8***	***1141.0***	***1422.3***	***1277.3***	***903.1***
06.2.1	Attitude Determination and Control	**53.6**	**53.6**	**53.6**	**53.6**	**53.6**
06.2.2	Command and Data Handling	**37.3**	**37.3**	**37.3**	**37.3**	**37.3**
06.2.3	Communications and Tracking	**30.7**	**30.7**	**30.7**	**30.7**	**30.7**
06.2.4	Electric Power	**133.7**	**133.7**	**133.7**	**133.7**	**133.7**
06.2.5	Thermal Control	**47.3**	**47.3**	**47.3**	**47.3**	**47.3**
06.2.6	Propulsion	**208.1**	**209.3**	**217.2**	**179.5**	**154.3**
06.2.7	Propellant	**425.1**	**451.8**	**726.9**	**620.5**	**272.8**
06.2.8	Structures and Mechanisms	**175.7**	**176.9**	**175.2**	**174.3**	**173.2**
06.3	*Sample Return Craft (total, empty)*	***110.2***	***61.4***	***110.2***	***110.2***	***61.4***
06.3.1	Electrical Power Subsystem	**7.2**	**7.2**	**7.2**	**7.2**	**7.2**
06.3.2	Thermal Control (Non-Propellant)	**5.4**	**5.4**	**5.4**	**5.4**	**5.4**
06.3.3	Structures and Mechanisms	**97.6**	**48.8**	**97.6**	**97.6**	**48.8**
	Estimated Spacecraft Dry Mass	**826.2**	**810.9**	**834.9**	**796.3**	**721.0**
	Estimated Spacecraft Wet Mass	1251.3	1262.8	1561.8	1416.8	993.8
System LeveL Growth Calculations						
	Dry Mass w/ Desired System Level Growth	**927.1**	**913.7**	**937.5**	**884.9**	**799.5**
	Additional Growth (carried at system level)	100.9	102.8	102.6	88.6	78.5
	Total Wet Mass with Growth	**1352.2**	**1365.5**	**1664.4**	**1505.4**	**1072.3**
	Available Launch Performance to C3 (kg)	1375.4	1421.9	1686.4	1600.0	1200.0
	Launch margin available (kg)	**23.2**	**56.4**	**22.0**	**94.6**	**127.7**
	Estimated Spacecraft Inert Mass (for traj.)	**1021.5**	**1009.8**	**1058.6**	**996.0**	**856.0**
			0	*0*	*0*	*0*
	Sample Return Craft Total Mass	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)	Total Mass (kg)
	Estimated Sample Return Craft Mass	***110.2***	***61.4***	***110.2***	***110.2***	***61.4***
	Total with System Level Growth	**121.3**	**67.5**	**121.3**	**121.3**	**67.5**
	Number of Sample Return Craft	**2**	**1**	**2**	**2**	**1**
	Total Mass per Sample Return Craft (empty)	**60.6**	**67.5**	**60.6**	**60.6**	**67.5**
	Total Mass, Sample Returned	1	1	1	1	1
	Total Mass, Sample Return Capsule (Full)	**61.6**	**68.5**	**61.6**	**61.6**	**68.5**

8.1 Case 1—Baseline

Case 1 using the NEXT thruster system for electric propulsion was chosen as the baseline for this study report and is documented in detail in the subsystem sections.

