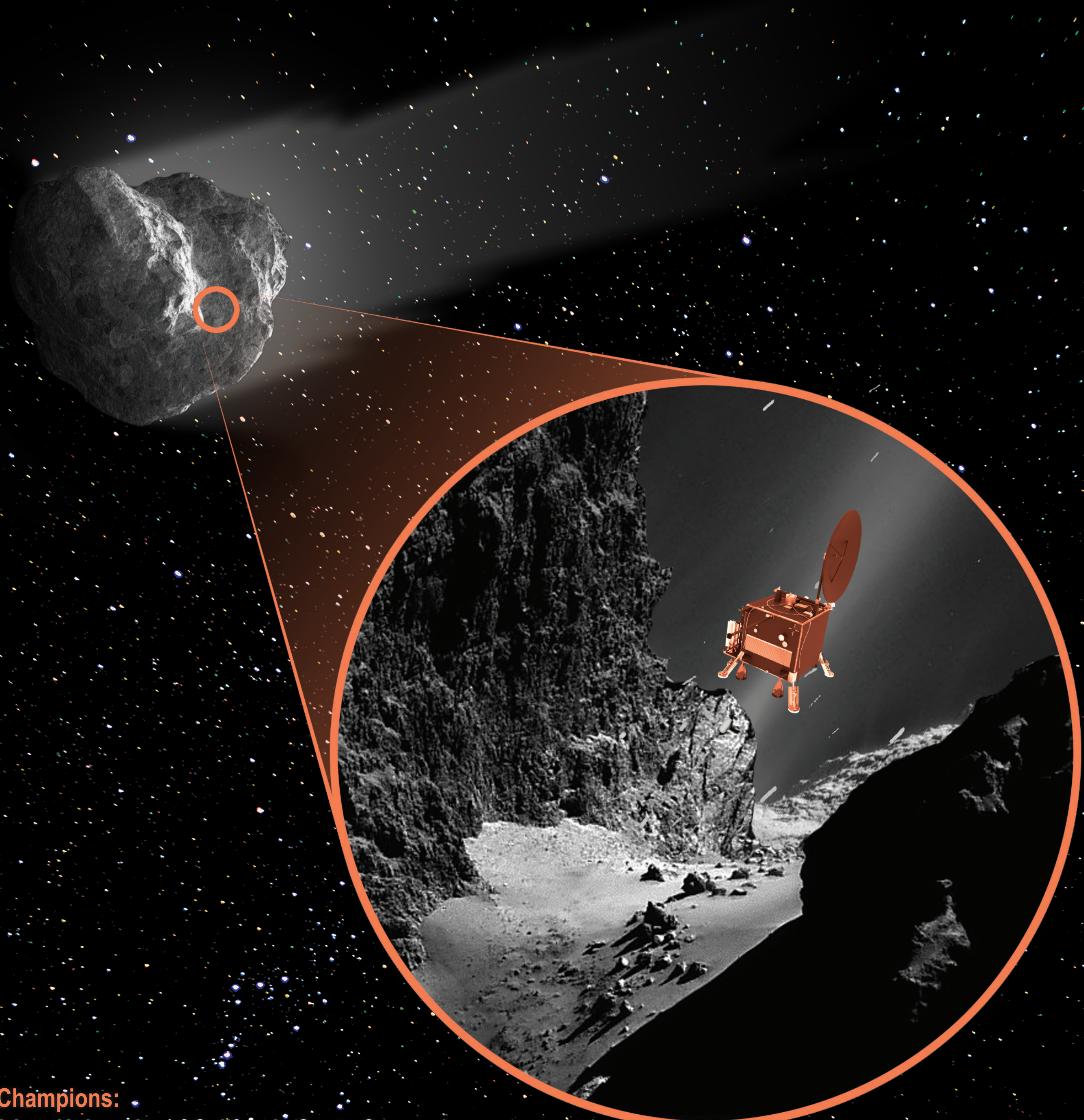
CORAL

Centaur ORbiter And Lander:

Mission Concept Study to Report to the NRC Planetary Science Decadal Survey



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EXECUTIVE SUMMARY

This mission concept demonstrates the feasibility of globally characterizing and obtaining landed *in situ* compositional measurements of a Centaur within a New Frontiers class mission. This study was conducted by Goddard Space Flight Center. Guidelines include a launch between 2036 and 2040 and a cost cap of $1.1B (FY25). In case New Frontiers funding is better aligned with earlier launch dates, launches between 2030 and 2035 were also examined.

The goal of the Centaur Orbiter and Lander (CORAL) mission is to measure the chemical and physical properties of a Centaur, which are small icy bodies from the Kuiper Belt, but they currently reside between Jupiter and Neptune. These dynamically evolved but compositionally primitive small bodies provide an opportunity to conduct a comprehensive study of the geochemical and physical properties of primordial ice-rich planetesimals, which trace the composition of nebular volatiles such as H_2O, CO_2, CO and NH_3. Prior mission concept studies and proposals have focused on Centaur orbiters and flybys for Discovery class missions. Adding landed geochemical analyses adds another layer of complexity and increases the costs substantially, thus requiring a New Frontiers class mission.

To meet the science objectives, the CORAL payload includes UV and IR spectrometers and high-resolution cameras to support global characterization of the surface composition of the target. Samples from the surface of the Centaur will be analyzed with a mass spectrometer, an X-ray fluorescence (XRF) spectrometer and a combined Raman and UV system. Samples would be collected from surface and subsurface depths and then distributed to the instruments via a carousel using a system similar to PlanetVac. The spacecraft also includes a magnetometer and a panoramic camera.

The core mission trade for the study is to identify centaurs and propulsion options that enable a lander mission. Centaur 2015 BQ311 was chosen out of several viable targets that allow for delivered masses greater than 2000 kg using a Falcon Heavy Expendable launch vehicle. The absolute magnitude and colors are known for 2015 BQ311, but otherwise it is not well-characterized. The estimated size of 2015 BQ311 indicates that it is comparable to that of Arrokoth, a cold classical Kuiper Belt Object, recently visited by New Horizons.

A mission to Centaur BQ311 requires that the spacecraft launch in January 2040 on a Falcon Heavy Expendable. The spacecraft spends 9 years in interplanetary transfer and rendezvous with BQ311 in January 2049. Proximity operations last approximately 4 years. After proximity operations are completed and the landing site is chosen, the spacecraft will land and carry out *in situ* compositional analyses for 8 weeks. The mission supports the option to continue analyses at landing site 1 or takeoff to a second landing site for additional analyses.

The total costs estimated for Phase A-D, including the full instrument payload, is $1.29B (FY25). These costs are likely over-estimated due to 50% reserves for high-heritage instruments. Descoping the UV spectrometer, the XRF, and the magnetometer bring mission costs down to $1.16B and does not affect the top priority science goals.

Overall, CORAL is a viable mission concept that fits within New Frontiers scope and has implications for the composition and evolution of icy planetesimals, which are important for understanding the evolution of the protoplanetary disk. This concept study was conducted with a goal of Concept Maturity Level (CML) 5; however, certain aspects of the design concept could be classified as a CML 4. It presents an implementation concept at the subsystem level, as well as science traceability, mission requirements traceability, key technologies, heritage, risks and mitigations. Detailed cost models were also developed for the threshold and baseline payload. Further development is necessary to mature the final design architectures and approaches.

1.0 SCIENTIFIC OBJECTIVES

1.1 CORAL Science Overview

Centaurs are dynamically evolved, small, icy bodies from the Kuiper Belt, but they currently reside between Jupiter and Neptune. They are more accessible than Kuiper Belt Objects and they have experienced less thermal processing than comets. This makes Centaurs a high priority science target for *in situ* compositional analysis of primordial icy planetesimals, the building blocks of planets.

The goal of the *CORAL* mission is to measure the chemical and physical properties of these dynamically evolved but compositionally primitive small bodies to constrain the composition and evolution of icy planetesimals. This mission has important implications for understanding the evolution of the Kuiper Belt, radial mixing in the protoplanetary disk, and delivery of volatiles to the inner solar system. Since Centaurs are transitional objects between Kuiper Belt Objects (KBOs) and Jupiter Family Comets (JFCs), this mission has broad implications for the formation history of three populations of small bodies (Figure 1-1). This mission concept study investigated the feasibility of global characterization of a Centaur and landed *in situ* compositional analyses at the surface within a New Frontiers class mission.

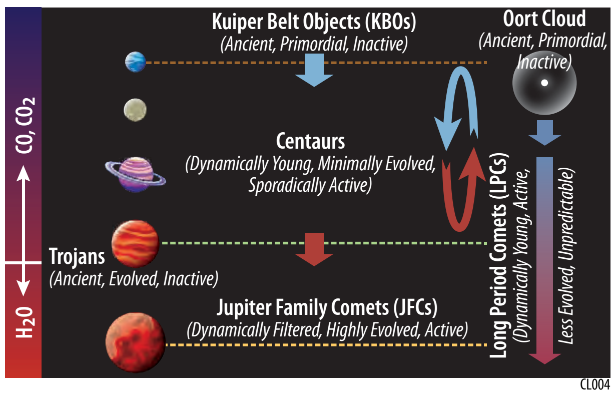

Figure 1-1: From Harris (2020). This illustrates the classification of ice-rich small bodies in the solar system. Centaurs are ice-rich bodies from the Kuiper Belt, but currently reside in orbits between Jupiter and Neptune. CORAL aims to carry out in situ geochemical analyses of a Centaur, which are more accessible samples of Kuiper Belt material, but they have not been processed by close encounters with the Sun as comets have.

Centaurs: Unlocking the icy planetesimal record of early Solar System history

Centaurs provide an opportunity to conduct a comprehensive study of the geochemical and physical properties of primordial ice-rich planetesimals. Planetesimals are the building blocks of planets; they are small planetary bodies that formed immediately following the solar system's formation. They formed through collisions and sticking of dust particles and pebbles. Their sizes ranged from a few meters to hundreds of kilometers. Planetesimals formed as the protoplanetary disk cooled. They were icy or rocky depending on where in the disk they accreted (Figure 1-2).

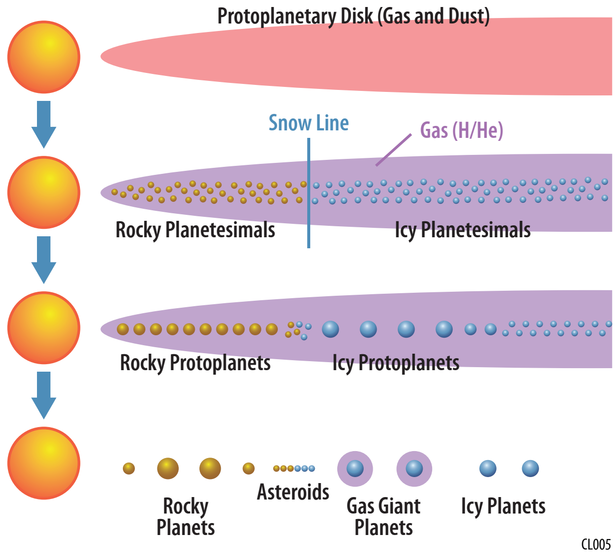

Figure 1-2: From Genda (2016). Centaurs provide a record of the composition of icy planetesimals that formed beyond Neptune. This provides a very broad overview of the evolution of the protoplanetary disk, including the formation of rocky and icy planetesimals and the subsequent formation of planets in the solar system. Ice-rich planetesimals formed beyond the snow line, the orbital distance where temperatures in the disk were sufficiently cold to condense water ice from nebular gas.

Meteorites provide an incredible record of rocky planetesimals that we can characterize in amazing detail in laboratories. However, we do not have a similar record of icy planetesimals, which trace the composition of nebular volatiles such as H_2O, CO_2, CO and NH_3. These molecules are not readily incorporated into rock-forming minerals. Although, the meteorite record of ice-rich planetesimals is limited, we can infer that ice-rich planetesimals formed with at least 50% water ice (based on the abundance and composition of hydrated minerals, carbonates and oxides in carbonaceous chondrites (e.g., Clayton and Mayeda, 1999). The "parent" bodies of these hydrated meteorites are thought to have formed in the outer Solar System, but were then mixed inward due to dynamical processes involving the migration of the giant planets (DeMeo and Carry, 2014). Such mixing of planetesimals to the inner solar system is consistent with the present-day compositional heterogeneity of material in the Asteroid Belt (Figure 1-3). A mission that allows for detailed geochemical characterization of ice-rich planetesimals is necessary for understanding the initial composition and distribution of nebular components (dust, ice, and gas) and subsequent mixing of accreted material following giant planet migration.

Centaurs are an ideal target for *in situ* geochemical analyses of ice-rich planetesimals. They are more accessible samples of Kuiper Belt material, but they have not been processed by close encounters with the Sun as comets have. Prior mission concept studies and proposals have focused on Centaur orbiters and flybys for Discovery class missions. Adding landed geochemical analyses is much more complicated and costlier, thus requiring a New Frontiers class mission. Experience and high-heritage instrumentation from Mars rovers and Rosetta make this next decade a favorable moment for *in situ* geochemical analyses of a Centaur. CORAL could unlock a treasure trove of new discoveries about the composition and evolution of multiple reservoirs of small primordial icy bodies, including KBOs, JFCs and ice-rich asteroids.

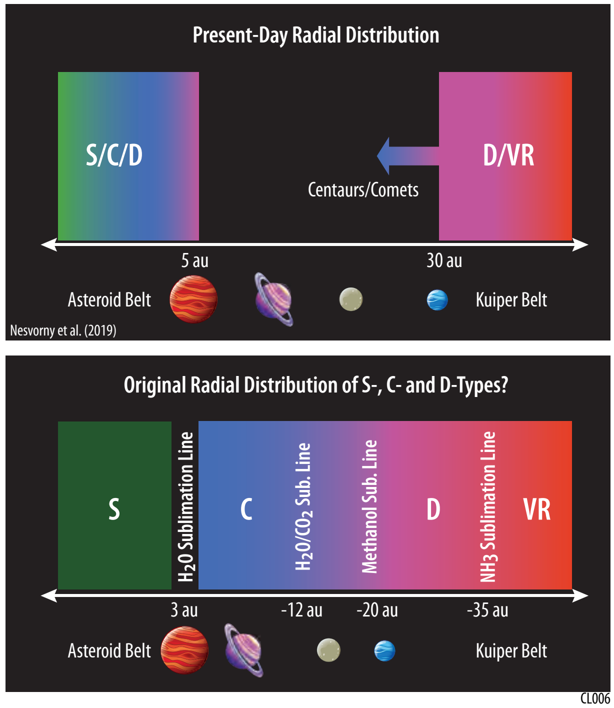

Figure 1-3: From Nesvorny (2021). Centaurs allow access to populations of objects that are distinct from those in the Asteroid Belt. This shows the spectrographic designations (S, C, D, VR) for various small bodies. The top panel shows the present-day radial distribution of small bodies and the bottom panel shows what the original distribution may have been based on dynamical models, including the possible sublimation lines for various ices that could have accreted to form ice-rich planetesimals (Levison et al. 2009).

Coral Science Goals

The four overarching science goals of CORAL are to:

1. **Understand early solar system compositional reservoirs.** The protoplanetary disk was composed of gas and dust. The composition of this material varied with time and space. Meteorites, micrometeorites, and sample-return missions provide a wealth of knowledge of rocky material and planetesimals in the solar nebula. Reservoirs of gas and ice are not well-represented in our sample collections because they are readily lost from rocks. Landed elemental, isotopic, and organic analyses of a Centaur will enable constraints on the composition of nebular gas and icy planetesimals, important for developing a comprehensive understanding of the composition and initial conditions of the protoplanetary disk.
2. **Understand the accretion and dynamical evolution of primordial icy planetesimals.** Centaurs are intermediary objects in the dynamical evolution of KBOs to JFCs (Figure 1-1). CORAL is particularly timely as it is directly relevant to the recent observations of Arrokoth, a KBO, by New Horizons and 67P/ Churyumov-Gerasimenko, a JFC, by Rosetta. Detailed compositional analyses of a Centaur through orbital and landed analyses would enable correlating spectral signatures with in situ data for better interpretation and cross-comparison of remote sensing of other bodies.
3. **Determine the geological and evolutionary processes that have influenced icy planetesimals.** The shape, topography, geological landforms, and landing site characteristics will provide key insights into the evolutionary history of this population of objects. Some Centaur have ring systems and others have ongoing activity, making them additionally compelling targets to investigate the evolutionary processes that influence ice-rich planetesimals.

4. **Investigate the biologic potential of icy planetesimals and potential brine reservoirs.** Centaurs are rich in organics and volatile material that were important for the development of life on Earth. Since the most primitive meteorites contain hydrated silicates and carbonates, minerals commonly associated with fluids, it is possible that these small icy bodies could potentially contain brine reservoirs (Rivkin et al., 2002).

1.2 CORAL Science Traceability

The CORAL Science Traceability Matrix (STM) (Table 1-1) maps the science objectives and measurements to the overarching science goals listed above and to the mission requirements.

Driving mission requirements

The driving mission requirements for CORAL include:

1. Have a lander that allows for in situ compositional measurements of at least one location on the surface, but preferably two locations;
2. Identify suitable landing locations using high-resolution imaging and/or lidar measurements; and
3. Provide sufficient observing time in orbit to fully characterize the environment and conduct operations to accommodate possible comet-like activity and/or rings.

Target selection

There are over 550 known Centaurs, but we limited our search to those with well-known orbits, perihelion distances within 10 AU, inclinations less than 60 degrees, cruise duration between 5 and 13 years, launch dates between 2036 and 2040, and a delivered mass above 2000 kg, which is necessary to accommodate landed geochemical analyses. This left 4 viable targets to choose from (Table 2-3) and we chose to focus on 2015 BQ311. The current mission design consists of a 9-year cruise with 4 years to characterize the target (see Section 3.2 for details). It is important to note that any of these 7 targets could potentially work for this study and that other interesting targets (not listed here) could be possible depending on the launch dates and mission design.

Currently, 2015 BQ311 is not well-characterized. From the Jet Propulsion Laboratory small bodies database, we know that it has an absolute magnitude of 12.4[1]. This target is blue in color (Jewitt, 2015). Assuming an albedo between 0.04 and 0.2[2], the diameter of 2015 BQ311 is estimated to be 10-23 km, which is larger than comets that have been visited previously but comparable to the size of Arrokoth, a cold classical KBO recently visited by New Horizons. Otherwise, 2015 BQ311 is a mysterious object. The rotation rate and the shape of this target are unknown. Whether or not 2015 BQ311 has activity or rings is also unknown. 2015 BQ311 and the other viable targets identified in the concept study would be characterized remotely in more detail prior to the mission launch in January 2040. Also, with a 9-year cruise, there should be at least 4 years for characterization of the target and for landed analyses.

The phasing between Jupiter and 2015 BQ311 is such that earlier launch dates in the 2030 to 2035 range for shorter cruise flight times (e.g. 9 years) do not perform as well as the baseline 2036 to 2040 period. However, 2015 BQ311 is still a viable centaur target with trajectories launching in 2033, 2034, or 2035 capable of delivering at least 3000 kg after centaur rendezvous. Unlike the 2036 to 2040 evaluation period, flight times near the 13-year maximum are necessary and would require additional power profile evaluation.