Table 8.3—Case 1—System Summary

GLIDE container	*Mars_Moons_Sampler: NEA_sampler_case1*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**CBE Mass (kg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
01	**Asteroids Sampler Spacecraft**	**1138.3**	**113.0**	**1251.3**	
06.1	Science Payload	25.2	4.1	**29.3**	16%
06.2	*Asteroids Sampler Lander*	***1019.8***	***92.1***	***1111.8***	
06.2.1	Attitude Determination and Control	44.3	18.7	**53.6**	42%
06.2.2	Command and Data Handling	29.3	8.0	**37.3**	27%
06.2.3	Communications and Tracking	24.2	6.5	**30.7**	27%
06.2.4	Electric Power	115.5	18.2	**133.7**	16%
06.2.5	Thermal Control	40.1	7.2	**47.3**	18%
06.2.6	Propulsion	192.1	16.0	**208.1**	8%
06.2.7	Propellant	425.1			
06.2.8	Structures and Mechanisms	148.9	26.8	**175.7**	18%
06.3	*Sample Return Craft (total, empty)*	***93.3***	***16.9***	***110.2***	18%
06.3.1	Electrical Power Subsystem	6.0	1.2	**7.2**	20%
06.3.2	Thermal Control (Non-Propellant)	4.6	0.8	**5.4**	18%
06.3.3	Structures and Mechanisms	82.7	14.9	**97.6**	18%
	Estimated Spacecraft Dry Mass	**713**	113	**826.2**	**16%**
	Estimated Spacecraft Wet Mass	1138	113	1251.3	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	713	214	**927.1**	**30%**
	Additional Growth (carried at system level)		101		**14%**
	Total Wet Mass with Growth	1138	214	**1352.2**	
	Available Launch Performance to C3 (kg)			1375.4	
	Launch margin available (kg)			**23.2**	
	Estimated Spacecraft Inert Mass (for traj.)	808	214	**1021.5**	

Table 8.4—Case 1—Sample Return Craft Summary

Sample Return Craft Total Mass	CBE Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
Estimated Sample Return Craft Mass	***93.3***	***16.9***	***110.2***	18%
Total with System Level Growth	93	28	**121.3**	**30%**
Number of Sample Return Craft	**2**			
Total Mass per Sample Return Craft (empty)	**60.6**	kg		
Total Mass, Sample Returned	1	kg		
Total Mass, Sample Return Capsule (Full)	**61.6**	kg		

8.2 Case 1a

Case 1a uses the same NEXT thrusters to perform all the interplanetary burns as Case 1. The mission still carries a single SRC and takes science from the two asteroids chosen in the mission analysis section. The change in this case is that the science payload used on cases 1 through 3, has been increased on Case 1a to what is called a super science payload. Note that in this table, the word "Moon" appears. This table was built off of the Mars Moon Sampler mission done previously by COMPASS and the words were not changed in time to make it into this report.

Table 8.5—Case 1a—System Summary

GLIDE container:	*Mars_Moons_Sampler: NEA_sampler_case1a*				
	Spacecraft Master Equipment List Rack-up (Mass)			**COMPASS S/C Design**	
WBS	**Main Subsystems**	**CBE Mass (kg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
01	**Moon Sampler Spacecraft**	**1154.7**	**108.1**	**1262.8**	
06.1	Science Payload	54.0	6.3	**60.3**	12%
06.2	*Moon Sampler Lander*	***1048.7***	***92.3***	***1141.0***	
06.2.1	Attitude Determination and Control	44.3	18.7	**53.6**	42%
06.2.2	Command and Data Handling	29.3	8.0	**37.3**	27%
06.2.3	Communications and Tracking	24.2	6.5	**30.7**	27%
06.2.4	Electric Power	115.5	18.2	**133.7**	16%
06.2.5	Thermal Control	40.1	7.2	**47.3**	18%
06.2.6	Propulsion	193.3	16.0	**209.3**	8%
06.2.7	Propellant	451.8			
06.2.8	Structures and Mechanisms	149.9	27.0	**176.9**	18%
06.3	*Sample Return Craft (total, empty)*	***52.0***	***9.5***	***61.4***	18%
06.3.1	Electrical Power Subsystem	6.0	1.2	**7.2**	20%
06.3.2	Thermal Control (Non-Propellant)	4.6	0.8	**5.4**	18%
06.3.3	Structures and Mechanisms	41.4	7.4	**48.8**	18%
	Estimated Spacecraft Dry Mass	**703**	108	**810.9**	**15%**
	Estimated Spacecraft Wet Mass	1155	108	1262.8	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	703	211	**913.7**	**30%**
	Additional Growth (carried at system level)		103		**15%**
	Total Wet Mass with Growth	1155	211	**1365.5**	
	Available Launch Performance to C3 (kg)			1421.9	
	Launch margin available (kg)			**56.4**	
	Estimated Spacecraft Inert Mass (for traj.)	799	211	**1009.8**	

Table 8.6—Case 1a—Sample Return Craft Summary

Sample Return Craft Total Mass	CBE Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
Estimated Sample Return Craft Mass	***52.0***	***9.5***	***61.4***	18%
Total with System Level Growth	52	16	**67.5**	**30%**
Number of Sample Return Craft	**1**			
Total Mass per Sample Return Craft (empty)	**67.5**	kg		
Total Mass, Sample Returned	1	kg		
Total Mass, Sample Return Capsule (Full)	**68.5**	kg		

8.3 Case 2

Case 2 ran the same mission to the same two asteroids using the BPT-4000 thruster model for the electric propulsion system. Two SRCs were returned to Earth as in case 1. Note that in this table, the word "Moon" appears. This table was built off of the Mars Moon Sampler mission done previously by COMPASS and the words were not changed in time to make it into this report.

Table 8.7—Case 2—System Summary

GLIDE container:	*Mars_Moons_Sampler: NEA_sampler_case2*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**CBE Mass (kg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
01	**Moon Sampler Spacecraft**	**1448.1**	**113.8**	**1561.8**	
06.1	Science Payload	25.2	4.1	**29.3**	16%
06.2	*Moon Sampler Lander*	***1329.6***	***92.8***	***1422.3***	
06.2.1	Attitude Determination and Control	44.3	18.7	**53.6**	42%
06.2.2	Command and Data Handling	29.3	8.0	**37.3**	27%
06.2.3	Communications and Tracking	24.2	6.5	**30.7**	27%
06.2.4	Electric Power	115.5	18.2	**133.7**	16%
06.2.5	Thermal Control	40.1	7.2	**47.3**	18%
06.2.6	Propulsion	200.5	16.8	**217.2**	8%
06.2.7	Propellant	726.9			
06.2.8	Structures and Mechanisms	148.5	26.7	**175.2**	18%
06.3	*Sample Return Craft (total, empty)*	***93.3***	***16.9***	***110.2***	18%
06.3.1	Electrical Power Subsystem	6.0	1.2	**7.2**	20%
06.3.2	Thermal Control (Non-Propellant)	4.6	0.8	**5.4**	18%
06.3.3	Structures and Mechanisms	82.7	14.9	**97.6**	18%
	Estimated Spacecraft Dry Mass	**721**	114	**834.9**	**16%**
	Estimated Spacecraft Wet Mass	1448	114	1561.8	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	721	216	**937.5**	**30%**
	Additional Growth (carried at system level)		103		**14%**
	Total Wet Mass with Growth	1448	216	**1664.4**	
	Available Launch Performance to C3 (kg)			1686.4	
	Launch margin available (kg)			**22.0**	
	Estimated Spacecraft Inert Mass (for traj.)	842	216	**1058.6**	

Table 8.8—Case 2—Sample Return Craft Summary

Sample Return Craft Total Mass	CBE Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
Estimated Sample Return Craft Mass	***93.3***	***16.9***	***110.2***	18%
Total with System Level Growth	93	28	**121.3**	**30%**
Number of Sample Return Craft	**2**			
Total Mass per Sample Return Craft (empty)	**60.6**	kg		
Total Mass, Sample Returned	1	kg		
Total Mass, Sample Return Capsule (Full)	**61.6**	kg		

8.4 Case 3

Case 2 ran the same mission to the same two asteroids using the HiVAC thruster model for the electric propulsion system. Two SRCs were returned to Earth as in case 1 and 2. Note that in this table, the word "Moon" appears. This table was built off of the Mars Moon Sampler mission done previously by COMPASS and the words were not changed in time to make it into this report.