Orbital and landed science

The mission is divided into two main science phases, an orbital phase and a landed phase. The main objectives for the orbital phase are to determine 1) the global mineralogical composition of the target, 2) the impact his-

[1]https://ssd.jpl.nasa.gov/sbdb.cgi?sstr=2015+BQ311

[2]http://www.johnstonsarchive.net/astro/tnodiam.html

Table 1-1: Centaur Orbiter & Lander (CORAL) Mission Concept Study: Science Traceability Matrix

Science Priority	Science Objective	Measurement Objectives	Mission Phase	Measurements	Instruments	Mission Requirements	Comments/Analog Instruments
CORAL Science Goal #1: Understand early solar system compositional reservoirs							
1	Determine the isotopic composition of icy planetesimals, including their interstellar component	isotopic composition	surface	D/H, C, H, O, N isotopic composition at the surface	Mass Spectrometer	Sample via PlanetVac, heat sample to measure composition of evolved gas	Rosetta-style Ptolemy GCMS
1	Determine the large-scale mineralogical make-up of icy planetesimals	global mineralogical composition	orbit	spectral mapping of surface in UV and IR	UV imager/spectrometer; IR imaging spectrometer	orbital operations to enable global mapping	UV imaging spectrometer, such as Rosetta's ALICE; IR imager such as LEISA (New Horizons)
1	Determine the grain scale composition and mineralogy of icy planetesimals, including if high-temperature inner solar system condensates are present	mineralogy at grain scale	surface	mineralogy at the surface	XRF; UV and Raman spectrometer	requires sampling system to bring surface material to instrument	XRF: PIXL-like system (Mars 2020), Combined UV Raman system: SHERLOC (Mars 2020)
3	Determine the interior volatile composition	composition of volatiles released by activty	orbit	abundances and isotopic ratios of gases	Mass spectrometer	use sniffing mode of GCMS	
CORAL Science Goal #2: Understand the accretion and dynamical evolution of primordial icy planetesimals							
1	Determine the impact history and relative ages	Crater counting	orbit	Collect high resolution images suitable for crater counting	High resolution cameras, lidar	global imaging at ≤50 m resolution	Such as PolyCam, MapCam, same as OSIRIS-REx
1	Determine the physical characteristics of the body	Measure shape, determine the mass/density/porosity, determine the rotation state/period	orbit	gravity field, topography	Radio science, high-res cameras, lidar	orbital operations to constrain mass	OSIRIS-REx and other examples
2	Determine the internal mass distrubution	state of interior mass	orbit	low degree gravity field	Radio science	operations at low altitudes to enable mass distribution measurements	standard S/C telecom package
3	Determine the magnetism present during formation and accretion	Determine the remnant magnetization of the Centaur	orbit & surface	Measure the magnetic properties from orbit and near the surface	Magnetometer	magnetometer measurements while in orbit and while landed	Rosetta-like system
CORAL Science Goal #3: Determine the geological and evolutionary processes that have influenced icy planetesimals							
1	Determine landforms and any evidence for changes over the mission	topography, imaging, spectral mapping	orbit	global images of the body	high resolution cameras, lidar	global imaging at ≤50 m resolution	Such as PolyCam, MapCam, same as OSIRIS-REx
1	Determine the icy regolith characteristics, such as grain size	high-resolutio imaging of the surface	surface	surface images	landed camera suite	cameras to cover the landing site and local region sampled for chemical analysis	Dragonfly examples
2	Determine how the surface is affected by space weathering	Compare subsurface (below regolith) to the surface and iR mapping	orbit, surface	Infrared mapping, repeated landed compositional measurements on disturbed surface	IR imaging spectrometer; landed compositional measurements	disrupt surface while landed	IR imager such as LEISA (New Horizons)
3	Determine the source and cause of activity (if present)	Monitor outgassing activity, constrain driver of activity, constrain composition and size of dust	orbit	Measure volatile and dust composition	Mass spectrometer	Orbital opeartion of >3 months to enable long baseline for monitoring	Rosetta-style system i.e., COSIMA, DFMS ROSINA, GIADA, MIDAS, plasma sensors
3	Characterize ring systems and/or binaries (if present), and constrain the formation of ring systems.	composition of the ring(s), particle size distribution.	orbit	mapping of the ring structure and spectral characteristics	UV imager/ spectrometer; IR imaging spectrometer; high resolution cameras	visit Centaur with rings or binary system	no new payload beyond already used for global mapping of the main body
CORAL Science Goal #4: Investigate the biologic potential of icy planetesimals and potential brine reservoirs							
1	Determine the thermal history of Centaurs by looking for alteration minerals	compositions and distribution of minerals	orbit, surface	global mineralogy and surface mineralogy	UV imager/ spectrometer; IR imaging spectrometer; XRF; Raman spectrometer	see Goal 1 mineralogy	See Goal 1 mineralogy
1	Determine the composition, form, and distribition of organic material	Constrain the composition of organic material	surface	Measure the composition of organic material at the surface	XRF; UV/Raman spectrometer	requires sampling system to bring surface material to instrument	

tory and relative ages of the surface, 3) the physical characteristics of the target, and 4) the landforms and any evidence for changes over the course of the mission. This work will be carried out using high-heritage infrared and ultraviolet spectrometers, high-resolution cameras (WAC and NAC) and a magnetometer (see schematic of the spacecraft in Figure 2-1).

The objectives for the landed phase of CORAL are to determine the isotopic composition, mineralogy, thermal history and the organic composition of material at the surface of the target. Samples will be collected from the surface via PlanetVac on the landing pads. To analyze samples at the surface, we will use a combination of gas-chromatograph mass spectrometer (GCMS), X-ray fluorescence (XRF) and a combined Raman and UV spectrometer system. All of these are high-heritage systems have been flown or will be flown soon (see Section 3.1 Instrument Payload Details). The GCMS will be used to analyze the abundance and isotopic composition of volatiles and organics. The XRF will provide constraints on the elemental abundances for the samples. The Raman and UV system will be used to constrain mineralogy. Additionally, a panoramic camera will image the landing site and sampling locations and the magnetometer will acquire landed measurements. The STM (Table 1-1) provides more detail on the science goals and instruments.

Table 1-2: Expected measurement precision[a] for Ptolemy[b]-like gas chromatograph mass spectrometer.

Species	Ratio	Precision
H_2	δD	± 100‰
N_2	δ15N	± 10‰
CO	δ13C	± 1‰
	δ18O	± 1‰
	δ17O	± 22‰
CO_2	δ13C	± 1‰
	δ18O	± 1‰
	δ17O	± 22‰
O_2	δ18O	± 1‰
	δ17O	± 22‰

[a]S. Barber, 2021, personal communication.

[b]Ptolemy is the GCMS on Philae, the lander for the Rosetta mission to 67P/ Churyumov-Gerasimenko.

Isotope analyses of pristine ice at the surface of a centaur could provide isotopic constraints on ice and gas in the solar nebula. Strong constraints on the composition of nebular gas and ice are lacking because these components are readily lost in rocky material. The oxygen isotopic composition of different planetary reservoirs determined from geochemical analyses of meteorites and other planetary material span a range from -100 to +600‰ (Hashizume, 2015), which is significantly larger than the uncertainties expected for landed GCMS analyses (Table 1-2). This is also the case for hydrogen and nitrogen isotopes (Füri and Marty, 2015). The H and O isotopic composition of a body is thought to be linked to the region in the disk where it formed. Therefore, these analyses for a Centaur could put strong constraints on the composition of the KBO region of the nebula. The isotopic composition is also very sensitive to mixing between reservoirs so these analyses would provide insight into radial mixing of material in the nebula. Additionally, models indicate that icy bodies similar in diameter to 2015 BQ311 should not have experienced any heating from the decay of radioactive isotopes if they formed 3 Myr after solar system formation (Castillo-Rogez and Young, 2017). This is consistent with ages inferred from analysis carbonaceous chondrites (Jilly-Rehak et al., 2014), indicating that these types of Centaurs should have a pristine compositional record, making them ideal for *in situ* isotopic and geochemical analyses.

Possible descopes:

The grand total for Phase A-D, including the full instrument payload is $1.29B (FY25). These costs are likely over-estimated due to 50% reserves for high-heritage instruments. The panoramic camera is listed under flight systems to support navigation, so it is not included in the instrument payload. To bring costs down, we considered a payload without the magnetometer (both landed and orbital phase), the XRF (landed phase) and the UV spectrometer (orbital phase). With this descope, the mission costs come down to $1.16B, more in line with New Frontiers cost cap. This descope does not affect the top priority science goals of CORAL (see STM). The magnetometer is low-cost and does not require large data volume, but it is not necessary for the top science goals. The XRF is useful for elemental analyses at the surface, but we can meet our science goal of understanding the bulk composition with the Raman/UV system alone. Similarly, we can descope the UV spectrometer and use only the infrared spectrometer to understand the global composition. See Section 5.0 for details of the mission cost analysis.

2.0 HIGH-LEVEL MISSION CONCEPT

2.1 Concept Overview

The goal of the CORAL mission is to measure the chemical and physical properties of these dynamically evolved but compositionally primitive small bodies to constrain the composition and evolution of icy planetesimals. This mission concept demonstrates the feasibility of globally characterizing and obtaining landed *in situ* compositional measurements of a Centaur within a New Frontiers class mission.

The CORAL Spacecraft is shown in Figure 2-1. The Spacecraft is launched on January 23, 2040 by a Falcon Heavy Expendable with a 5m fairing from Cape Canaveral, Florida. The Spacecraft will spend nine-years in interplanetary transfer and rendezvous with Centaur BQ311 on January 20, 2049 (Figure 2-2).

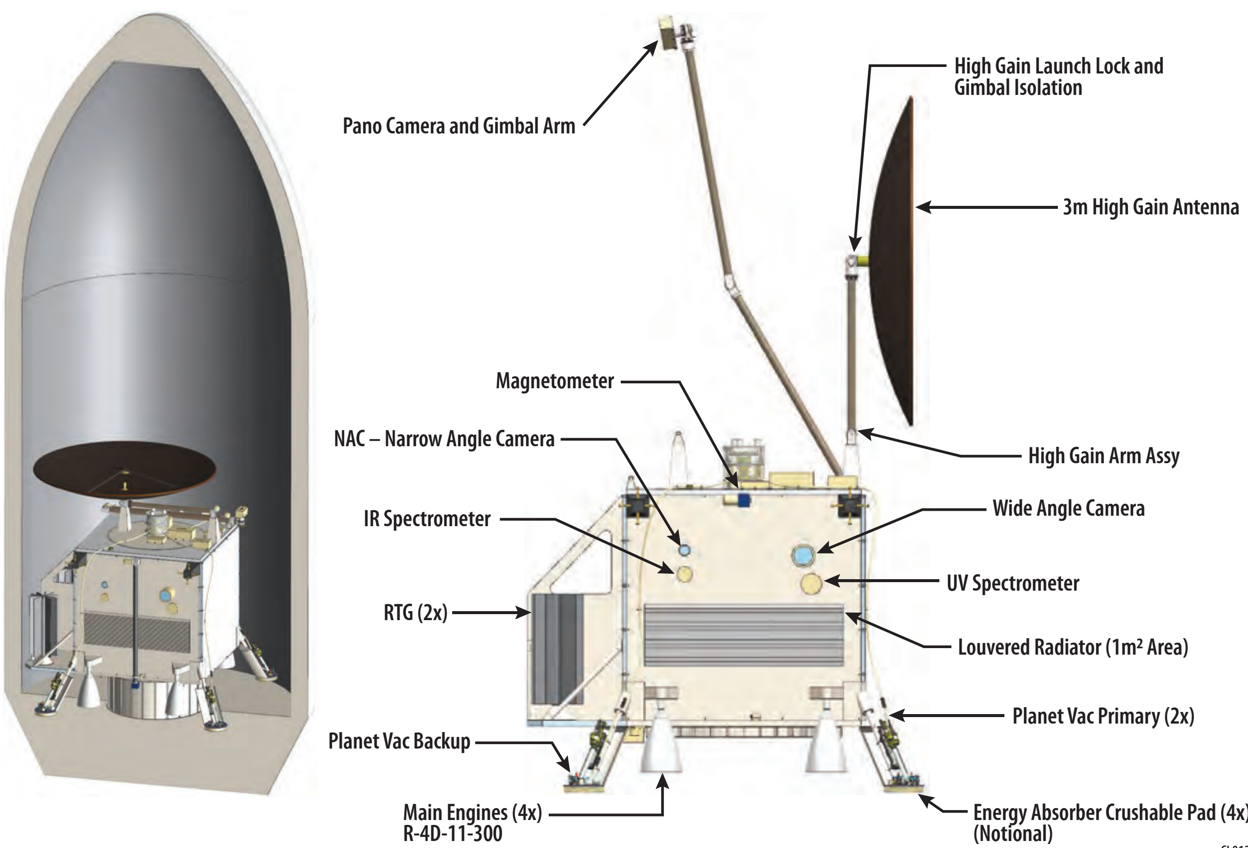

Figure 2-1: CORAL spacecraft

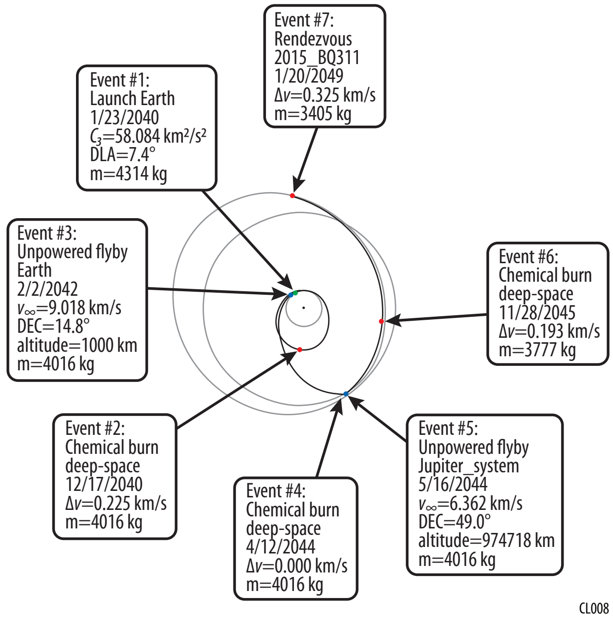

Figure 2-2: CORAL trajectory

After rendezvousing with BQ311 the spacecraft will begin proximity operations (Section 3.2) that last ~4 years. The length of the proximity operations is driven by the assumption that the mapping subphases will require the same volume of imagery as was used by OSIRIS-Rex. Since proximity operations around BQ311 occur at distances from Earth of >6.7 AU the data rate is limited and the resulting duration required to return all the images is long (Section 3.3).

After proximity operations have completed and the landing site has been chosen the spacecraft will land and conduct landed operations for 8 weeks. Upon completion of landed science operations, the option exists for the lander to takeoff and land at a second site.

2.2 Concept Maturity Level

This study was conducted with a goal of Concept Maturity Level (CML) 5, however, certain aspects of the design concept could be classified as a CML 4 (see Table 2-1 for CML definitions). It presents an implementation concept at the subsystem level, as well as science traceability, mission requirements traceability, key technologies, heritage, risks and mitigations. Detailed cost models were developed. Further development is necessary to mature the final design architectures and approaches.

Table 2-1 Concept Maturity Level Definitions

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

2.3 Technology Maturity

CORAL uses high heritage instruments and spacecraft hardware that is at TRL >6. The CORAL study team did not identify any areas that technology development is needed. The study team recommends that during the early CORAL mission formulation phases the engineering team validate the instrument design and ensure the availability of parts since some of them may be unavailable or discontinued (obsolete) as captured in one of the risks (Table 3-9).

2.4 Key Trades

Centaurs pose challenging mission problems. They are characterized by large orbits and often have high eccentricities and inclinations, necessitating high ΔV trajectories. While propellant-efficient electric propulsion can enable high ΔV missions, the thrusting required for rendezvous at a centaur can drive solar array sizes that are prohibitively large because of the large solar distance. Similarly, radioisotope electric propulsion options are typically limited to low power because of the cost of RTGs and are often unable to provide adequate thrust for a high-mass spacecraft such as lander to rendezvous with a centaur.

The core mission trade for the study is to identify centaurs and propulsion options that enable a lander mission given the inherent high propellant cost of rendezvous and launch vehicle limitations. Top Centaurs of interest are shown in Table 2-2 particular interest, is the viability of landing on Chiron, which was previously targeted in a 2010 Decadal Study.

Table 2-2: Top-Interest Centaurs from Science Team

Name	Orbit Condition Code	JPL SBDB Designation	Interest
Chiron	0	Centaur	Activity, rings
Chariklo	1	Centaur	Rings
Bienor	0	Centaur	Rings?
Ceto	1	TransNeptunian Object	Binary
Typhon	1	TransNeptunian Object	Binary
Echeclus	0	Centaur	Activity
29P/Schwassmann-Wachmann	0	Jupiter-family Comet	Activity
P/2019 LD2	3	Jupiter-family Comet	Activity
2014 OG392	3	Chiron-type Comet	Activity
39P/Oterma	0	Chiron-type Comet	Activity
165P/Linear	4	Chiron-type Comet	Activity
166P/2001 T4	2	Chiron-type Comet	Activity
167P/2004 PY42	3	Chiron-type Comet	Activity
C/2001 M10	3	Jupiter-family Comet	Activity
P/2004 A1	3	Jupiter-family Comet	Activity
2003 QD112	4	Centaur	Activity
P/2005 T3	6	Chiron-type Comet	Activity
P/2005 S2	5	Chiron-type Comet	Activity
2006 SX368	2	Centaur	Activity

A broad mission design trade (Section 3.2) was conducted in order to both determine a design reference mission for detailed evaluation as well as to characterize the target and mission architecture design space. Over 550 centaur targets are evaluated based on 'centaur' or 'Chiron-type comet' designations in the JPL Small-Body

Database and science team input. The top perforin Centaurs that meet requirements for launch date and interplanetary cruise duration are shown in Table 2-3.

Table 2-3: Top Performing Centaur Targets (<13-year interplanetary cruise, launch between 2036 and 2040)

Centaur	Preliminary Delivered Mass [kg]					aphelion distance [AU]	Science Rank	Science Notes
	Chemical	REP +chem	REP only	SEP +chem	Best			
2015 BQ311	8905	3404	700	4200	8905	9.19	3	colors (blue)
2004 RW141	4483	1707	833	6372	6372	10.41		small
2008 SJ236	3654	978	396	3990	3990	15.67	2	colors (red), relatively large, ~high albedo
2020 OD8	3446	1836	1276	2767	3446	10.70		small
2016 EX	3140	1992	1628	3963	3963	11.10	4	no colors
2005 TS100	3124	1655	1133	3191	3191	6.50		
2017 UV43	2617	1660	1530	3588	3588	8.49		
2010 NK83	2048	520	2026	1758	2048	9.33		
1998 SG35	2008	1088	965	1175	2008	10.91	1	most information, "large" object
39P/Oterma	1981	1192	527	596	1981	8.97		
Chiron	1832	NF	NF	407	1832	18.87		
2019 LD2	1372	560	NF	2847	2847	6.01		
2010 WZ71	NF	751	1861	2402	2402	8.73		
2015 DB198	1517	1558	1446	1991	1991	11.72		
2000 VU2	866	1648	1123	1736	1736	10.68		

In addition to target centaur, a variety of mission architecture options are considered using different propulsion options including traditional chemical-only bi-propellant propulsion with the option of propulsion stages, RTG electric propulsion (REP), solar electric propulsion (SEP), as well hybrid options with chemical propulsion for the arrival maneuver and electric propulsion for the rest interplanetary transfer. For REP missions, one to three RTGs were evaluated, with a preference for one or two units given the high cost of RTGs. The 16-GPHS STEM-RTG, the only RTG option considered as it is the highest performing allowed in the study rules, provides 400 W (BOL) per RTG to be divided for EP and non-EP power, with the remaining power available for spacecraft electronics and payload. For SEP missions, solar arrays providing between 10 and 40 kW at one AU in 10 kW increments were traded. Both NASA's Evolutionary Xenon Thruster (NEXT) and XIPS-25 thrusters were considered with two to four active thrusters. NEXT thrusters are notably efficient at high powers, but have a high minimum input power, making the low input power of XIPS-25 potentially advantageous for some architectures. The preferred baseline was briefly discussed above in the Mission Overview (Section 2.1) and is further discussed in the remainder of this document.

Additional trades conducted included whether to include both Ka- and X-Band hardware. It was determined that the benefit of an only Ka-Band system outweighed any redundancy or flexibility having the X-Band components offered. The power constraints on the system dictated that only one communication system could be active at a time so the excess complexity of two systems was unnecessary and with the higher gain and higher data rates Ka-Band only was the most logical choice.

Another key trade was determining the need for the gimbal on the HGA. The gimbal provides the necessary flexibility for the unknown landing site parameters and therefore outweighed the concern of higher losses in the communications path from the added length of waveguide and cables this choice requires.

The HGA size was also up for trade as there are multiple heritage examples of differing parabolic dish sizes. Dishes with 2- and 3-meter diameters have been successfully used on other deep space missions so those were the main choices with flight heritage, but larger antennas were also considered as they provide significant gain increases. It was decided that keeping within heritage examples provided a known quantity that outweighed the benefit of a larger, more costly dish and allowed the spacecraft to fit comfortably in the launch fairing without introducing a complex, high cost, deployable antenna. The decision to go with the 3-meter antenna was obvious as it more than doubles the gain of the system without a large increase in cost or complexity as compared with a 2-meter dish.

3.0 TECHNICAL OVERVIEW

3.1 Instrument Payload Description

The CORAL payload is comprised of: 1) Science instruments, 2) Navigation hardware, a panoramic camera and robotic arm, and 3) a Sampling acquisition and handling system as shown in Table 3-1. The science heritage instruments were selected to be a set of representative instruments that meet the high priority science requirements of the STM (Table 1-1). It is based on previously flown instruments that would allow for implementation in the mission design without a need for technology development. Figures 3-1 and 3-2 show how the payload subsystems are distributed on the lander.

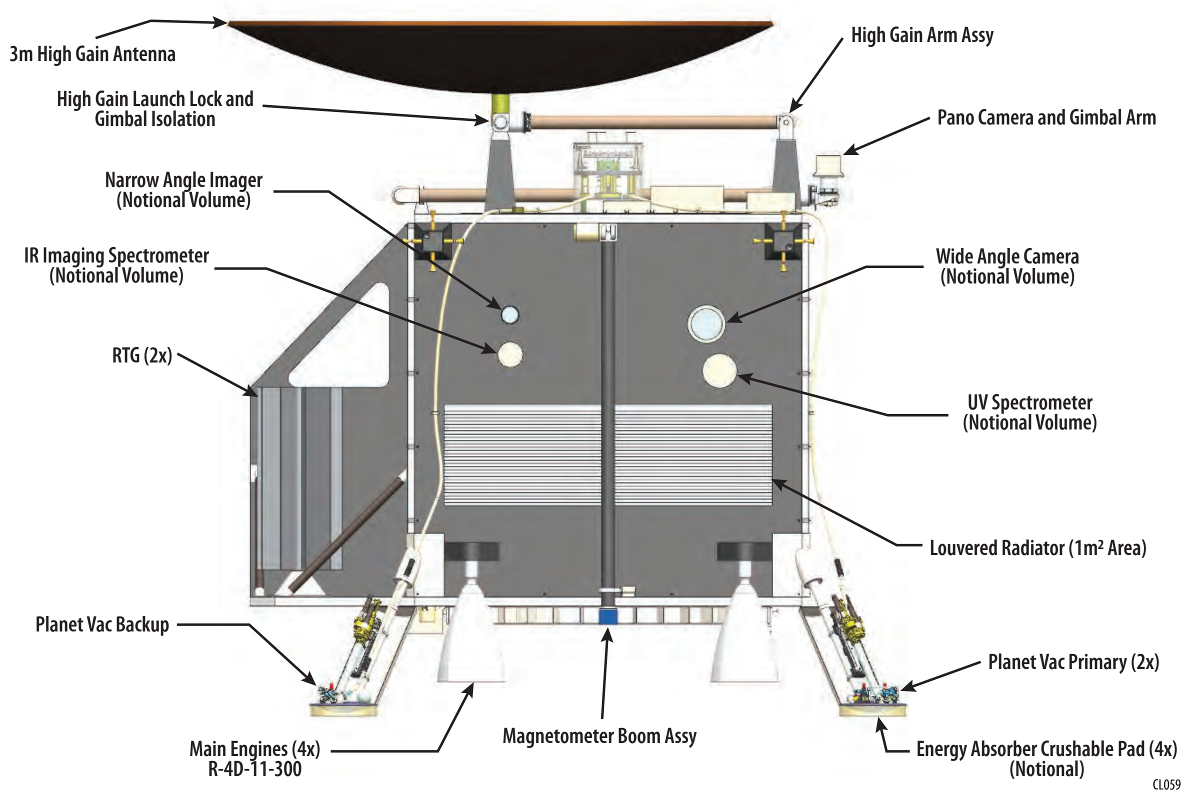

Figure 3-1 CORAL Lander

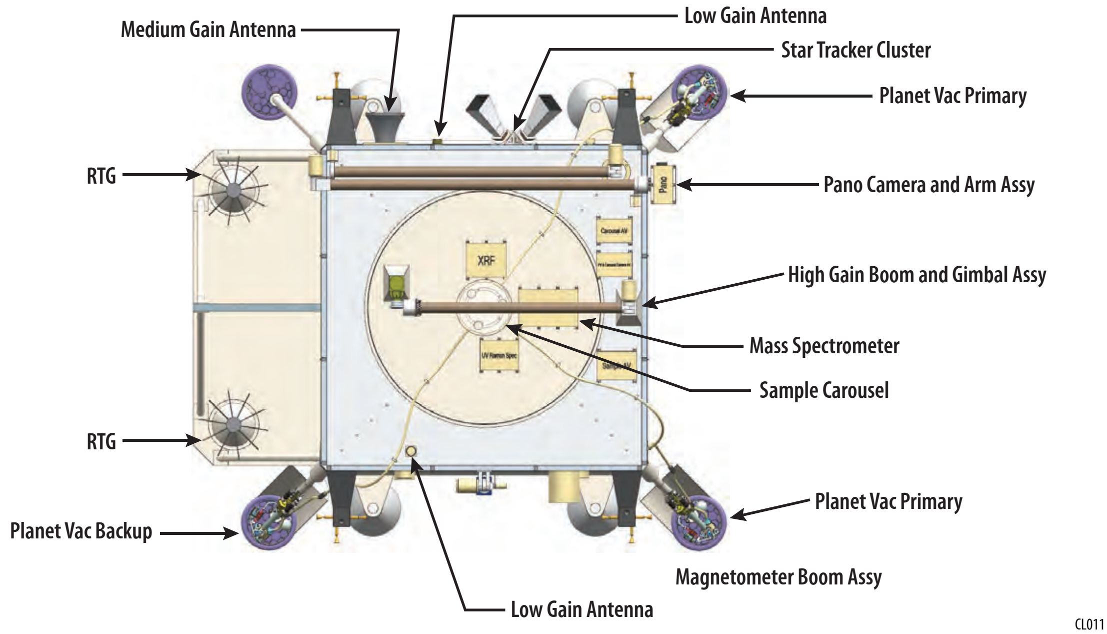

Figure 3-2: CORAL lander top view (HGA not shown)

Table 3-1 CORAL Payload

Science Instruments
GCMS (Gas-chromatograph mass spectrometer)
NAC (Narrow Angle Camera)
UV and Raman Spectrometer (Ultraviolet Raman Spectrometer)
UV Spectrometer (Ultraviolet Spectrometer)
IR Spectrometer (Visible and Infrared Spectrometer)
WAC (Wide Angle Camera)
XRF (X-Ray fluorescence)
Magnetometer
Navigation Hardware
Laser Range finder
LiDAR
Optical/IR Cameras
Sample Acquisition and Handling System
Panoramic Camera and Arm
Drill
SAS (Sample Acquisition System)
Carousel

During proximity operations, the infrared spectrometer (OVIRS/OSIRIS REx), ultraviolet spectrometer (Alice/Rosetta) and high-resolution imagers (WAC and NAC/LRO) are used for characterizing the environment and target. The imagers and imaging spectrometers are mounted on the same side of spacecraft and will face nadir during most of the orbital phase.

The landed phase has the Gas-chromatograph mass spectrometer which is modeled on flown mass spectrometers including Ptolemy on Rosetta/Philae, X-ray fluorescence instrument (PIXL/Mars 2020 Perseverance), and a combined Raman and UV spectrometer system (SHERLOC/Mars 2020 Perseverance) to perform in-situ elemental, isotopic, and organic analyses of the samples on the Centaur surface. The in-situ instruments are mounted on the top deck encircling the sample carousel (see Figure 3-2).

The standard fluxgate magnetometer (Magnetometer/MAVEN) is mounted on a deployable boom to measure the magnetic properties from orbit and near the surface. The spacecraft uses a deployable panoramic camera for contextual imaging of the landing site and local region sampled for chemical analysis.

A 3m mechanical arm with two 1.5m sections and gimbals at the base, elbow and wrist provides maximum flexibility for the Panorama Camera to obtain context imagery around the landed spacecraft.

A pneumatic sample acquisition system (SAS) which is similar to PlanetVac has a drill and is responsible for collecting samples from surface and sub surface depths and distributing samples via a carousel to the GCMS, XRF and UV and Raman combined instruments. Three SAS will be implemented in this design, each attached near the lander footpads. The SAS located closest to the NTRGs will be used as a backup system.

The drill is 10 cm in length, and 1.25 cm in diameter. This should provide enough material volume for the intended number of analysis at the various depths with sufficient margin. This is the same class size of drill used on MSL which is a rotary percussive drill of 6 cm in length. Since the depth to which we can drill depends on the unknown surface topography and the placement of the SAS cone on the surface there is no guarantee the drill depth corresponds to sample at that depth, the sample collected could be from shallower depths of the same hole. Based on experience, it has been proven that the sample collected is no deeper than the extension of the drill depth. The operations concept is to drill in incremental depths of of 2 or 3 cm and then collect that sample. The drilling operation will be conducted 3-4 times before the drill is fully extend. Drill extension information will be accurate to mm resolution.

Gas Chromatograph Mass Spectrometer:

This mission carries a Gas Chromatograph Isotope Ratio Mass Spectrometer (GC-IR-MS) modeled on elements with high heritage.

The instrument is responsive to the applicable needs of the Science Traceability Matrix (Table 1-1), such as isotope ratios. It also has capability for objectives beyond those levied by the STM, such as other organics. Capabilities beyond those levied by the STM were not prioritized for design or analyzed for capability, so they will only be mentioned in rare cases.

The sample can enter the instrument via the paths indicated in the block diagram (Figure 3-3). Solid sample enters via loading into the oven. Freely volatilizing sample directly from the surface of the centaur can also enter via a gas inlet tube.

Minimum m/z capability is suitable to measure D/H ratios in water and ammonia, and maximum capability can go well beyond carbon dioxide. The isotopic and organic analysis can be affected by trace gases from chemical propulsion. However, the study identified an approach in the sample acquisition technique that can help understand the chemical fuel impacts on signal (See Appendix section 3.5 for more details). Again, capability for measurements beyond satisfying the STM were not prioritized for design or analyzed for capability.

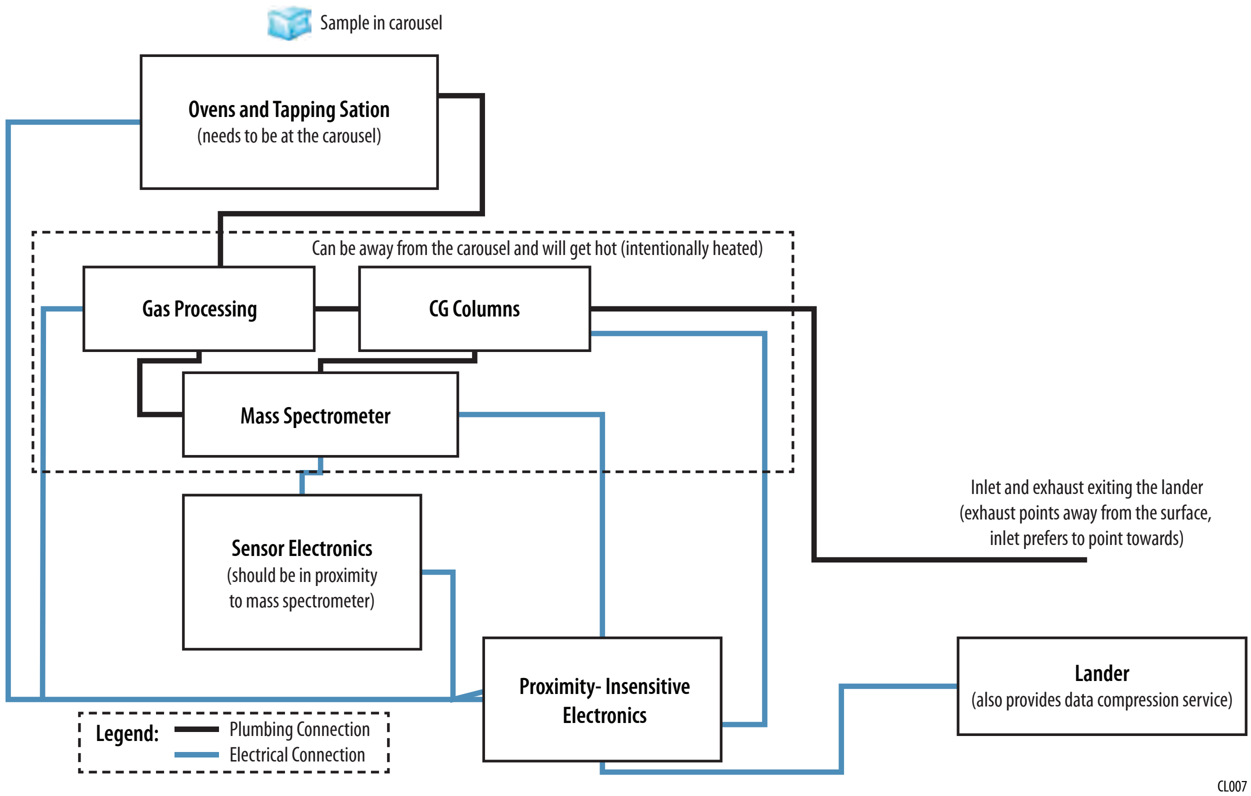

Figure 3-3. GCMS Block Diagram

XRF:

The X-Ray Fluorescence spectrometer is based on the Planetary Instrument for X-ray Lithochemistry (PIXL) flown on the NASA Mars 2020 Perseverance rover. This instrument measures the fine-scale chemical makeup of rocks using an X-ray spectrometer and camera. It is a microfocus instrument that has a high sensitivity for detection of trace chemical elements at sub-millimeter spatial resolution levels. The PIXL design incorporates a high-resolution camera for 2D fast mapping of rock samples. Its rapid spectral acquisition system can measure major and minor elements within seconds. This operationally efficient XRF instrument meets the required science measurement of mineralogy at the surface.

Combined Raman and UV Spectrometer:

The combined Raman and Ultraviolet Spectrometer is based on the Scanning Habitable Environments with Raman & Luminescence for Organics & Chemicals (SHERLOC) instrument. It has also flown on NASA Mars 2020 Perseverance rover which combines Raman and Deep UV-induced native fluorescence for fine scale detection. This 2D spectral mapper use: cameras, spectrometers and a UV laser to detect and classify organics and minerals present in rocks and help understand the environment in which the rock sample formed. SHERLOC's mapping mode is used for nondestructive, sub-picogram sensitive organic detection. SHERLOC spectra can be complementary with measurements made by other payload elements, including elemental abundances measured by PIXL.

Infrared spectrometer:

The IR spectrometer is modeled after the NASA OSIRIS-REx visible and infrared spectrometer (OVIRS). This is a point spectrometer covering wavelengths from 0.4 µm to 4.3 µm. It will provide spectral mapping of the surface composition and global context for the sampling site.

UV spectrometer:

The heritage Alice ultraviolet imaging spectrometer on board Rosetta was used as a model for this instrument. It focuses on spectral features in the far-ultraviolet wavelength range from 70 nm to 205 nm. It will be used to characterize the target and study the surface properties.

Magnetometer:

The NASA/GSFC standard fluxgate magnetometer has direct heritage with the MAVEN mission. It will provide continuous, high resolution coverage of the magnetic field about the Centaur.

Cameras:

The cameras were based on the NASA Lunar Reconnaissance Orbiter Camera (LROC) Wide Angle Camera (WAC) and the Narrow Angle Cameras (NACs) heritage imagers. There are two positions used for collecting high resolution images suitable for crater counting and global imaging of the body. The WAC is a 7-color push-frame camera (100 m/pixel and 400 m/pixel visible and UV, respectively), while the two NACs are monochrome narrow-angle line-scan imagers (0.5 m/pixel). The NAC includes a sequence and compression system for data processing prior to data transfer to the spacecraft command and data handling.

TABLE 3-2: CORAL Instrument Payload Characteristics

Item	GCMS	XRF	NAC	WAC	Spectrometers			Magneto-meter
					UV and Raman	Visible and Infrared	Ultraviolet	
Type of instrument	Mass Spectrometer	X-ray fluorescence	Monochrome Camera	Color Filter Camera	Imaging Spectrometer	Imaging Spectrometer	Imaging Spectrometer	Magnetometer
Number of channels	TBD	TBD	1	1	TBD	TBD	TBD	TBD
Size/dimensions (m x m x m)	0.25 x 0.40 x 0.15	0.21 x 0.27 x 0.23	0.70 x 0.27 diameter	0.16 x 0.23 x 0.32 (incl. radiator)	0.26 x 0.20 x 0.06	0.49 x 0.41 x 0.29	0.2 x 0.41 x 0.14	0.08 x 0.10 x 0.12
Instrument mass without contingency (Kg CBE*)	5.2	6.9	8.2	0.9	4.7	17.7	4.5	1.5
Instrument mass contingency (%)	30	30	30	30	30	30	30	30
Instrument mass with contingency (Kg CBE+Reserve)	6.8	9.0	10.7	1.2	6.1	23	5.9	2.0
Instrument average payload power without contingency (W)	48	25	9.3	2.7	48.8	13.5	4.5	1

Instrument average payload power contingency (%)	30	30	30	30	30	30	30	30
Instrument average payload power with contingency (W)	62.4	32.5	12.1	3.5	63.4	17.6	5.9	1.3
Instrument average science data rate^ without contingency (kbps)	1.4	2.22	20,000	20,000	11.1	183	0.69	2
Instrument average science data^ rate contingency (%)	30	30	30	30	30	30	30	30
Instrument average science data^ rate with contingency (kbps)	1.82	2.89	26,000	26,000	14.4	238	0.89	2.6
Instrument Fields of View (degrees)			2.85	92 (mono-chrome) 61 (visible) 59 (UV)				

*CBE = Current Best Estimate.

^Instrument data rate defined as science data rate prior to on-board processing

NOTE: The WAC and NAC pointing requirements in the table are based on LRO mission. They are applicable for the CORAL mission and are typical capabilities for flight systems.

Table 3-3 Payload Mass and Power Table

Instrument Name	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
GCMS	5.2	30	6.8	48.0	30	62.4
XRF	6.9	30	9.0	25.0	30	32.5
NAC (2)	16.4	30	21.3	18.6	30	24.2
WAC	0.9	30	1.2	2.7	30	3.5
UV and Raman Spectrometer	4.7	30	6.1	48.8	30	63.4
IR Spectrometer	17.7	30	23.0	13.5	30	17.6
UV Spectrometer	4.5	30	5.9	4.5	30	5.9
Magnetometer	1.5	30	2.0	1.0	30	1.3
Total Payload Mass	**57.8**	**30**	**75.3**			

3.2 Concept of Operations and Mission Design

2015 BQ311 is both a compelling science target and offers the potential for low-ΔV trajectories. The body is in a small orbit for a centaur, and a Jupiter gravity assist can provide the needed 24-degree inclination change to reach the centaur with relatively low ΔV demands.

Mission requirements are shown in Table 3-4. The primary driving requirements for the mission are: 1) Provide sufficient observing time in orbit to fully characterize the environment and conduct operations to accommodate possible comet-like activity and/or rings, 2) Identify suitable landing locations and map them using high-resolution imaging and/or lidar measurements, 3) conduct *in situ* compositional measurements of at least one location on the surface, and 4) obtain at least once surface and one subsurface sample while keeping the maximum temperature of the sample during acquisition and handling below 200K. These mission requirements are all derived from the four CORAL goals in the science traceability matrix (STM): 1) Understand early solar system compositional reservoirs, 2) Understand the accretion and dynamical evolution of primordial icy planetesimals, 3) Determine the geological and evolutionary processes that have influenced icy planetesimals and 4) Investigate the biologic potential of icy planetesimals and potential brine reservoirs. Details on how the requirements are satisfied can be found in the report Appendix.