Table 8.9—Case 3—System Summary

GLIDE container:	Mars_Moons_Sampler: NEA_sampler_case3				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**CBE Mass (kg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
01	**Moon Sampler Spacecraft**	**1301.2**	**115.6**	**1416.8**	
06.1	Science Payload	25.2	4.1	**29.3**	16%
06.2	*Moon Sampler Lander*	***1182.7***	***94.6***	***1277.3***	
06.2.1	Attitude Determination and Control	44.3	18.7	**53.6**	42%
06.2.2	Command and Data Handling	29.3	8.0	**37.3**	27%
06.2.3	Communications and Tracking	24.2	6.5	**30.7**	27%
06.2.4	Electric Power	115.5	18.2	**133.7**	16%
06.2.5	Thermal Control	40.1	7.2	**47.3**	18%
06.2.6	Propulsion	160.8	18.7	**179.5**	12%
06.2.7	Propellant	620.5			
06.2.8	Structures and Mechanisms	147.7	26.6	**174.3**	18%
06.3	*Sample Return Craft (total, empty)*	***93.3***	***16.9***	***110.2***	18%
06.3.1	Electrical Power Subsystem	6.0	1.2	**7.2**	20%
06.3.2	Thermal Control (Non-Propellant)	4.6	0.8	**5.4**	18%
06.3.3	Structures and Mechanisms	82.7	14.9	**97.6**	18%
	Estimated Spacecraft Dry Mass	**681**	116	**796.3**	**17%**
	Estimated Spacecraft Wet Mass	1301	116	1416.8	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	681	204	**884.9**	**30%**
	Additional Growth (carried at system level)		89		**13%**
	Total Wet Mass with Growth	1301	204	**1505.4**	
	Available Launch Performance to C3 (kg)			1600.0	
	Launch margin available (kg)			**94.6**	
	Estimated Spacecraft Inert Mass (for traj.)	792	204	**996.0**	

Table 8.10—Case 3—Sample Return Craft Summary

Sample Return Craft Total Mass	CBE Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
Estimated Sample Return Craft Mass	***93.3***	***16.9***	***110.2***	18%
Total with System Level Growth	93	28	**121.3**	**30%**
Number of Sample Return Craft	**2**			
Total Mass per Sample Return Craft (empty)	**60.6**	kg		
Total Mass, Sample Returned	1	kg		
Total Mass, Sample Return Capsule (Full)	**61.6**	kg		

8.5 Case 3a

Case 3 went to the same two asteroid targets, but this time only brought along minimal science package onboard and only returned one SRC to the Earth. The NEXT thruster system was used for electric propulsion. Note that in this table, the word "Moon" appears. This table was built off of the Mars Moon Sampler mission done previously by COMPASS and the words were not changed in time to make it into this report.

Table 8.11—Case 3a—System Summary

GLIDE container:	*Mars_Moons_Sampler: NEA_sampler_case3a*				
Spacecraft Master Equipment List Rack-up (Mass)				**COMPASS S/C Design**	
WBS	**Main Subsystems**	**CBE Mass (kg)**	**Growth (kg)**	**Total Mass (kg)**	**Aggregate Growth (%)**
01	**Moon Sampler Spacecraft**	**887.8**	**106.0**	**993.8**	
06.1	Science Payload	25.2	4.1	**29.3**	16%
06.2	*Moon Sampler Lander*	***810.6***	***92.5***	***903.1***	
06.2.1	Attitude Determination and Control	44.3	18.7	**53.6**	42%
06.2.2	Command and Data Handling	29.3	8.0	**37.3**	27%
06.2.3	Communications and Tracking	24.2	6.5	**30.7**	27%
06.2.4	Electric Power	115.5	18.2	**133.7**	16%
06.2.5	Thermal Control	40.1	7.2	**47.3**	18%
06.2.6	Propulsion	137.5	16.8	**154.3**	12%
06.2.7	Propellant	272.8			
06.2.8	Structures and Mechanisms	146.7	26.4	**173.2**	18%
06.3	*Sample Return Craft (total, empty)*	***52.0***	***9.5***	***61.4***	18%
06.3.1	Electrical Power Subsystem	6.0	1.2	**7.2**	20%
06.3.2	Thermal Control (Non-Propellant)	4.6	0.8	**5.4**	18%
06.3.3	Structures and Mechanisms	41.4	7.4	**48.8**	18%
	Estimated Spacecraft Dry Mass	**615**	106	**721.0**	**17%**
	Estimated Spacecraft Wet Mass	888	106	993.8	
System LeveL Growth Calculations					**Total Growth**
	Dry Mass Desired System Level Growth	615	184	**799.5**	**30%**
	Additional Growth (carried at system level)		78		**13%**
	Total Wet Mass with Growth	888	184	**1072.3**	
	Available Launch Performance to C3 (kg)			1200.0	
	Launch margin available (kg)			**127.7**	
	Estimated Spacecraft Inert Mass (for traj.)	672	184	**856.0**	