Table 3-4 Mission Requirements Traceabiltiy

Mission Requirement (Top Level)	Mission Design Requirements	Spacecraft	Ground System Requirements	Operations Requirements
Mission Lifetime 14 years Rendezvous, globally map and land on a Centaur Global imaging at <= 50 m resolution Identify and map potential landing sites Characterize the environment of the Centaur for comet-like activity and/or rings. Provide communication with Earth during all critical events Mission Reliability Category 2, Class B Conduct Science Operations as defined in the operations concept • Low orbital altitudes to enable mass distribution measurements • Surface sample • Subsurface sample • Max temperature the sample should see before analyses of 200 K	Maximum interplanetary cruise of 13 years Minimum proximity operations and landed science duration of 1 year Launch mass (kg): 4,314 Launch date: 2036 – 2040 Launch Window of at least 21 consecutive days Falcon Heavy Expendable with 5m fairing Launch DLA: +/- 28.5 deg Minimum 4 hours daily contact with Earth	Reliability Category 2, Class B Perform all orbit maneuvers and land at 1 site with a goal of takeoff and landing at a 2nd site Perform global orbital mapping to determine suitable landing sites Operate in environment with potential comet-like activity and/or rings Ka-Band ≥ 40 kbps to Earth with two-way tracking 1 ms timing accuracy with 1e-15 stability relative to ground station Data Storage 3.5 Tbits Conduct Science Operations as defined in the operations concept • Low orbital altitudes to enable mass distribution measurements • Surface sample • Subsurface sample • Max temperature the sample should see before analyses of 200K Accommodate instrument interfaces 3-Axis Stabilized Nadir pointing Lander final actual position within 10 m of target site Target site identified with 1 km clear region of hazards Lander final position knowledge within 1 m Lander velocity at touchdown < 1 m/s vertical, < 0.1 m/s horizontal	34m DSN Antenna, Ka-Band at maximum of 100 Mbps Receive housekeeping & science data telemetry Provide commanding Record/Store science data DDOR Tracking of Spacecraft Provide critical event telecom coverage	Manage time correlations Maneuvers Support DSN passes Monitor Spacecraft state of health Implement contingency procedures Implement science sequences Inventory data & re- transmit if needed Perform ops sim testing

Trajectory options to 2015 BQ311 with launch dates between 2036 and 2040 also offer the ability to readily trade time of flight and delivered mass as the Jupiter flyby can place the spacecraft on a time-tunable, catch-up trajectory that only requires a relatively low-ΔV rendezvous maneuver. Substantially faster catch-up times from Jupiter to the centaur are possible with often acceptable increases the ΔV of the final rendezvous maneuver as shown in Figure 3-4, which depicts the maximum allowable dry mass versus interplanetary time of flight (TOF). In this trade, the maximum propellant load available for maneuvering is 1400 kg, resulting in either 1000 kg or 1100 kg for the interplanetary transfer given 300 and 400 kg proximity operations propellant allocations.

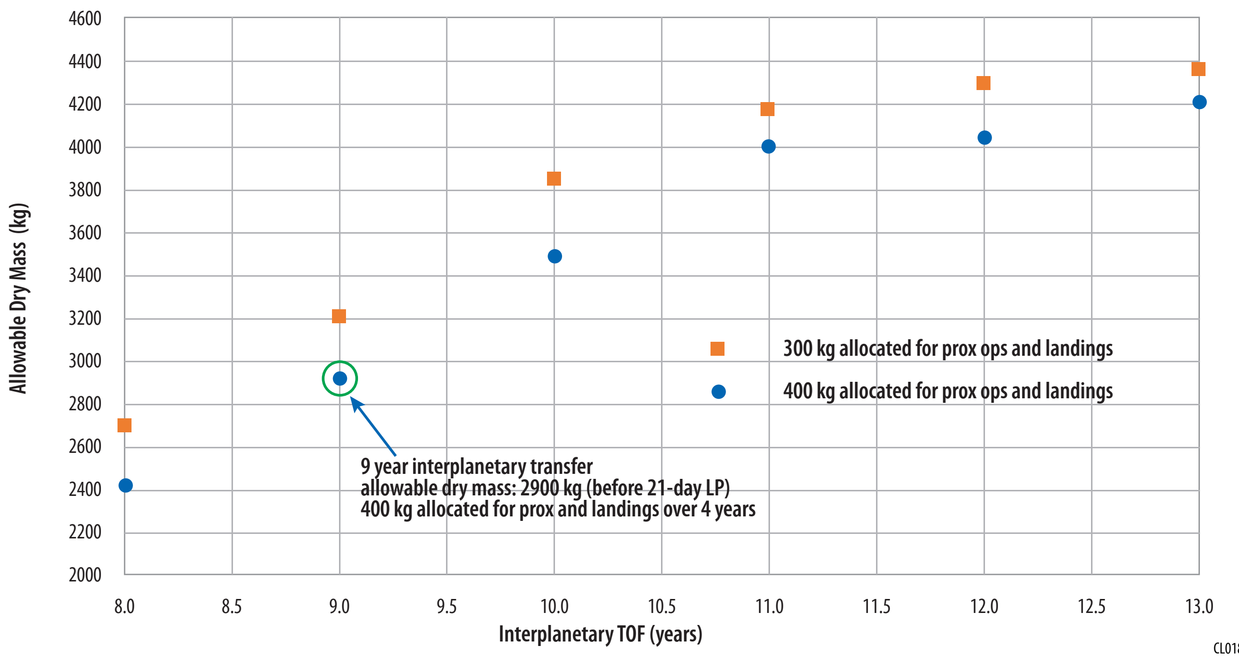

Figure 3-4: 2015 BQ311 dry mass performance versus interplanetary time of flight with different propellant allocations for proximity operations and landings

A trajectory with a nine-year interplanetary transfer time is selected as the design reference mission to balance interplanetary cruise duration and dry mass margin. A nine-year flight time allows ample time for centaur proximity operations and landed operations. The DRM launches in January 2040 on a Falcon Heavy Expendable and utilizes an Earth and Jupiter flyby before rendezvousing with 2015 BQ311 in 2049 as depicted in Figure 5. The propellant required is constrained to be less than 1000 kg with margin for the interplanetary transfer and 400 kg is then allotted for proximity operations, landing, and ACS given the 1,400 kg total propellant load. At centaur arrival the solar distance is approximately 6 AU and the Earth range is roughly 5.1 AU and increasing as shown in Figure 3-6.

In case New Frontiers funding is better aligned with earlier launch dates, launches between 2030 and 2035 were also examined. The phasing between Jupiter and 2015 BQ311 is such that earlier launch dates for shorter cruise flight times do not perform as well as the baseline 2036 to 2040 period. However, 2015 BQ311 is still a viable centaur target with trajectories launching in 2033, 2034, or 2035 capable of delivering at least 3000 kg after centaur rendezvous (Appendix Section 1.3)

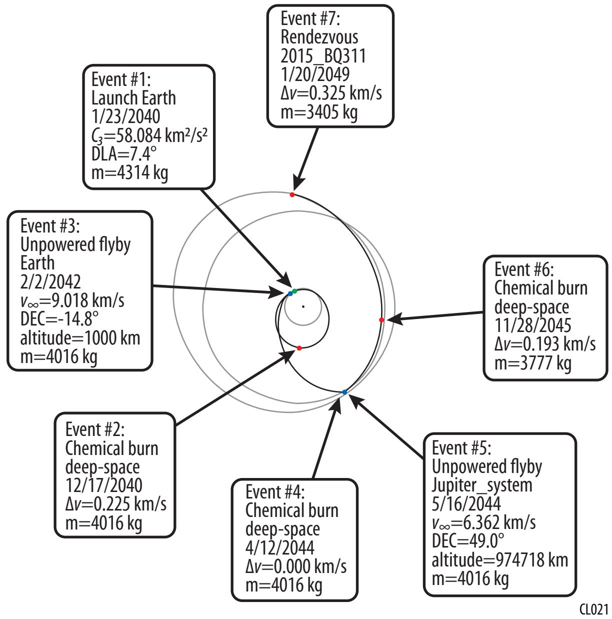

Figure 3-5: 2015 BQ311 optimal trajectory

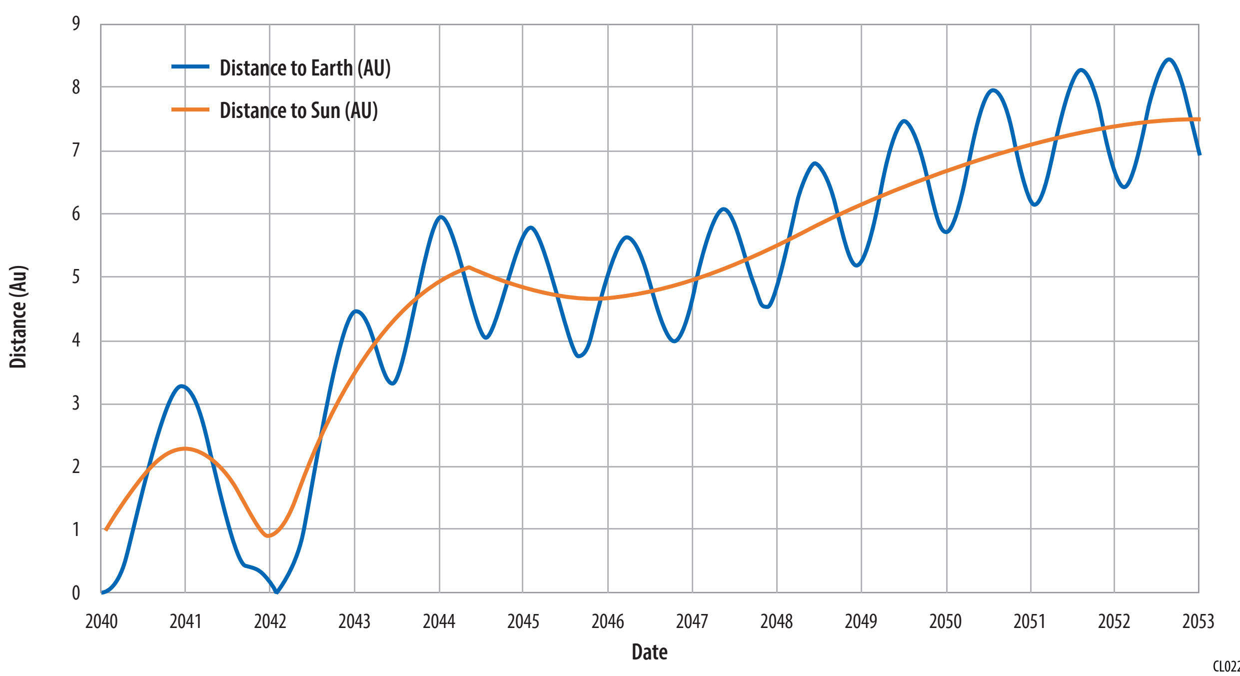

Figure 3-6: 2015 BQ311 DRM distance to Sun and Earth during interplanetary transfer

Table 3-5: Mission Design Table

Parameter	Value	Units
Orbit Parameters (apogee, perigee, inclination, etc.)	**High Altitude Mapping Orbit** 50 km Altitude Polar Near Circular	
Mission Lifetime	168	months
Maximum Eclipse Period	N/A	min
Launch Site	Cape Canaveral	
Spacecraft Mass with contingency (includes instruments)	1,698.3	kg
Propellant Mass without contingency	1,288	kg
Propellant contingency	112.0	%
Propellant Mass with contingency	1,400	kg
Launch Adapter Mass with contingency	72	kg
Total Launch Mass	3,170.3	kg
Launch Vehicle	Falcon Heavy Expendable	Type
Launch Vehicle Lift Capability	3,599.0	kg
Launch Vehicle Mass Margin	428.7	kg
Launch Vehicle Dry Mass Margin (%) [(Capability – Dry Mass) / Dry Mass]	103.3	%
Total Launch Margin % [(Capability – Wet Mass) / Capability]	13	%

Table 3-6 Mission Operations and Ground Data Systems

Communications	Launch and Cruise	Rendezvous, Proximity and Mapping	Landed Science
Number of Contacts	1 per month	1 per day	1 per day
Number of Weeks for Mission Phase, weeks	432	190	8
Downlink Frequency Band, GHz	32	32	32
Telemetry Data Rate(s), kbps	HGA > 77.5 MGA > 0.021 LGA > 0.0007	HGA > 77.5 MGA > 0.021 LGA > 0.0007	HGA > 77.5 MGA > 0.021 LGA > 0.0007
Transmitting Antenna Type(s) and Gain(s), DBi	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4
Transmitter Peak Power, Watts	200	200	200
Downlink Receiving Antenna Gain, DBi	79	79	79
Transmitting Power Amplifier Output, Watts	100	100	100
Total Daily Data Volume, (MB/day)	>139.5	>139.5	>139.5
Uplink Information			
Number of Uplinks per Day	1 per month	1 per day	1 per day
Uplink Frequency Band, GHz	34.45	34.45	34.45
Telecommand Data Rate, kbps	HGA > 137 MGA > 0.613 LGA > 0.036	HGA > 137 MGA > 0.613 LGA > 0.036	HGA > 137 MGA > 0.613 LGA > 0.036
Receiving Antenna Type(s) and Gain(s), DBi	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4

3.3 Flight System

Figure 3-7 shows the Spacecraft Block diagram. The flight system design characteristics are summarized in Table 3-8. Details on the spacecraft subsystems are provided in the Study Report Appendix. Flight System Mass and Power are summarized in Table 3-7.

Tanks were sized for 1,400 kg of fuel and oxidizer, which still allows for up to 3,599 kg of delivered mass.

Since BQ311 is > 9 AU from the sun the use of Solar Arrays would require very large and massive arrays or the use of RTGs. The baseline power system is 2 16-GPHS STEM-RTGs that provide 2290 W (EOL) each for a total of 580 W (EOL). Given the RTG lifetime of 14 years and a nine-year interplanetary transfer, the four years of proximity operations provides allows an additional year for contingencies or extended mission options. The mission power profile is shown in the report Appendix.

The design of the structure is a typical "cylinder-in-a-box" with composite and titanium bracketry and honeycomb panels with composite face sheets, aluminum honeycomb cores and titanium inserts. The panel components are assembled using the clip and post method employed on other composite structures such as LRO. Although the primary role of the CORAL lander is to accommodate the suite of science instruments, the structural design was heavily driven by the propulsion system tanks and accommodating the RTGs.

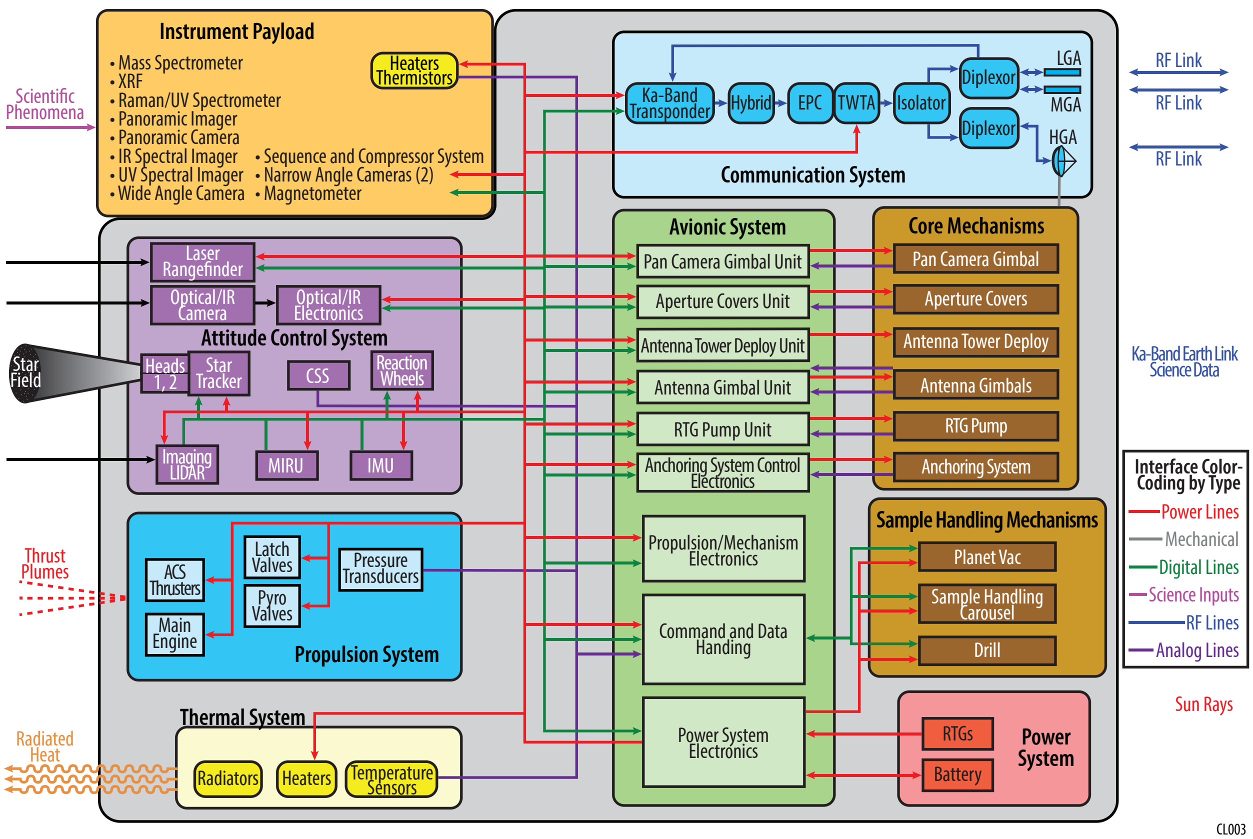

Figure 3-7. Spacecraft Block Diagram.

Table 3-7: Flight System Element Mass and Power Table

	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Structures & Mechanisms	637.8	30	829.1	N/A	N/A	N/A
Thermal Control	61.2	30	78.7	N/A	N/A	N/A
Propulsion (Dry Mass)	193.0	30	250.8	N/A	N/A	N/A
Attitude Control	32.4	10	35.6	103.9	10	114.3
Avionics	65.0	10	71.5	178.0	10	195.8
Telecommunications	34.4	12	38.6	219.0	30	284.7
Power	276.3	17	322.0	5.0	10	5.5
Total Flight Element Dry Bus Mass	1,300.1	25.1	1,626.3	505.9	18.7	600.3

The thermal control approach is to passively control the spacecraft to minimize the need for heater power by a) packaging as much of the temperature sensitive equipment as possible, within the spacecraft to allow the dissipative heat generated to be distributed and shared throughout, to maintain 0°C to 30° within this Warm Electronics Module (WEM), and b) minimizing radiator area by using louvers to "close" during colder environments (most of mission).

The spacecraft uses a communication system that minimizes mass and maximizes data rate using only Ka-Band. A 3m Ka-band High Gain Antenna (HGA) as its primary means of communicating with Earth. Medium gain antenna (MGA) and low gain antenna (LGA) are used for telemetry and safe mode.

Table 3-8 Flight System Element Characteristics Table

Flight System Element Parameters (as appropriate)	Value/ Summary, units
General	
Design Life, months	168
Structure	
Structures material (aluminum, exotic, composite, etc.)	Composite
Number of articulated structures	3
Number of deployed structures	3
Aeroshell diameter, m	N/A
Thermal Control	
Type of thermal control used	Passive, heaters and louvers
Propulsion	
Estimated delta-V budget, m/s	1,376 m/s (21-day LP mission total)
Propulsion type(s) and associated propellant(s)/oxidizer(s)	Regulated bipropellant
Number of thrusters and tanks	20 ACS Thrusters, 4 Main Engines, 2 MMH Tanks, 2 NTO Tanks, 2 Pressurant Tanks
Specific impulse of each propulsion mode, seconds	Primary, ME Mode: 315s (299.7s at -3σ) Secondary, ACS Mode: 300s (285s at -3σ)
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis, spinner

Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial, Venus-Nadir, Solar
Attitude control capability, degrees	< 0.1 degrees
Attitude knowledge limit, degrees	< 30 arcsec
Agility requirements (maneuvers, scanning, etc.)	DSM, Landing, TRN, HA
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	HGA, PanCam Arm
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	CSS: 2π stradian ST: 30 arcsec boresight IMU: ARW = 0.07 deg/root-hour, Bias: 1 deg/hr RCS: 5 lb Wheel: 0.2 Nm, 250 NMS
Command & Data Handling	
Flight Element housekeeping data rate	1 kbps
Data storage capacity	3.5 Tbits
Maximum storage record rate	2,000 kbps
Maximum storage playback rate	2,000 kbps
Power	
Type of array structure (rigid, flexible, body mounted, deployed, articulated)	N/A
Array size, meters x meters	N/A
MRTG	Two 16-GPHS STEM-RTGs
Solar cell type (Si, GaAs, Multi-junction GaAs, concentrators)	N/A
Expected power generation at Beginning of Life (BOL) and End of Life (EOL)	800 W (BOL), 580 W (EOL)
On-orbit average power consumption	450 W
Battery type	Li-ion
Battery storage capacity	15.25 amp-hours

3.4 Mission Risks

The CORAL team developed a risk register to identify areas for consideration during the next steps of developing the mission. The top risks are listed in Table 3-9.

Table 3-9: CORAL Top Mission Risks

Risk Name LxC	Risk Statement	Approach	Comments
Landing 2x5 (L x C) Expected Closure: Upon landing	**Given that:** the CORAL mission relies on landing on the surface of the centaur **There is a probability of:** Crash or hard landing on the centaur surface **Resulting in:** instrument damage and degradation of science return	Research/ Mitigate/ Accept (R/M/A)	• The CORAL spacecraft includes an imaging lidar to aid with landing • The CORAL mission plan includes 52 months of proximity operations that will map the surface of the centaur and before landing
Debris at Landing Side 2 x 2 (L x C) Expected Closure: During proximity operations	**Given that:** the target body may have comet-like activity **There is a possibility that:** there will be large debris on the surface **Resulting in:** extended proximity operations to identify areas for safe landing	Research/ Mitigate/ Accept (R/M/A)	• The CORAL team will develop operational scenarios and hazard avoidance algorithms that consider landing in a variety of surfaces. • CORAL uses an imaging LIDAR and ACS thrusters to perform hazard avoidance maneuvers.

Surface Dust 3 x 2 (L x C) Expected Closure: PDR	**Given that:** the surface of the centaur will be dusty **There is a possibility that:** the dust mitigation requirements may grow **Resulting in:** increases in mass and cost	Research/ Mitigate/ Accept (R/M/A)	• Sensitive instruments will be equipped with covers for optical surfaces • The system design will make use of debris deflection shields to direct sampling induced debris on a ballistic path away from the space-craft/lander
Sample Collection 3 x 2 (L x C) Expected Closure: PDR	**Given that:** the centaur surface properties are unknown **There is a possibility that:** the sample acquisition system requirements will grow to enable more options **Resulting in:** increases in mass and cost	Research/ Mitigate/ Accept (R/M/A)	• The sample acquisition system is a well developed system • The sample collection system will be extensively tested with a variety of samples to clearly identify and document its capabilities • The team will develop detail proximity operation protocol to enable landing side selection
Obsolete Parts 5 x 1(L x C) Expected Closure: PDR	**Given that:** the instrument designs used during the development of the mission concept are heritage from previous missions **There is a possibility that:** the design will include obsolete parts and will need to be updated **Resulting in:** cost and schedule overruns	Watch/ Research/ Mitigate (W/R/M)	• The team will perform a detail review of all instrument designs during the early phases of the project
Stuck Drill 2x2(L x C) Expected Closure: During Surface Operations	**Given that:** sample acquisition depends on drilling **There is a probability that:** the drill will get stuck during drilling operations **Resulting in:** inability to acquire sample from the planned depth	Research/ Mitigate/ Accept (R/M/A)	• The lander carries redundant drills and sample acquisition systems • The team will consider alternate designs during the early phases of the project formulation.
Hopping 1x3 (L x C) Expected Closure: During Surface Operations	**Given that:** the lander employs three drills **There is a probability that:** one or more drills will get stuck in the centaur surface disabling the lander from hoping to another location **Resulting in:** degradation of science return	Research/ Mitigate/ Accept (R/M/A)	• Extensive testing of drills during early project phases • Consider drill ejection technologies
Sample size 1x2 (L x C) Expected Closure: During Surface Operations	**Given that:** the surface properties and material granular size is unknown **There is a probability that:** the PlanetVac will be unable to acquire and pneumatically transfer sufficient sample quantity to the carousel **Resulting in:** degradation of science return	Research/ Mitigate/ Accept (R/M/A)	• PlaneVac will undergo extensive testing with a variety of sample granular sizes • The team will consider ways to enhance the capabilities of the PlaneVac system • The team will consider alternate sample collection options in the early phases of the project
Sample Contamination 2x3 (L x C) Expected Closure: During Surface Operations	**Given that:** thrusters will be used for landing **There is a probability of:** sample contamination due to either plume material deposition or disruption of the natural surface **Resulting in:** degraded science return	Research/ Mitigate/ Accept (R/M/A)	• The team will carefully consider the location of the thrusters relative to the sample collection locations to avoid sample contamination • The team will simulate the plume structure and try to identify approaches to mitigate the possibility of contamination

Anchoring 1x1 (L x C) Expected Closure: During Surface Operations	**Given that:** the lander will need to be anchored on the centaur surface using harpoons **There is a probability that:** the harpoon will fail to properly secure the lander on the surface **Resulting in:** the lander tilting or sliding from landed position	Research/ Mitigate/ Accept (R/M/A)	• redundant harpoons will be available to attempt again at a different location • consider using lower preload on the drill for sampling • the team will consider alternate approaches for securing the lander on the surface

4.0 DEVELOPMENT SCHEDULE AND SCHEDULE CONSTRAINTS

4.1 High-Level Mission Schedule

The CORAL high-level mission schedule is shown in Figure 4-1. For a January 2040 launch, it was assumed that a NF Announcement of Opportunity (AO) will come during the early 2030 decadal. The schedule is characteristic of a new frontier mission. The schedule would slide accordingly for a slightly different NF AO date, provided it does not violate the launch window.

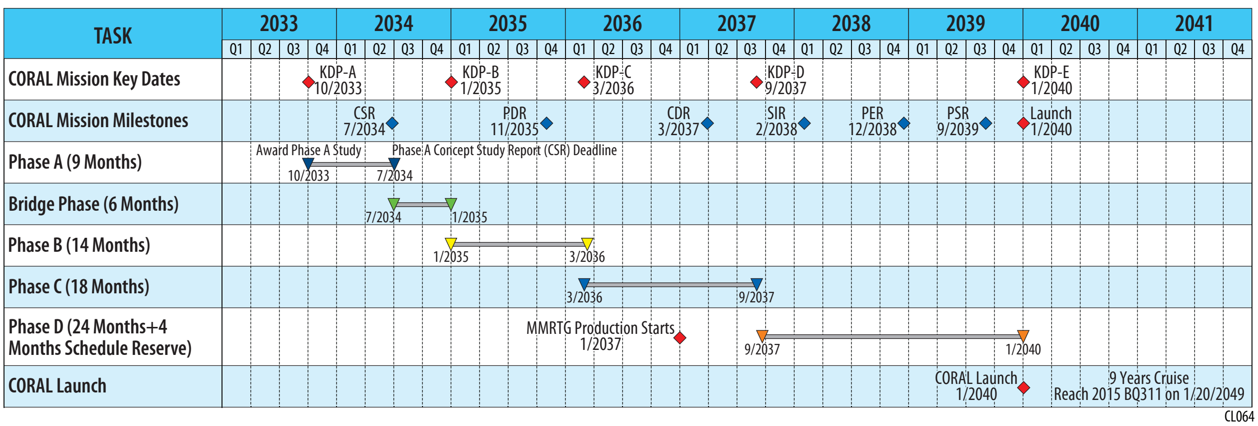

Figure 4-1: CORAL high-level schedule

The CORAL key phase durations are shown in Table 4-1

Table 4-1: CORAL Key Phase Duration Table.

Phase A – Conceptual Design	9 Months
Phase B – Preliminary Design	14 Months
Phase C – Detailed Design	18 Months
Phase D – Integration & Test	28 Months
Start of Phase B to PDR	10 Months
Start of Phase B to CDR	27 Months
TOTAL Development time Phase B-D	60 Months
Launch Date	January 2, 2040
Phase E – Operations and Sustainment	164 Months
Cruise Phase Duration (9 years)	108 Months
Arrival at Destination (2008 SJ236)	January 20, 2049
Proximity operations (4 years and 4 months)	52 Months
Landing	May 2053
Surface Operations	4 Months
Phase F - Closeout	6 Months

5.0 MISSION LIFE-CYCLE COST

The CORAL mission concept is a mature concept as it uses technologies demonstrated in past missions. The mission concept described in this report is technically feasible since most instruments and subsystems have excellent heritage and were used in previous missions. The team determined that the CORAL maturity level (CML) is at CML 5. The mission life-cycle cost of the full mission was determined to be slightly above the New Frontiers class. Furthermore, it was determined that by exercising selective descopes (while maintaining threshold science) the cost can be reduced to NF levels and the mission can fit under the NF cost cap.

The study team followed all ground rules and assumptions described in the "Ground Rules for Mission Concept Studies in Support of Planetary Decadal Survey." Cost estimates are presented in fiscal year 2025 dollars (FY25$). The estimate assumes that NASA will bear all the costs associated with the development of all instruments, the spacecraft/lander, and other special purpose instruments on the spacecraft. The team used a standard mission WBS and the cost estimate covers activities through the end of Phase F. It was assumed that the Launch Vehicle Services Program will provide the vehicle to deliver the CORAL spacecraft/lander to the centaur. It is anticipated that at the time of the AO release the CORAL team will be able to collaborate (through a competitive Partnering Opportunity Document (POD) or other equivalent process) with commercial instrument providers to ensure that the mission will take advantage of the latest technological advances of the time.

5.1 Costing Methodology and Basis of Estimate

CORAL costing methodology is based on a mix of approaches such as parametric cost modeling for the spacecraft/lander, analogous costing based on heritage instruments from past missions, and historic cost wrap factors (to account for WBSs such as project management, system engineering, etc.). The team did not use grassroots estimates for CORAL. A reserve of 50% for Phases A-D and 25% for Phases E-F was added to the derived cost. No cost or reserve were added to the estimate for the Launch Vehicle. No reserves were added for the MMRTG as recommended in the Ground rules. All costs are in Fiscal Year 2025 dollars (FY25$).

The CORAL study team developed the mission cost using GSFC's cost estimation process for early formulation. During the study the team focused primarily on the wholeness of the technical design.

The mission cost was developed mostly through analogous costing by identifying heritage instrument designs used in past missions. The team was able to identify analogous instruments from the past and used their cost as the basis of estimate for CORAL instruments. The process included identifying the analogous system, retrieving the cost from the CADRe (Cost Analysis Data Requirement) database, and then inflating the cost to FY25$. This process artificially inflated some of the instrument cost since on top of the cost identified in CADRe (which represents the real cost of the instrument and includes the reserves) an additional 50% was added.

Table 5-1 provides the list of CORAL instruments and their analogous heritage instruments used during costing activities. The first column lists the CORAL instrument name/description while the second column contains information of the heritage instrument and the mission.

Table 5-1: CORAL Instruments and Analogous Instruments

CORAL Instrument/Subsystem	Analogous Instrument/Subsystem
Gas Chromatograph-Isotope Ratio Mass Spectrometer	Ptolemy - Gas Chromatograph–Isotope Ratio–Mass Spectrometer aboard the Philae lander element of the Rosetta mission
X-ray Lithochemistry Instrument,	PIXL - Planetary Instrument for X-ray Lithochemistry, MARS 2020 mission
Ultraviolet (UV) Raman Spectrometer	Cost estimate provided by SHERLOC (Scanning Habitable Environments with Raman & Luminescence for Organics & Chemicals) team, MARS 2020 mission
Visible/Infrared Spectrometer and Mapper	OVIRS- OSIRIS-REx Visible and InfraRed Spectrometer, OSIRIS-REx (Origins, Spectral Interpretation, Resource Identification, Security, Regolith Explorer) mission

UV imaging telescope/spectrometer	Alice - a compact, general-purpose UV imaging telescope/spectrometer, flying aboard NASA's New Horizons mission
Wide Angle Camera (WAC) and Narrow Angle Camera (NAC)	Lunar Reconnaissance Orbiter Camera (LROC), aboard Lunar Reconnaissance Orbiter (LRO)
CORAL Magnetometer	Cost borrowed from "2020 Venus Flagship Mission Study"
CORAL Sample Acquisition and Handling System	Cost borrowed from CERES Habitability mission Study
Navigation LIDAR	OLA - OSIRIS-REx Laser Altimeter, OSIRIS REx mission
Lander Harpoon Anchoring System	Philae lander anchoring harpoon, Rosetta mission
Boom Pan-Cam	Cost borrowed from "2020 Venus Flagship Mission Study"

The CORAL cost is organized, defined, and estimated in accordance with the NASA Standard Work Breakdown Structure (WBS), which is compliant with NPR 7120.5E. The majority of instrument WBS numbers are estimated by analogy. For WBS elements not estimated by analogy, wrap factors derived from historical missions are used. Wrap factors were used for Project Management (WBS 1), System Engineering (WBS 2), Safety and Mission Assurances (WBS 3), Science (WBS 4), Payload Management (WBS 5.1), Payload System Engineering (WBS 5.2), Flight Segment Management (WBS 6.1), Flight Segment System Engineering (WBS 6.2), Flight Segment Safety and Mission Assurance (WBS6.3), Mission Operations System (WBS 7), Ground Data System (WBS 9) and Project Integration and Test (WBS 10).

The spacecraft/lander cost was derived using the cost from a previous study where a detailed design of the spacecraft/lander was developed and costed, and by adding CORAL specific subsystems. The cost was verified by using a parametric model in combination with the Master Equipment List (MEL). The cost estimate using the parametric model was done by a member of the GSFC's Cost Estimating, Modeling and Analysis (CEMA) Office.

The cost estimate presented in this report is intended for informational, budgetary, and planning purposes only and does not constitute a commitment on the part of GSFC. It lacks the rigor of more detail grassroot estimates characteristic of longer effort carried out in the early phases of a project.

5.2 Cost Estimate(S)

The cost estimate for the CORAL mission is presented in Table 5-2. The table includes two estimates; one is for the full mission while the second one is the cost for the descoped version of the CORAL mission. The A-D cost for the full mission is \$1.3B while the cost of the descoped mission is \$1.16B (all in FY25\$). Based on this cost estimate and considering the high reserve posture mandate for the study, a version of the CORAL mission can fit in the New Frontiers envelope.

Table 5-2: CORAL Cost Estimate in FY25$M

WBS #	Description	Phase A	Phase B-D Full Mission	Phase B-D Descoped Mission
01	Project Management	4.00	69.87	62.68
02	System Engineering		44.92	40.29
03	Safety & Mission Assurance		39.93	35.82
04	Science		24.96	22.39
05	Payload Total		272.83	170.04
05.01	Payload/Observatory Management		12.52	7.80
05.02	Payload System Engineering		10.01	6.24
05.03	Gas Chromatograph-Isotope Ratio Mass Spectrometer		13.50	13.50
05.04	X-ray Lithochemistry Instrument,		71.51	
05.05	Ultraviolet (UV) Raman Spectrometer		45.00	45.00

05.06	Visible/Infrared Spectrometer and Mapper		50.25	50.25
05.07	UV imaging telescope/spectrometer		14.10	
05.08	Wide Angle Camera (WAC) and Narrow Angle Camera (NAC)		47.25	47.25
05.09	CORAL Magnetometer		8.70	
06	Flight System Total (Spacecraft/Lander)		677.88	677.88
06.01	Flight Segment Management		38.62	38.62
06.02	Flight Segment System Engineering		24.61	24.61
06.03	Flight Segment Safety & Mission Assurance		18.46	18.46
06.04	Spacecraft/Lander		412.50	412.50
06.05	Next Gen MMRTG ($70M+$25M=$95M)		95.00	95.00
06.06	CORAL Sample Acquisition and Handling System		40.70	40.70
06.07	Navigation LIDAR		22.50	22.50
06.08	Lander Harpoon Anchoring System (x4)		18.00	18.00
06.09	Boom Pan-Cam		7.50	7.50
07	Mission Operations System		29.95	26.86
08	Launch Vehicle			
09	Ground Data System		64.88	58.20
10	Project Integration & Test		71.30	63.59
	Total	**4.00**	**896.02**	**803.50**
	Reserves (Note: No Reserves for RTG)		**400.51**	**354.25**
	Grand Total Phases A-D		**1300.52**	**1161.75**
	Phase E-F Cost (includes 25% Reserves)		**431.25**	**431.25**

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APPENDIX: DESIGN STUDY DETAILS

1.0 MISSION OVERVIEW

The goal of the *CORAL* mission is to measure the chemical and physical properties of these dynamically evolved but compositionally primitive small bodies to constrain the composition and evolution of icy planetesimals. This mission concept demonstrates the feasibility of globally charactering and obtaining landed *in situ* compositional measurements of a Centaur within a New Frontiers class mission.

The CORAL Spacecraft is shown in Figure 1-1. The Spacecraft is launched on January 23, 2040 by a Falcon Heavy Expendable with a 5m fairing from Cape Canaveral, Florida. The Spacecraft will spend nine-years in interplanetary transfer and rendezvous with Centaur BQ311 on January 20, 2049 (Figure 1-2).

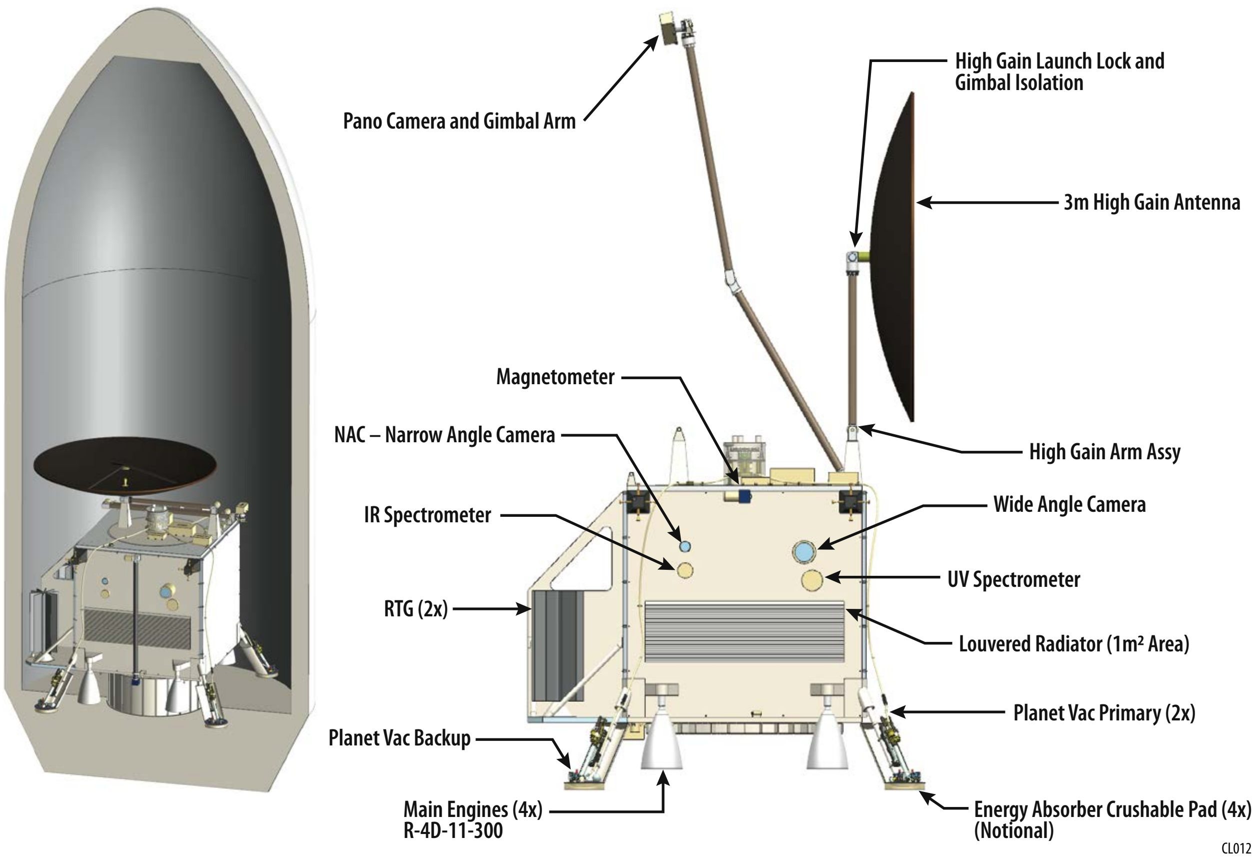

Figure 1-1: CORAL Spacecraft

After rendezvousing with BQ311 the spacecraft will begin proximity operations (Section 2.2) that last ~4 years. The length of the proximity operations is driven by the assumption that the mapping subphases will require a similar volume of imagery as was used by OSIRIS-Rex. Since proximity operations around BQ311 occur at distances from Earth of >6.7 AU the data rate is limited and the resulting duration required to return all the images is long (Section 2.2).