Table 8.12—Case 3a—Sample Return Craft Summary

Sample Return Craft Total Mass	CBE Mass (kg)	Growth (kg)	Total Mass (kg)	Aggregate Growth (%)
Estimated Sample Return Craft Mass	***52.0***	***9.5***	***61.4***	18%
Total with System Level Growth	52	16	**67.5**	**30%**
Number of Sample Return Craft	**1**			
Total Mass per Sample Return Craft (empty)	**67.5**	kg		
Total Mass, Sample Returned	1	kg		
Total Mass, Sample Return Capsule (Full)	**68.5**	kg		

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APPENDIX A—ACRONYMS AND ABBREVIATIONS

ACS	Attitude Control System
AO	Announcement of Opportunity
APXS	Alpha Particle X-Ray Spectrometer
ARU	Array Regulator Unit
AU	Astronomical Unit
BAE	British Aerospace
BCCU	Battery Charge Control Unit
BOL	beginning of life
C&DH	Command and Data Handing
C_3	Launch energy per unit mass
CBE	current best estimate
CCB	Common Core Booster
CEV	Crew Exploration Vehicle
CG	Center of Gravity
Comm	Communications
COMPASS	COllaborative Modeling and Parametric Assessment of Space Systems
COPV	Composite Overwrapped Pressure Vessel
COTS	commercial off-the-shelf
DCIU	Digital Control Interface Unit
DMR	Design for Minimum Risk
DOF	Degree of Freedom
DSN	Deep Space Network
DTE	direct to Earth
EELV	Evolved Expendable Launch Vehicle
ELV	Expendable Launch Vehicle
EM	Electro Magnetic
EP	Electric Propulsion
FEA	finite element analysis
FOM	figure of merit
GSFC	NASA Goddard Space Flight Center
GLIDE	GLobal Integrated Design Environment
GN&C	Guidance, Navigation and Control
GRC	NASA Glenn Research Center
GSE	Ground Support Equipment
HiVAC	High-Voltage Hall Accelerator
HPA	High Pressure Assembly
HQ	NASA Headquarters
IACO	Integration, Assembly and Checkout
IMDC	I M Design Center
IMU	Inertial Measuring Unit
IP	Internet protocol
JPL	NASA Jet Propulsion Laboratory
KSC	NASA Kennedy Space Center
LIDAR	Light Detection and Ranging
LPA	Low Pressure Assembly
LSP	Launch Service Program
LSTO	Launch Service Task Order
MEL	Master Equipment List
MEOP	Manufacturers Expected Operating Pressure
MLI	multilayer insulation
MGA	Mass Growth Allowance
MPU	Makeup Power Unit
NASA	National Aeronautics and Space Administration
Nav	navigation
NEA	Near Earth Asteroid
NEARER	Near Earth Asteroids Rendezvous and sample Earth Returns
NEXT	NASA Evolutionary Xe Thruster
NLS	NASA Launch Services
OTS	off-the-shelf
PEL	Power Equipment List
PICA	Phenolic Impregnated Carbon Ablator