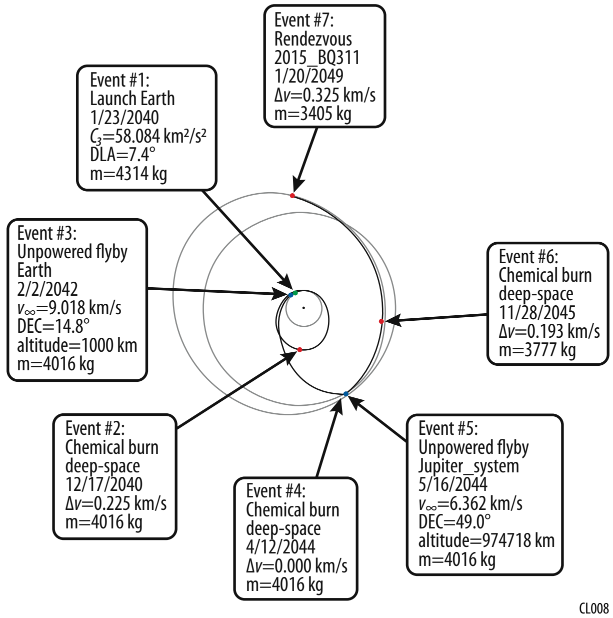

Figure 1-2: CORAL Trajectory

After proximity operations have completed and the landing site has been chosen the spacecraft will land and conduct landed operations for 8 weeks. Upon completion of landed science operations the option exists for the lander to takeoff and land at a second site.

The CORAL payload is comprised of: 1) Science instruments, 2) Navigation hardware, a panoramic camera and robotic arm, and 3) a Sampling acquisition and handling system with a panoramic camera and robotic arm, as shown in Table 1-1. The science heritage instruments were selected to be a set of representative instruments that meet the high priority science requirements of the STM. It is based on previously flown instruments that would allow for implementation in the mission design without a need for technology development. Figures 1-1 and 1-3 show how the payload subsystems are distributed on the lander.

Table 1-1: CORAL Payload

Science Instruments
GCMS (Gas-chromatograph mass spectrometer)
NAC (Narrow Angle Camera)
UV and Raman Spectrometer (Ultraviolet Raman Spectrometer)
UV Spectrometer (Ultraviolet Spectrometer)
IR Spectrometer (Visible and Infrared Spectrometer)
WAC (Wide Angle Camera)
XRF (X-Ray fluorescence)
Magnetometer
Navigation Hardware
Laser Range finder
LiDAR
Optical/IR Cameras

Sample Acquisition and Handling System
Panoramic Camera and Arm
Drill
SAS (Sample Acquisition System)
Carousel

During proximity operations, the instruments which are similar to heritage infrared spectrometer (ex. OVIRS/OSIRIS REx), ultraviolet spectrometer (ex. Alice/Rosetta) and high-resolution imagers (ex. WAC and NAC/LRO) are used for characterizing the environment and target. The imagers and imaging spectrometers are mounted on the same side of spacecraft and will face nadir during most of the orbital phase.

The landed phase has a Gas-chromatograph mass spectrometer (ex. Ptolemy/Rosetta), X-ray fluorescence instrument (ex. PIXL/Mars 2020 Perseverance), and a combined Raman and UV spectrometer system (ex. SHERLOC/Mars 2020 Perseverance) to perform in-situ elemental, isotopic, and organic analyses of the samples on the Centaur surface. The in-situ instruments are mounted on the top deck encircling the sample carousel (see Figure 1-3).

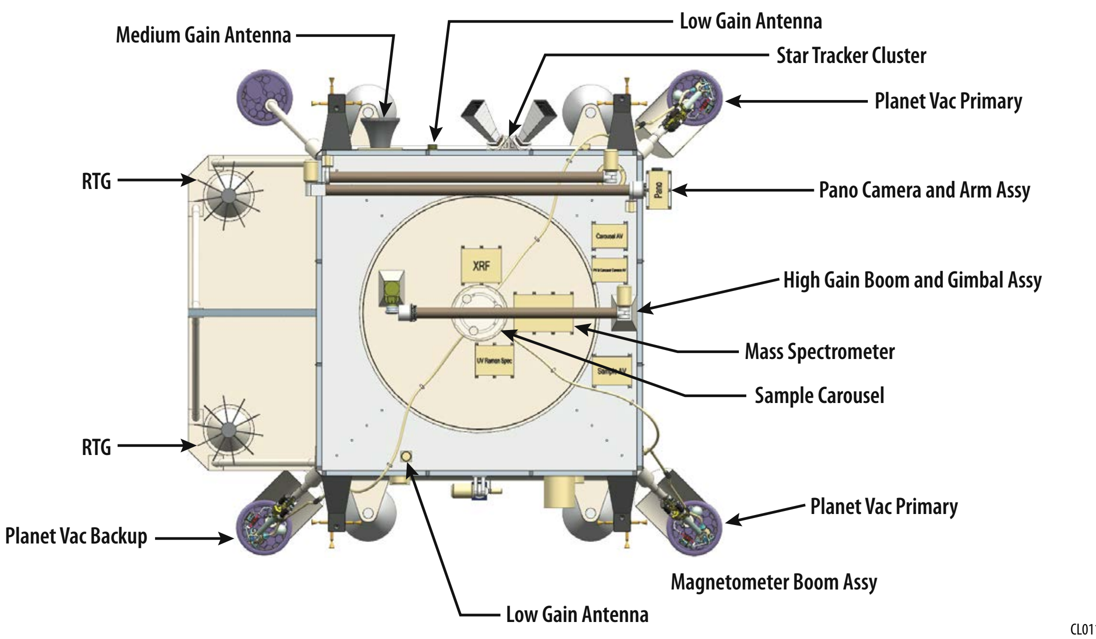

Figure 1-3: CORAL top deck and RTG Patio

The standard fluxgate magnetometer (Magnetometer/MAVEN) is mounted on a deployable boom to measure the magnetic properties from orbit and near the surface. The spacecraft uses a deployable panoramic camera for contextual imaging of the landing site and local region sampled for chemical analysis.

Since BQ311 is ~ 6.7 AU from the sun the use of Solar Arrays would require very large and massive arrays or the use of RTGs. The baseline power system is 2 (two) 16-GPHS STEM-RTGs that together provide 580 W (EOL). Given the RTG lifetime of 14 years and a nine-year interplanetary transfer, the four years of proximity operations provides allows an additional year for contingencies or extended mission options. The use of the RTGs allow the use of a heat pump to transfer heat and eliminates ~200 W of power that would have been needed by the thermal subsystem if Solar Arrays had been used. The mission power profile is shown in Section 3.6.

A 3m mechanical arm with two 1.5m sections and gimbals at the base, elbow and wrist provides maximum flexibility for the Panorama Camera to obtain context imagery around the landed spacecraft.

A pneumatic sample acquisition system (SAS) which is similar to PlanetVac has a drill and is responsible for collecting samples from surface and sub surface depths and distributing samples via a carousel to the GCMS, XRF and UV and Raman combined instruments. Three SAS will be implemented in this design, each attached near the lander footpads. The SAS located closest to the NTRGs will be used as a backup system.

The drill is 10 cm in length, and 1.25 cm in diameter. This should provide enough material volume for the intended number of analysis at the various depths with sufficient margin. This is the same class size of drill used on MSL which is a rotary percussive drill of 6 cm in length. Since the depth to which we can drill depends on the unknown surface topography and the placement of the SAS cone on the surface there is no guarantee the drill depth corresponds to sample at that depth, the sample collected could be from shallower depths of the same hole. We do know that the sample collected is no deeper than the extension of the drill depth. The operations concept is to drill in incremental depths of of 2 or 3 cm and then collect that sample. We will conduct the drilling operation 3-4 times before the drill is fully extend. Drill extension information will be accurate to mm resolution.

The design of the structure is a typical "cylinder-in-a-box" with composite and titanium bracketry and honeycomb panels with composite face sheets, aluminum honeycomb cores and titanium inserts. The panel components are assembled using the clip and post method employed on other composite structures such as LRO. Although the primary role of the CORAL lander is to accommodate the suite of science instruments, the structural design was heavily driven by the propulsion system tanks and the accommodating the RTGs (Section 3.10).

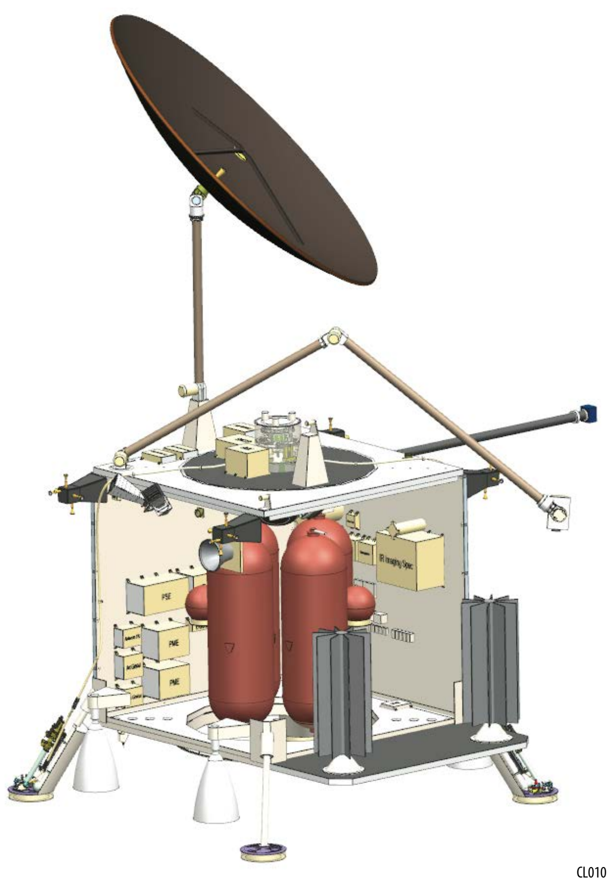

Figure 1-4: Warm Electronics Module keeps the electronics and propellant tank warm

The thermal control approach is to passively control the spacecraft to minimize the need for heater power by a) packaging as much of the temperature sensitive equipment as possible, within the spacecraft to allow the dissipative heat generated to be distributed and shared throughout, to maintain 0°C to 30° within this Warm Electronics Module (WEM) as shown in Figure 1-4, and b) minimizing radiator area by using louvers to "close" during colder environments (most of mission).

The spacecraft uses a communication system (Section 3.4) that minimizes mass and maximizes data rate using only Ka-Band. A 3m Ka-band High Gain Antenna (HGA) as its primary means of communicating with Earth. Medium gain antenna (MGA) and low gain antenna (LGA) are used for telemetry and safe mode.

Figure 1-5 shows the Spacecraft Block diagram. Details on the spacecraft subsystems are provided in Section 3.0.

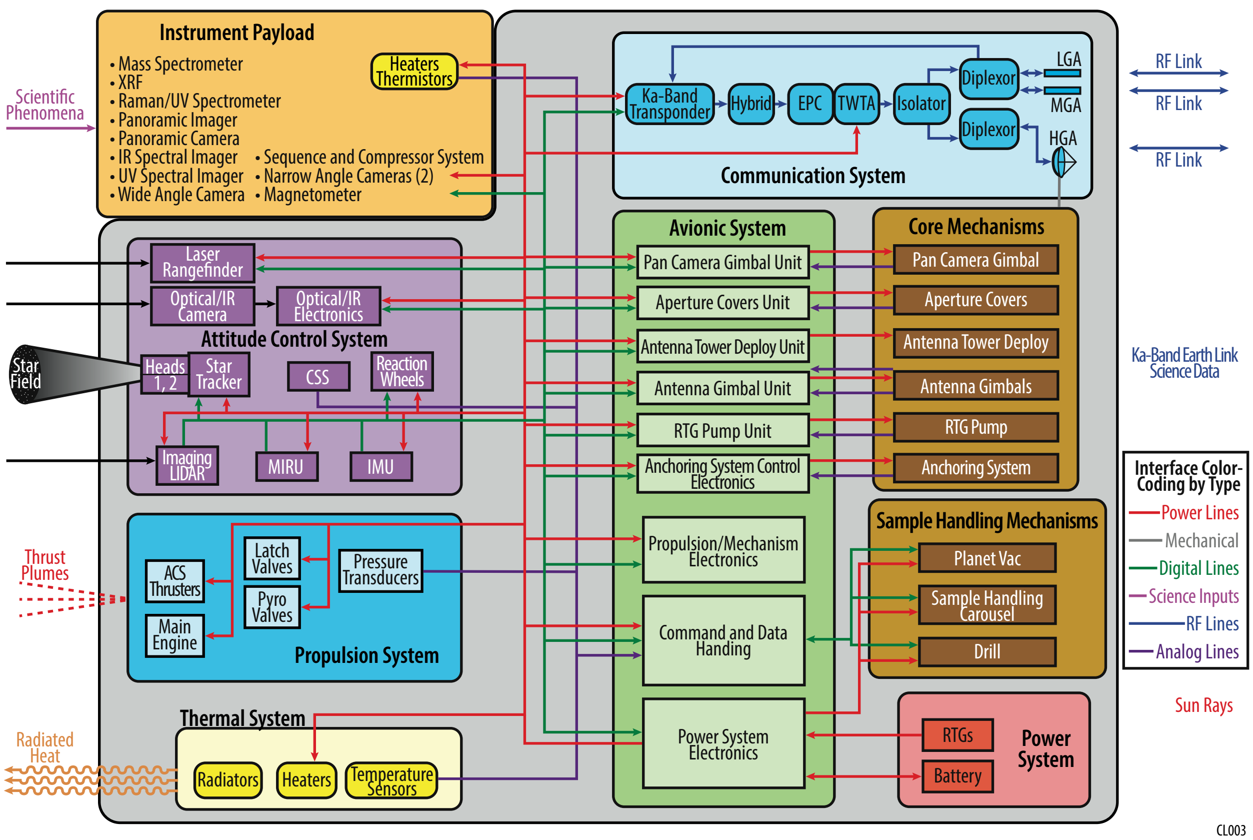

Figure 1-5: Spacecraft Block Diagram

1.1 Key Architecture and Mission Trades

Centaurs pose challenging mission design problems. They are characterized by large orbits and often have high eccentricities and inclinations, necessitating high ΔV trajectories. While propellant-efficient electric propulsion can enable high ΔV missions, the thrusting required for rendezvous at a Centaur can drive solar array sizes that are prohibitively large because of the large solar distance. Similarly, radioisotope electric propulsion options are typically limited to low power because of the cost of RTGs and are often unable to provide adequate thrust for a high-mass spacecraft such as lander to rendezvous with a Centaur.

The core mission trade for the study is to identify Centaurs and propulsion options that enable a lander mission given the inherent high propellant cost of rendezvous and launch vehicle limitations. Of particular interest, is the viability of landing on Chiron, which was previously targeted in a 2010 Decadal Study.

A broad mission design trade (Section 1.3) was conducted in order to both determine a design reference mission for detailed evaluation as well as to characterize the target and mission architecture design space. Over 550 Centaur targets are evaluated based on 'Centaur' or 'Chiron-type comet' designations in the JPL Small-Body Database and science team input.

In addition to determining the target Centaur, a variety of mission architecture options are considered using different propulsion options including traditional chemical-only bi-propellant propulsion with the option of propulsion stages, RTG electric propulsion (REP), solar electric propulsion (SEP), as well hybrid options with chemical propulsion for the arrival maneuver and electric propulsion for the rest interplanetary transfer. For REP missions, one to three RTGs were evaluated, with a preference for one or two units given the high cost of RTGs. The 16-GPHS STEM-RTG, the only RTG option considered as it is the highest performing allowed in the study rules, provides 400 W per RTG to be divided for EP and non-EP power, with the remaining power available for spacecraft electronics and payload. For SEP missions, solar arrays providing between 10 and 40 kW at one AU in 10 kW increments were traded. Both NASA's Evolutionary Xenon Thruster (NEXT) and XIPS-25 thrusters were considered with two to four active thrusters. NEXT thrusters are notably efficient at high powers, but have a high minimum input power, making the low input power of XIPS-25 potentially advantageous for some architectures. The trades are discussed in Section 1.3. The preferred baseline was briefly discussed above in the Mission Overview (Section 1.0) and is further discussed in the remainder of this document.

Additional trades conducted included whether to include both Ka- and X-Band hardware. It was determined that the benefit of an only Ka-Band system outweighed any redundancy or flexibility having the X-Band components offered. The power constraints on the system dictated that only one communication system could be active at a time so the excess complexity of two systems was unnecessary and with the higher gain and higher data rates Ka-Band only was the most logical choice.

Another key trade was determining the need for the gimbal on the HGA. The gimbal provides the necessary flexibility for the unknown landing site parameters and therefore outweighed the concern of higher losses in the communications path from the added length of waveguide and cables this choice requires.

The HGA size was also up for trade as there are multiple heritage examples of differing parabolic dish sizes. Dishes with 2- and 3-meter diameters have been successfully used on other deep space missions so those were the main choices with flight heritage, but larger antennas were also considered as they provide significant gain increases. It was decided that keeping within heritage examples provided a known quantity that outweighed the benefit of a larger, more costly dish and allowed the spacecraft to fit comfortably in the launch fairing without introducing a complex, high cost, deployable antenna. The decision to go with the 3-meter antenna was obvious as it more than doubles the gain of the system without a large increase in cost or complexity as compared with a 2-meter dish.

1.2 Mission Requirements

Mission requirements are shown in Table 1-2. The primary driving Requirements for the mission are: 1) Provide sufficient observing time in orbit to fully characterize the environment and conduct operations to accommodate possible comet-like activity and/or rings, 2) Identify suitable landing locations and map them using high-resolution imaging and/or lidar measurements, 3) conduct in situ compositional measurements of at least one location on the surface, and 4) obtain at least once surface and one subsurface sample while keeping the maximum temperature of the sample during acquisition and handling below 200K. These mission requirements are all derived from the four CORAL goals in the science traceability matrix (STM): 1) Understand early solar system compositional reservoirs, 2) Understand the accretion and dynamical evolution of primordial icy planetesimals, 3) Determine the geological and evolutionary processes that have influenced icy planetesimals and 4) Investigate the biologic potential of icy planetesimals and potential brine reservoirs.

Table 1-2: Mission Requirements

Mission Requirement (Top Level)	Mission Design Requirements	Spacecraft	Ground System Requirements	Operations Requirements
Mission Lifetime 14 years Rendezvous, globally map and land on a Centaur Global imaging at <= 50 m resolution Identify and map potential landing sites Characterize the environment of the Centaur for comet-like activity and/or rings. Provide communication with Earth during all critical events Mission Reliability Category 2, Class B Conduct Science Operations as defined in the operations concept • Low orbital altitudes to enable mass distribution measurements • Surface sample • Subsurface sample • Max temperature the sample should see before analyses of 200 K	Maximum interplanetary cruise of 13 years Minimum proximity operations and landed science duration of 1 year Launch mass (kg): 3,599 Launch date: 2036 – 2040 Launch Window of at least 21 consecutive days Falcon Heavy Expendable with 5m fairing Launch DLA: +/- 28.5 deg Minimum 4 hours daily contact with Earth	Reliability Category 2, Class B Perform all orbit maneuvers and land at 1 site with a goal of takeoff and landing at a 2nd site Perform global orbital mapping to determine suitable landing sites Operate in environment with potential comet-like activity and/or rings Ka-Band $\geq$ 40 kbps to Earth with two-way tracking 1 ms timing accuracy with 1e-15 stability relative to ground station Data Storage 3.5 Tbits Conduct Science Operations as defined in the operations concept • Low orbital altitudes to enable mass distribution measurements • Surface sample • Subsurface sample • Max temperature the sample should see before analyses of 200K Accommodate instrument interfaces 3-Axis Stabilized Nadir pointing Lander final actual position within 10 m of target site Target site identified with 1 km clear region of hazards Lander final position knowledge within 1 m Lander velocity at touchdown < 1 m/s vertical, < 0.1 m/s horizontal	34m DSN Antenna, Ka-Band at maximum of 100 Mbps Receive housekeeping & science data telemetry Provide commanding Record/Store science data DDOR Tracking of Spacecraft Provide critical event telecom coverage	Manage time correlations Maneuvers Support DSN passes Monitor Spacecraft state of health Implement contingency procedures Implement science sequences Inventory data & re- transmit if needed Perform ops sim testing

1.3 Mission Design

Centaurs pose challenging rendezvous mission design problems. They are characterized by large orbits and often have high eccentricities and inclinations, necessitating high ΔV trajectories. While propellant-efficient electric propulsion can enable high ΔV missions, the thrusting required for rendezvous at a Centaur can drive solar array sizes that are prohibitively large because of the large solar distance. Similarly, radioisotope electric propulsion options are typically limited to low power because of the cost of RTGs and are often unable to provide adequate thrust for a high-mass spacecraft such as lander to rendezvous with a Centaur.

The core mission design objective for the study is to identify Centaurs and propulsion options that enable a lander mission given the inherent high propellant cost of rendezvous and launch vehicle limitations. Chiron was previously targeted in a 2010 Decadal Study and was given special consideration as a target. Ultimately, this study aims to identify Centaurs that allow for robust mission performance and to develop a design reference mission for a single Centaur target for detailed mission systems evaluation.

A series of mission design rules and assumptions were enforced to constrain the study as shown in Table 1-3. One of the key driving guidelines is that the mission launch between January 2036 and December 2040 to fit within a New Frontiers 6 funding timeline. Given the large orbits of Centaurs and the likely need to power the spacecraft with RTGs, the mission duration, including proximity and landing operations, was limited to 14 years given study guidelines on RTG lifetime. Also of note, the Falcon Heavy Expendable launch vehicle is considered along with the launch vehicle performance curves in the study guidelines as it is significantly more capable than the Falcon Heavy Reusable and can be enabling for high-ΔV missions.

1.3.1 Mission Design Trades

A broad mission design trade is conducted in order to both determine a design reference mission for detailed evaluation as well as to characterize the target and mission architecture design space. Over 550 Centaur targets are evaluated based on 'Centaur' or 'Chiron-type comet' designations in the JPL Small-Body Database and science team input. The Centaurs of highest interest provided by the science team have a range of features (e.g., activity, rings, or a satellite) as listed in Table 1-4.

Table 1-3: Mission Design Assumptions

Parameter	Values	Notes
Launch date range	January 1, 2036 – December 31, 2040	Assumed New Frontiers 6 target range
Maximum mission duration	14 years	Limited by RTG life
Minimum proximity operations duration	1 year	Limits interplanetary transfer to a maximum of 13 years
Launch vehicle	Study launch curves + Falcon Heavy Expendable (FHE)	FHE performance based on NLSII, launch declination: -28.5 – 28.5 deg
Electric propulsion duty cycle	90%	
Electric propulsion power margin	10%	Power margin applied to power available for electric propulsion
Max radioisotope electric propulsion power	1.2 kW (beginning of life)	Next Gen RTG assumed at 400 W per unit; 1.9% decay rate; max. of only 2 RTGs preferred
Max solar electric propulsion power	40 kW (end of life)	Flexibility to trade if needed
Bus power during thrusting	100 W	Power not available to EP system for thrusting
Propellant Margin	10%	For both EP and chemical propellant
Electric propulsion thrusters considered	NEXT-C, XIPS-25	

Forced coasts if EP thrusting	60 days after launch, 30 days before flybys, 30 days before Centaur arrival	
Isp for interplanetary chemical maneuvers	320 s	Bi-propellant system assumed
Minimum propellant for proximity operations	300 kg	
Minimum Earth flyby altitude	1000 km if REP, 300 km if SEP or chemical	

Table 1-4: Top-Interest Centaurs from Science Team

Name	Orbit Condition Code	JPL SBDB designation	Interest
Chiron	0	Centaur	Activity, rings
Chariklo	1	Centaur	Rings
Bienor	0	Centaur	Rings?
Ceto	1	TransNeptunian Object	Binary
Typhon	1	TransNeptunian Object	Binary
Echeclus	0	Centaur	Activity
29P/Schwassmann-Wachmann	0	Jupiter-family Comet	Activity
P/2019 LD2	3	Jupiter-family Comet	Activity
2014 OG392	3	Chiron-type Comet	Activity
39P/Oterma	0	Chiron-type Comet	Activity
165P/Linear	4	Chiron-type Comet	Activity
166P/2001 T4	2	Chiron-type Comet	Activity
167P/2004 PY42	3	Chiron-type Comet	Activity
C/2001 M10	3	Jupiter-family Comet	Activity
P/2004 A1	3	Jupiter-family Comet	Activity
2003 QD112	4	Centaur	Activity
P/2005 T3	6	Chiron-type Comet	Activity
P/2005 S2	5	Chiron-type Comet	Activity
2006 SX368	2	Centaur	Activity

In addition to target Centaur, a variety of mission architecture options are evaluated using different propulsion options. Namely, traditional chemical bi-propellant propulsion, radioisotope electric propulsion (REP) using RTGs, solar electric propulsion (SEP), as well hybrid options with chemical propulsion for the arrival maneuver and electric propulsion for the interplanetary transfer are traded. In all architectures, staging components of the propulsion system are considered to maximize delivered mass. For REP missions, one to three RTGs are considered, with a preference for one or two units given the high cost of RTGs. The Next Gen RTG, the only RTG option considered as it is the highest performing allowed in the study rules, provides 400 W per RTG. For SEP missions, solar arrays providing between 10 and 40 kW at one AU in 10 kW increments are traded. Both NASA's Evolutionary Xenon Thruster (NEXT) and XIPS-25 thrusters are considered with two to four active thrusters. NEXT thrusters are notably efficient at high powers, but have a high minimum input power, making the low input power of XIPS-25 potentially advantageous for some architectures.

Numerous gravity-assist sequences are also traded to ensure capable trajectory performance. Sequences evaluated include: E, J, EE, EJ, EEJ, VJ, VEJ, VEEJ, MJ, MEJ, VEMJ, ES, EES, MS, EMS, where E indicates an Earth flyby, V for Venus, J for Jupiter, M for Mars, and S for Saturn. A Jupiter or Saturn flyby can significantly improve performance given the relatively high inclination of Centaurs. Earth flybys are advantageous to efficiently align the trajectory for a gas giant gravity assist.

The approach applied to identify high-performing Centaur targets for the mission is illustrated in Figure 1-6. To initially identify viable missions the target Centaurs are pruned according to key orbital elements in Step 1. To first reduce the number of Centaurs for in-depth design, bodies for further consideration were filtered based on a perihelion distance of less than 10 AU during the period from 2041 to 2053 given an assumed flight time duration of five to thirteen years and a target launch date between 2036 and 2040. That list of bodies was further reduced by enforcing a maximum inclination of 60 degrees given the ΔV cost associated with high inclination targets as depicted in Figure 1-7 (note that some Centaurs have a retrograde orbit). With this reduced target set, automated scans for REP+chemical and chemical-only trajectories are conducted in GSFC's EMTG tool and a Lambert grid search for ballistic trajectories with a variety of gravity assist sequences in Step 2. SEP-based propulsion was not applied in the initial target performance scan as it is assumed to be the highest cost mission given the need for both solar arrays and an RTG for power. In Step 3, five different mission architectures, including SEP-based options, and gravity assist sequences are then thoroughly evaluated at medium fidelity in EMTG for the top-performing targets from the broad search as well as the full list of Centaurs of highest science interest (Table 1-4) to identify the final set of mission options.

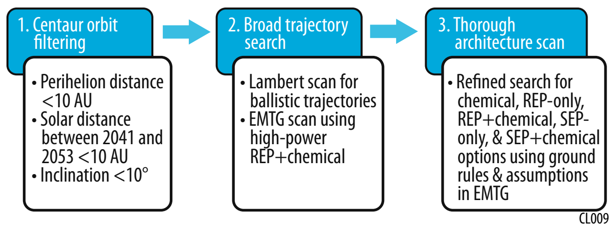

Figure 1-6: Centaur target search approach

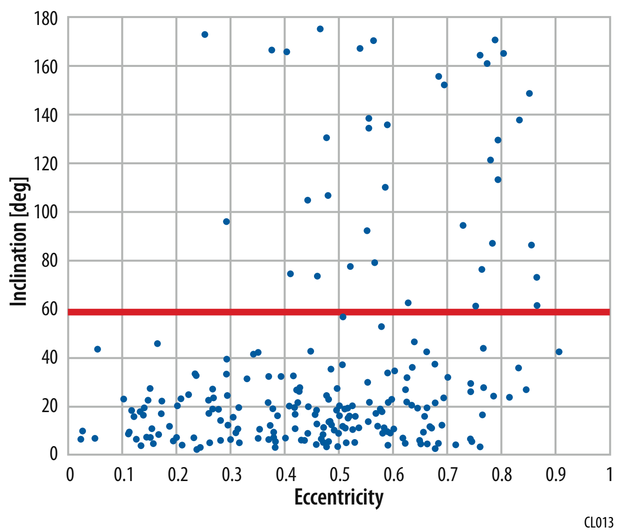

Figure 1-7: Inclination filtering of Centaur targets

Table 1-5 outlines the top 15 targets in terms of the highest delivered mass at Centaur arrival from all architectures considered. In comparing the performance across architectures, REP-only solutions are not able to deliver as much mass as chemical architectures with only 2 Next Gen RTGs available for both EP and non-EP power. The only Centaur identified with over 2000 kg delivered mass using REP-only propulsion is 2010 NK83. Alternatively, while SEP-only architectures can exploit high available power in the early part of the mission for efficient orbit raising, the increase in solar distances in the latter part of the trajectory and resulting decrease in power available and thrust authority is often too severe to effectively rendezvous. While not abundant, high-performing SEP-only missions to Centaurs with low eccentricities and low aphelion distances are possible. A SEP-only solution to 2015

BQ311 can deliver roughly 5500 kg with a 30 kW end-of-life solar array, but no other Centaurs with a SEP-only solution with a 30 kW array are identified for the launch dates considered in the study. No trajectories that could deliver over 2000 kg to the highest-interest Centaur targets from Table 1-4 are identified.

Table 1-5: Top Performing Centaur Targets (13-year flight time, launch between 2036 and 2040)

	Preliminary Delivered Mass [kg]					aphelion distance [AU]	Science Rank	
Centaur	Chemical	REP +chem	REP only	SEP +chem	Best			Science Notes
2015 BQ311	8905	3404	700	4200	8905	9.19	3	colors (blue)
2004 RW141	4483	1707	833	6372	6372	10.41		small
2008 SJ236	3654	978	396	3990	3990	15.67	2	colors (red), relatively large, ~high albedo
2020 OD8	3446	1836	1276	2767	3446	10.70		small
2016 EX	3140	1992	1628	3963	3963	11.10	4	no colors
2005 TS100	3124	1655	1133	3191	3191	6.50		
2017 UV43	2617	1660	1530	3588	3588	8.49		
2010 NK83	2048	520	2026	1758	2048	9.33		
1998 SG35	2008	1088	965	1175	2008	10.91	1	most information, "large" object
39P/Oterma	1981	1192	527	596	1981	8.97		
Chiron	1832	NF	NF	407	1832	18.87		
2019 LD2	1372	560	NF	2847	2847	6.01		
2010 WZ71	NF	751	1861	2402	2402	8.73		
2015 DB198	1517	1558	1446	1991	1991	11.72		
2000 VU2	866	1648	1123	1736	1736	10.68		

Hybrid EP-chemical architectures can be advantageous for Centaur rendezvous missions. Efficient EP thrusting at high Isp is exploited when there is sufficient power and time for orbit shaping. Then, when high thrust is necessary and power for the EP system is lacking, such as for the rendezvous arrival maneuver, the chemical system provides the necessary ΔV. When the EP system is no longer needed it can be jettisoned prior to Centaur rendezvous. While some of the best-performing solutions were combined EP-chemical architectures in terms of delivered mass, there is the potential for increased spacecraft complexity and dollar cost for these hybrid propulsion designs.

With a capable launch vehicle, chemical-only architectures to high-interest Centaurs are feasible and can provide the lowest cost mission option. A high-performing launch vehicle enables high-ΔV arrival maneuvers as the necessary propellant load can be carried for the rendezvous maneuver(s), while still delivering a high payload mass necessary for a lander mission. Chemical-only architectures can offer some simplicity compared to EP-based designs along with associated dollar cost savings. Eleven Centaurs with a delivered mass greater than 2000 kg are identified when launched on a Falcon Heavy Expendable, and three candidate Centaurs are selected for detailed analysis given high science interest to determine the best fit for a design reference mission: 1) 1998 SG35, 2) 2008 SJ236, and 3), 2015 BQ311. Additionally, while not a top-performing target for the launch dates considered, Chiron options are discussed for a broader range of launch dates and mission durations for informational purposes.

1998 SG35 Option

With high science interest and the potential for nearly 2000 kg of delivered mass, 1998 SG35, Okyrhoe, is examined in more detail. The optimal chemical-only trajectory launches in February 2039 on a Falcon Heavy

Expendable and employs an Earth and Jupiter gravity assist to reduce ΔV as illustrated in Figure 1-8. There are two large deep space maneuvers (DSM), one at aphelion before the Earth gravity assist and one in between the Jupiter gravity assist and Centaur arrival 13 years after launch. The final rendezvous sequence of maneuvers is approximately 2.3 km/s. The DSMs and rendezvous maneuver require up to 7050 kg of propellant given a 21-day launch period and would necessitate a massive propulsion stage. The design challenges for a propulsion stage of that size would likely push the mission outside of a New Frontiers cap. Moreover, the 21-day launch period delivered mass of 1945 kg does not provide any mass margin against the maximum expected dry mass with 300 kg allocated for proximity operations.

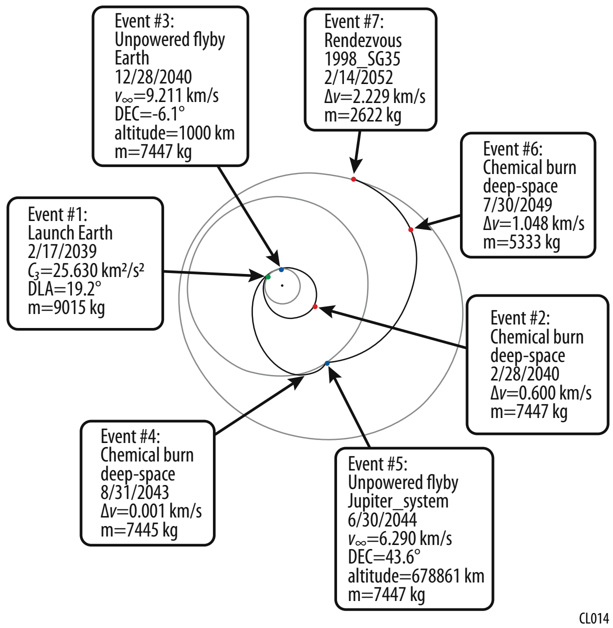

Figure 1-8: 1998 SG35 optimal trajectory

2008 SJ236 Option

As an alternative, 2008 SJ236 is also considered given its high science value and the potential for a high delivered mass with a chemical-only architecture. Over 3200 kg of delivered mass can be achieved to 2008 SJ236 with a launch in late 2040 to early 2041, allowing for substantial dry mass margin. Earth and Jupiter flybys are also advantageous for this Centaur with a ~560 m/s DSM roughly one year before the Earth flyby as depicted in Figure 1-9. As with 1998 SG35, the arrival maneuver is large, and the resulting propellant required is roughly 5100 kg for the interplanetary transfer. A design with a propulsion stage can close for 2008 SJ236, but ideally the design reference would not require the additional complications associated with a propulsion stage.

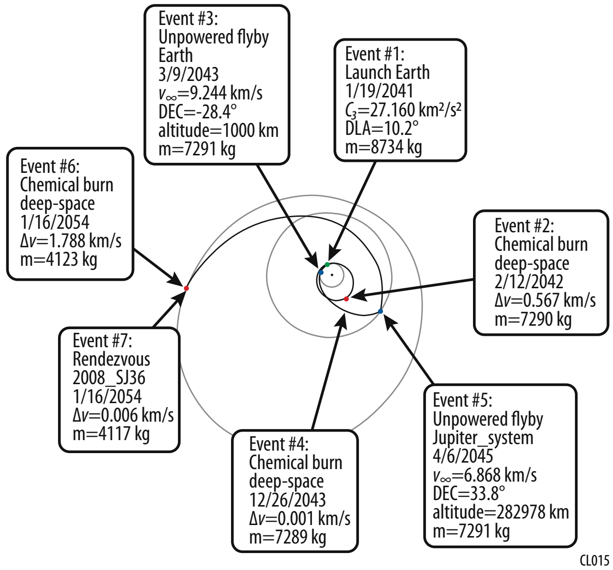

Figure 1-9: 2008 SJ236 optimal trajectory

2015 BQ311 Options

2015 BQ311 is both a compelling science target and offers the potential for low-ΔV trajectories. The body is in a small orbit for a Centaur, and a Jupiter gravity assist can provide the needed 24-degree inclination change to reach the Centaur with relatively low ΔV demands. Given the maximum interplanetary transfer time of 13 years, up to 8900 kg of mass can be delivered to the Centaur with only 1400 kg of propellant. With this exceptional performance, an extensive interplanetary cruise duration and propellant trade is conducted over the span of study launch dates for a range of flyby sequences with a FHE launch. Delivered mass performance as a function of maximum constrained propellant and maximum interplanetary cruise for each gravity assist sequence is show in Figure 1-10. The top performing solutions considering all gravity-assist sequences are combined in the lower, right-hand map that is outlined in gray. For the lower interplanetary cruise durations, the J, EJ, and EEJ flybys perform best, while the VEEJ flyby sequence is optimal for longer flight times.

Selecting a shorter interplanetary cruise duration than 13 years and allocating more mission time to Centaur proximity operations and landing operations is advantageous. 2015 BQ311 allows for 3315 kg of delivered mass with 9-year interplanetary cruise duration, EJ flyby, and 1000 kg of propellant allocated to the interplanetary transfer. With a 1400 kg tank, 400 kg can be allocated to proximity operations, while maintaining over 30% dry mass margin given a maximum expected dry mass of 1698 kg.

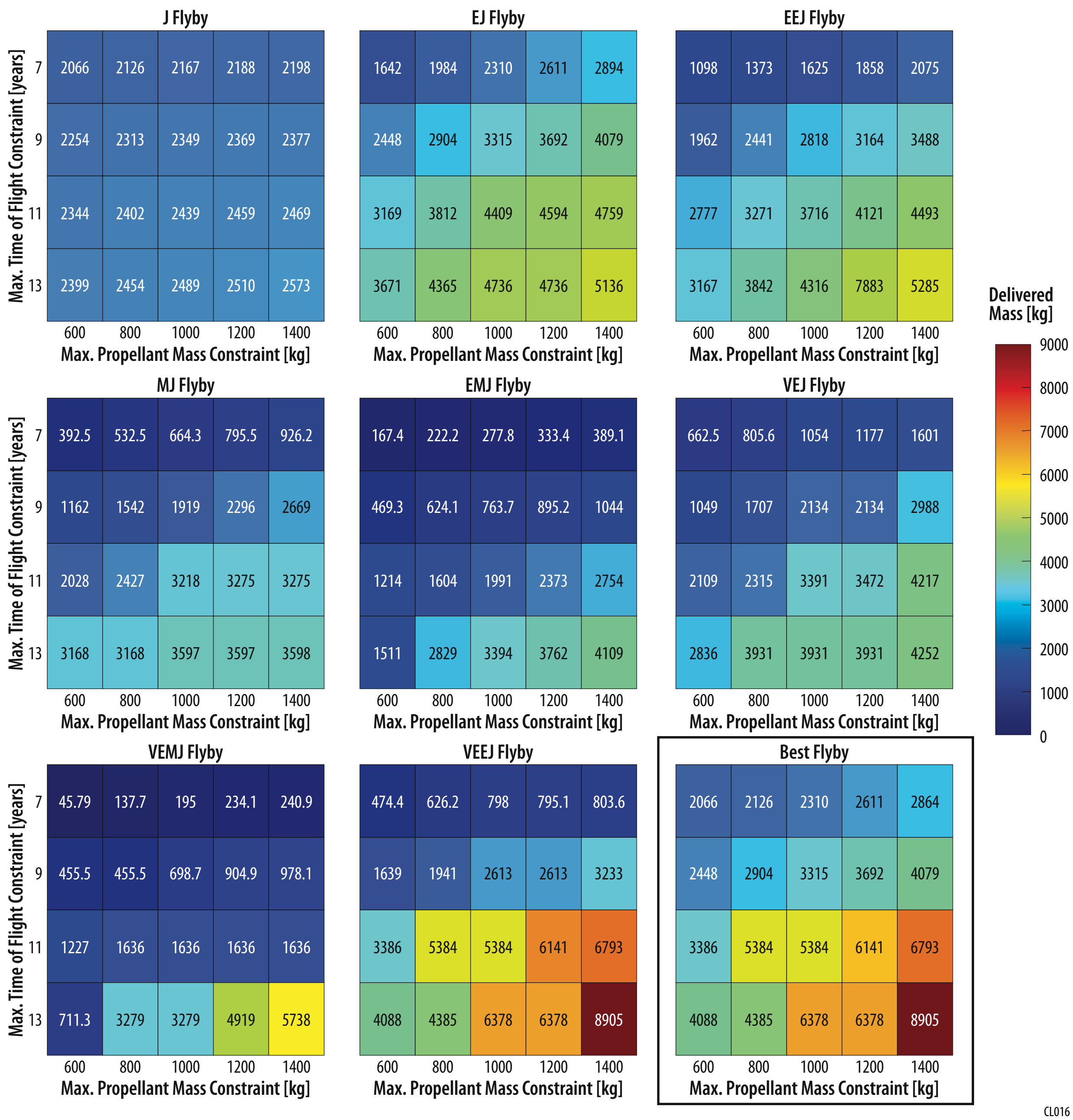

Figure 1-10: 2015 BQ311 delivered mass at Centaur arrival as a function of maximum interplanetary cruise duration constraint, maximum propellant constraint, and gravity assist sequence

2015 BQ311 also allows for ample launch opportunities throughout the 2036 to 2040 launch range as illustrated in Figures 1-11 and 1-12. Figure 1-11 shows the delivered wet mass at Centaur arrival versus launch date for the flyby sequences considered with color indicating the interplanetary cruise duration. Figure 1-12 combines the results for all flybys, illustrating nearly continuous opportunities to launch through the middle of 2039, while carrying over 30% dry mass margin over MEV if up to 13-year interplanetary cruise durations are selected.