PM	Program Management
PMAD	Power Management and Distribution
PMS	Propellant Management System
PM	preliminary model
PPM	Propellant and Propellant Management
PPU	power processing unit
RCS	Reaction Control System
RHS	radioisotope heater unit
S/C	spacecraft
SADA	SA Drive Assembly
SE&I	Systems Engineering and Integration
SEP	Solar Electric Propulsion
SLOC	single line of code
SN	signal-to-noise
SPACE	System Power Analysis for Capability Evaluation
SPU	solar power unit
SRC	sample return capsule
STO	System Test Operations
TDRSS	Tracking and Data Relay Satellite System
TPS	Thermal Protection System
TRL	Technology Readiness Level
TWTA	Traveling Wave Tube Amplifier
VACCO	Vacuum and Air Components Company of America

APPENDIX B—RENDERED DESIGN DRAWINGS

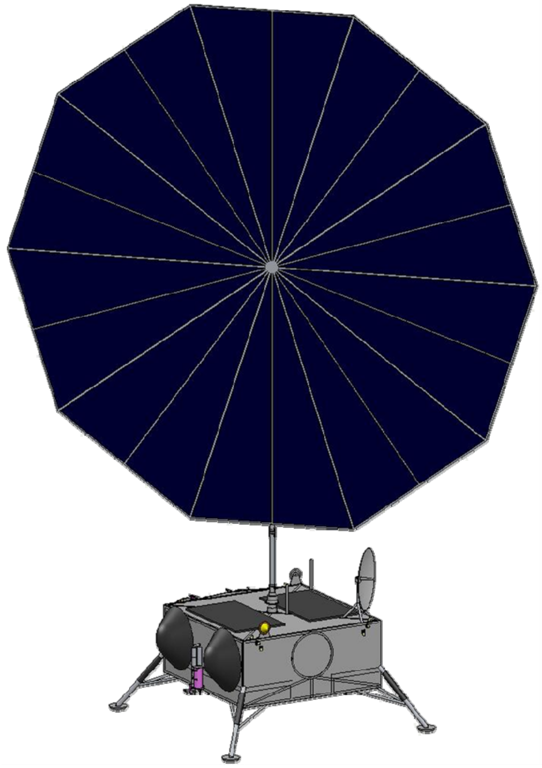

Figure B.1—Rendered Baseline Case 1—Deployed View

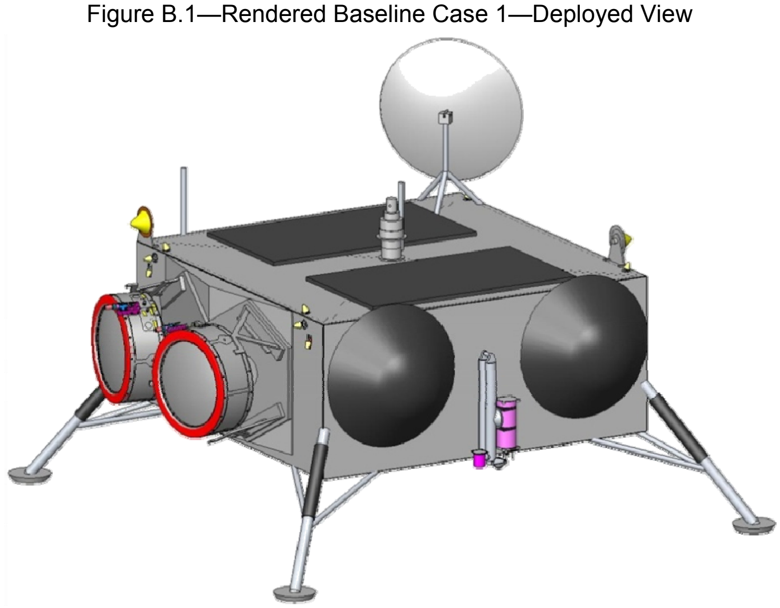

Figure B.2—Rendered Baseline Case 1—Close View of Main Body

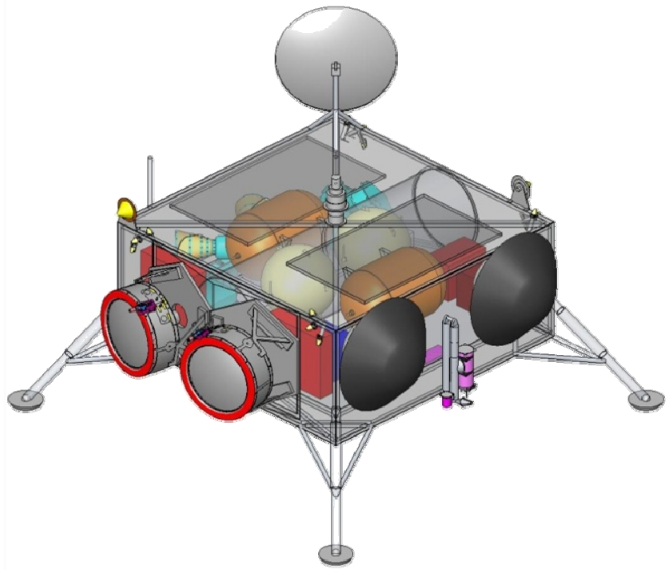

Figure B.3—Transparent Baseline Case 1—Main Bus View 1

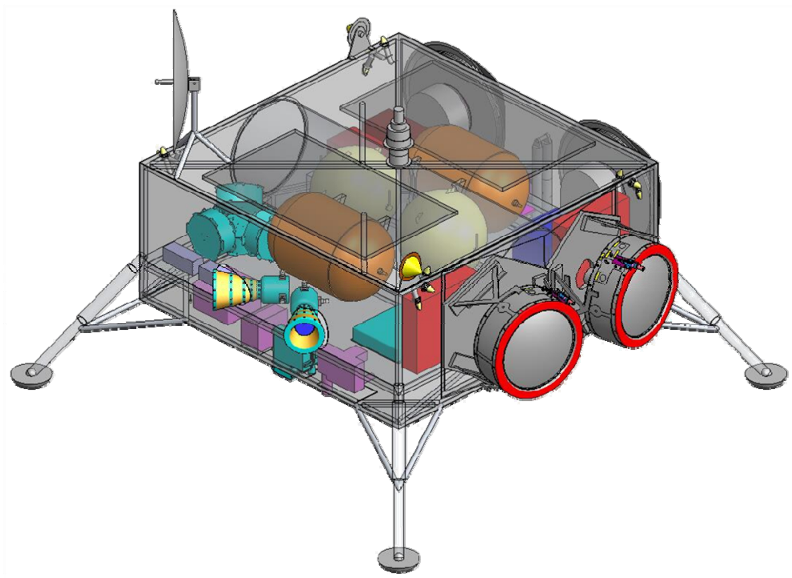

Figure B.4—Transparent Baseline Case 1—Main Bus View 2

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14. ABSTRACT In this study, the Collaborative Modeling for Parametric Assessment of Space Systems (COMPASS) team completed a design for a multi-asteroid (Nereus and 1996 FG3) sample return capable spacecraft for the NASA In-Space Propulsion Office. The objective of the study was to support technology development and assess the relative benefits of different electric propulsion systems on asteroid sample return design. The design uses a single, heritage Orion solar array (SA) (~6.5 kW at 1 AU) to power a single NASA Evolutionary Xenon Thruster ((NEXT) a spare NEXT is carried) to propel a lander to two near Earth asteroids. After landing and gathering science samples, the Solar Electric Propulsion (SEP) vehicle spirals back to Earth where it drops off the first sample's return capsule and performs an Earth flyby to assist the craft in rendezvousing with a second asteroid, which is then sampled. The second sample is returned in a similar fashion. The vehicle, dubbed Near Earth Asteroids Rendezvous and Sample Earth Returns (NEARER), easily fits in an Atlas 401 launcher and its cost estimates put the mission in the New Frontier's (NF's) class mission.						
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