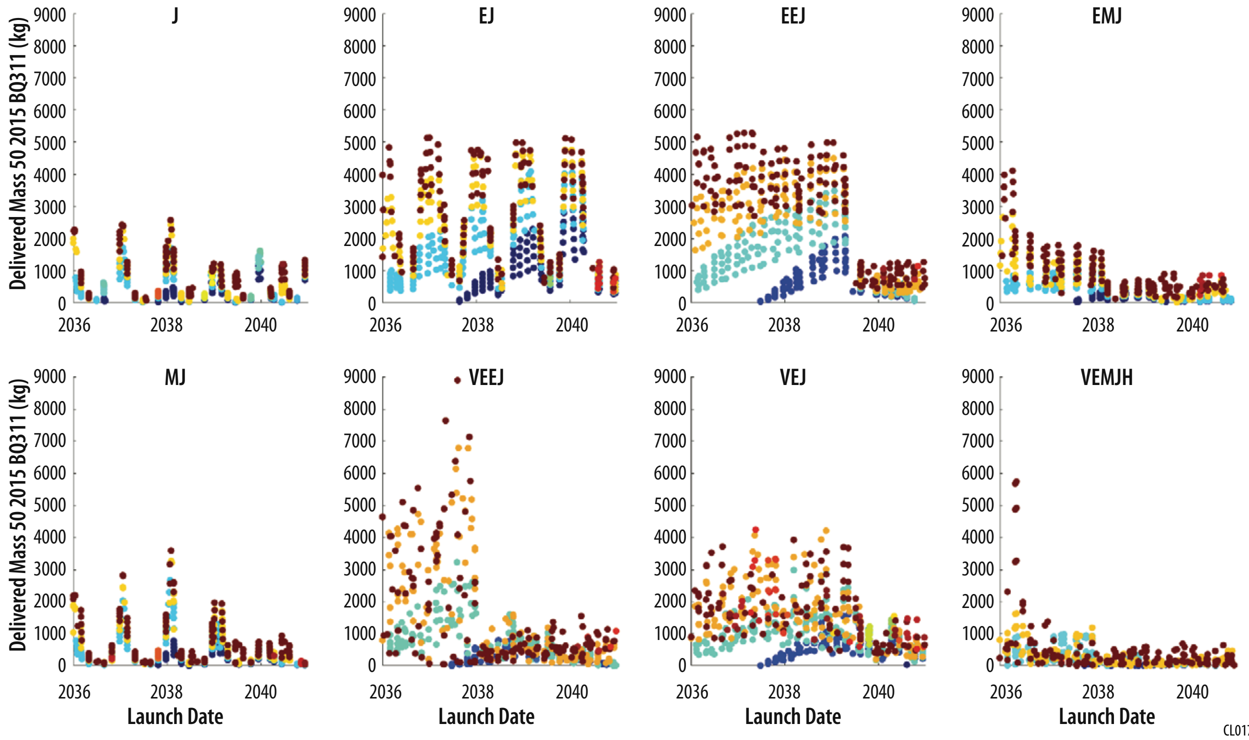

Figure 1-11: 2015 BQ311 delivered mass at Centaur arrival as a function of launch date, and maximum interplanetary cruise duration constraint for each gravity assist sequence considered

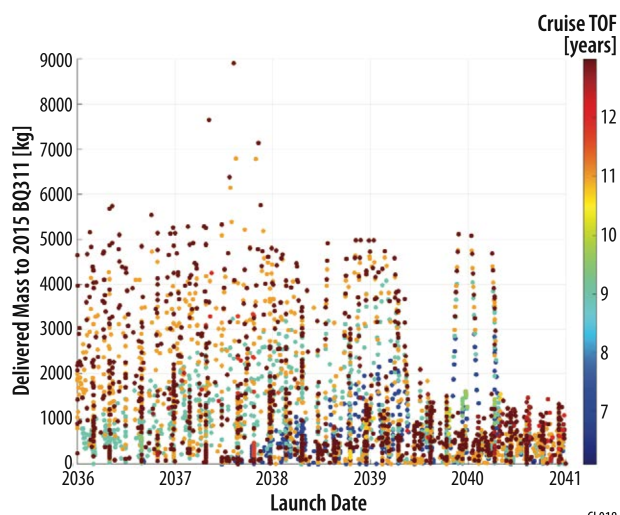

Figure 1-12: 2015 BQ311 delivered mass at Centaur arrival as a function of launch date, and maximum interplanetary cruise duration constraint for all gravity-assist sequences

Trajectory options to 2015 BQ311 with launch dates between 2036 and 2040 also offer the ability to readily trade interplanetary cruise duration and delivered mass as the Jupiter flyby can place the spacecraft on a time-tunable, catch-up trajectory that only requires a relatively low-ΔV rendezvous maneuver. Relatively short catch-up times from Jupiter to the Centaur are possible with often acceptable increases the ΔV of the final rendezvous maneuver as shown in Figure 1-13, which depicts the maximum allowable dry mass versus interplanetary cruise duration. In this trade, the maximum propellant load available for maneuvering is 1400 kg, resulting in either

1000 kg or 1100 kg for the interplanetary transfer given 300 and 400 kg proximity operations propellant allocations. Given such robust mission performance with chemical propulsion, no need for a propulsion stage, and a short interplanetary cruise duration 2015 BQ311 is selected as the target for the design reference mission.

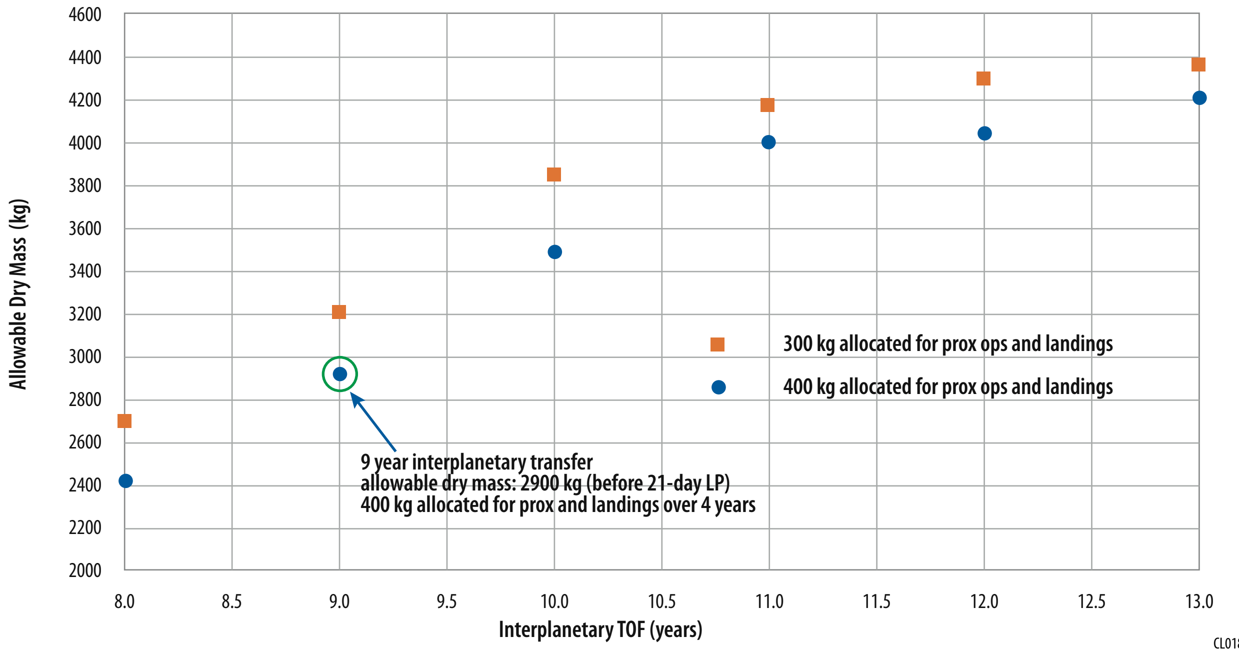

Figure 1-13: 2015 BQ311 dry mass performance versus cruise interplanetary cruise duration with different propellant allocations for proximity operations and landings

In case New Frontiers funding is better aligned with earlier launch dates, launches to 2015 BQ311 between 2030 and 2035 also examined. The phasing between Jupiter and 2015 BQ311 is such that earlier launch dates for shorter cruise flight times do not perform as well as the baseline 2036 to 2040 period. However, 2015 BQ311 is still a viable centaur target with trajectories launching in 2033, 2034, or 2035 capable of delivering at least 3000 kg after centaur rendezvous as illustrated in Figures 1-14 and 1-15. Figure 1-14 plots delivered mass to 2015 BQ311 versus launch dates between January 1, 2030 and December 31, 2035 with cruise propellant loads up to 1400 kg and color indicating cruise TOF. Alternatively, Figure 1-15 shows delivered mass to 2015 BQ311 versus launch date with cruise TOF up to 13 years and color demarking the interplanetary cruise propellant. Unlike the 2036 to 2040 evaluation period, flight times near the 13-year maximum are generally necessary to deliver at least 3000 kg to the centaur. Relatively low propellant loads are still possible, however, with several mission options in 2034 and 2035 only requiring 1000 kg or less of propellant and still delivering over 3000 kg to 2015 BQ311 as illustrated in Figure 1-15.

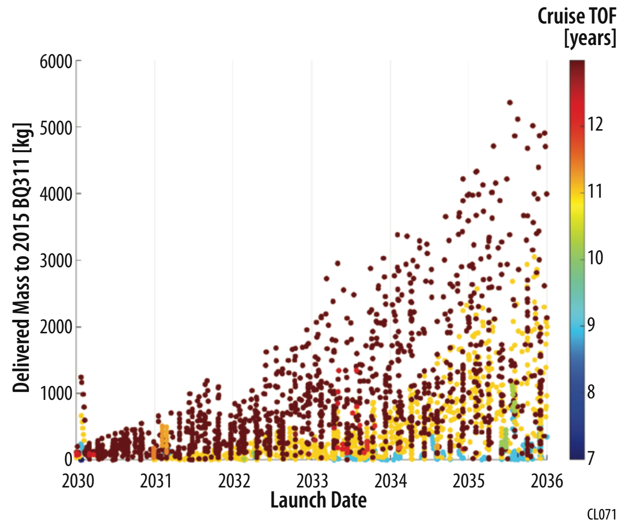

Figure 1-14: 2015 BQ311 delivered mass at centaur arrival as a function of launch dates (2030 to 2036), and cruise flight time for all gravity-assist sequences considered

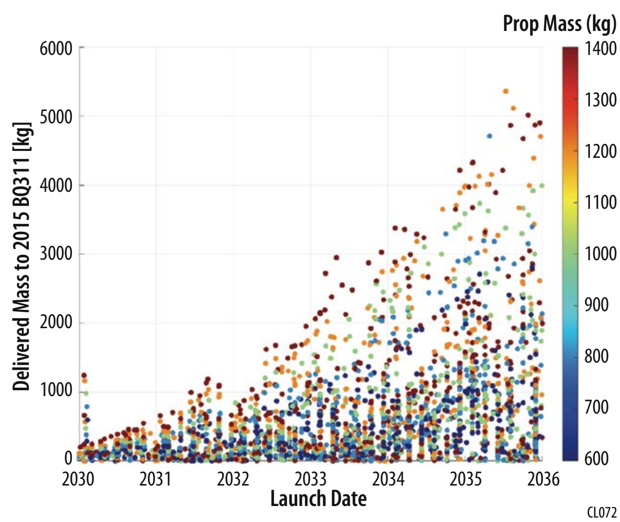

Figure 1-15: 2015 BQ311 delivered mass at centaur arrival as a function of launch dates (2030 to 2036), and maximum propellant constraint for all gravity-assist sequences considered

Chiron Performance

For the 2036 to 2040 launch range, no solutions with a delivered mass greater than 2000 kg to Chiron are identified with a interplanetary cruise duration less than 13 years as indicated in Table 1-5. However, a SEP+chemical option in 2042 with a 30-kW solar array, 13-year interplanetary cruise, and launching on Falcon Heavy Expendable can delivery approximately 2200 kg to Chiron before proximity operations. An extra year of interplanetary cruise can improve chemical mission performance to Chiron as illustrated in Figure 1-16, which shows delivered mass to Chiron versus launch date and interplanetary cruise duration for a Falcon Heavy Expendable launch for a variety of flyby sequences. Chemical missions with a 14-year interplanetary cruise can deliver up to 2630 kg using a Venus-Earth-Earth-Jupiter flyby sequence. Alternatively, the best 13-year interplanetary cruise, chemical solution through 2044 is roughly 1900 kg of delivered mass to Chiron also with a VEEJ flyby sequence. Other top-performing flyby sequences to Chiron include VEJ, EEJ, and EJ. While Chiron does not perform well enough for the launch dates, interplanetary cruise, and required mass performance for this study, later 2040 launch dates might be viable for Centaur orbiter missions or scenarios in which RTG life could be extended longer than 14 years.

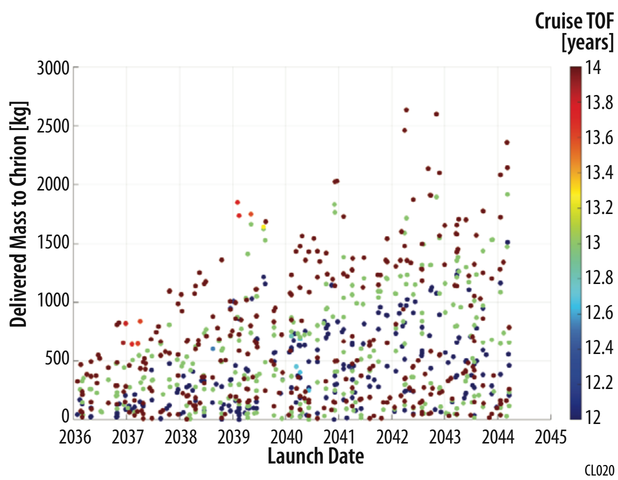

Figure 1-16: Chiron chemical mission delivered mass versus launch date using a Falcon Heavy Expendable for a variety of Venus, Earth, Mars, Jupiter, and Saturn flyby combinations

2.0 CONCEPT OF OPERATIONS

2.1 Launch and Cruise to Centaur

A trajectory to 2015 BQ311 with a nine-year interplanetary transfer time is selected as the design reference mission to balance interplanetary cruise duration and dry mass margin. A nine-year flight time allows ample time for Centaur proximity operations and landed operations. The DRM launches in January 2040 on a Falcon Heavy Expendable and utilizes an Earth and Jupiter flyby before rendezvousing with 2015 BQ311 in 2049 as depicted in Figure 2-1. The propellant required is constrained to be less than 1000 kg with margin for the interplanetary transfer and 400 kg is then allotted for proximity operations, landing, and ACS given the 1400 kg total propellant load. At Centaur arrival the solar distance is approximately 6 AU and the Earth range is roughly 5.1 AU and increasing as shown in Figure 2-2 (2015 BQ311 aphelion distance is 9.19 AU).

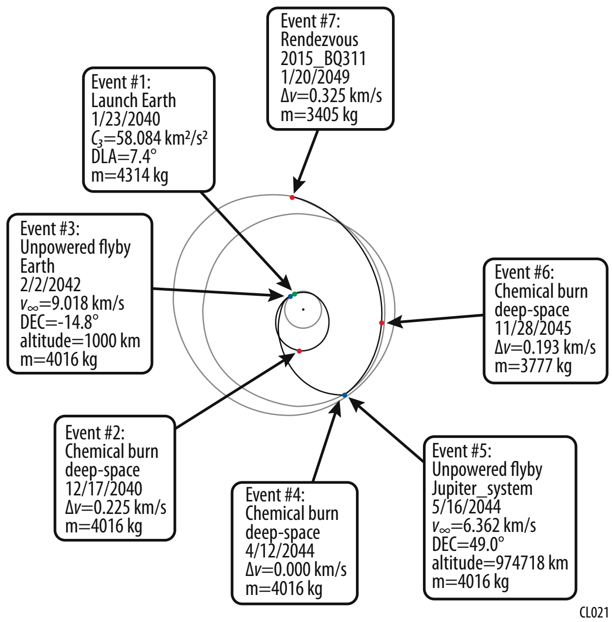

Figure 2-1: 2015 BQ311 optimal trajectory

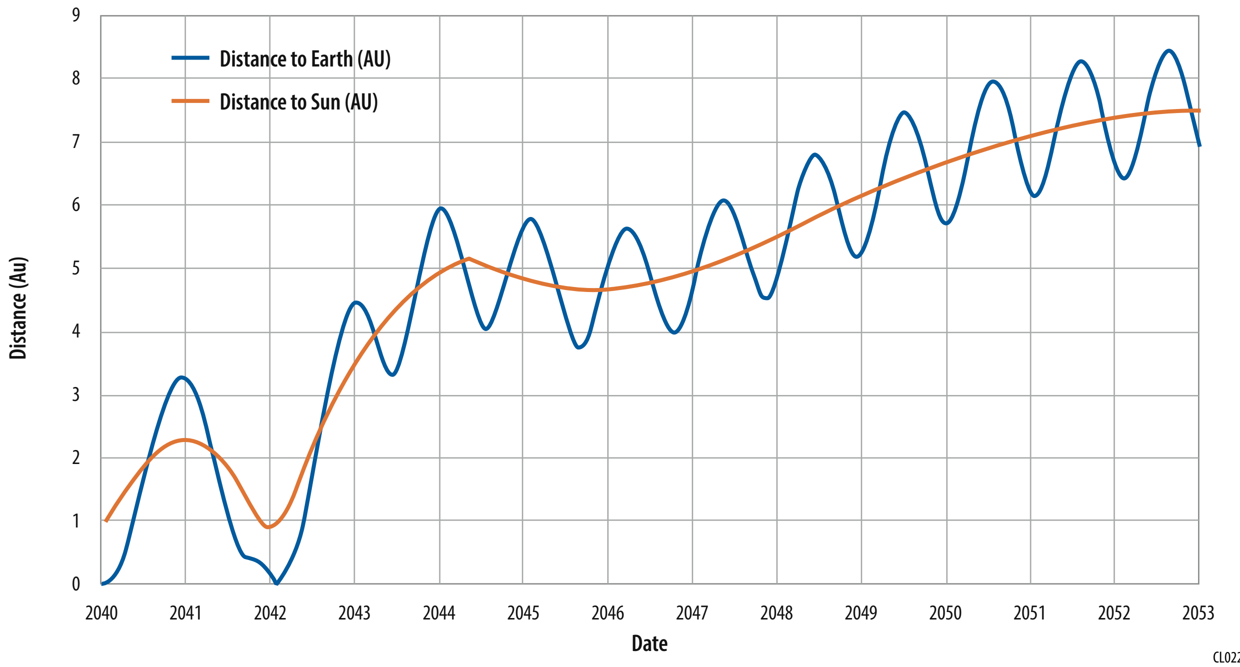

Figure 2-2: 2015 BQ311 DRM distance to Sun and Earth during interplanetary transfer

The 21-day launch period performance of the DRM to 2015 BQ311 is shown in Figure 2-3. The minimum 21-day delivered mass after the arrival rendezvous maneuver is 2599 kg and the minimum launch mass is 3599 kg. With only 1000 kg of required propellant to reach the Centaur, no expendable propulsion stage is needed, simplifying the spacecraft design. The maximum launch C3 required is 64.8 km^2/s^2, which is relatively high for a planetary mission and enabled by Falcon Heavy Expendable performance.

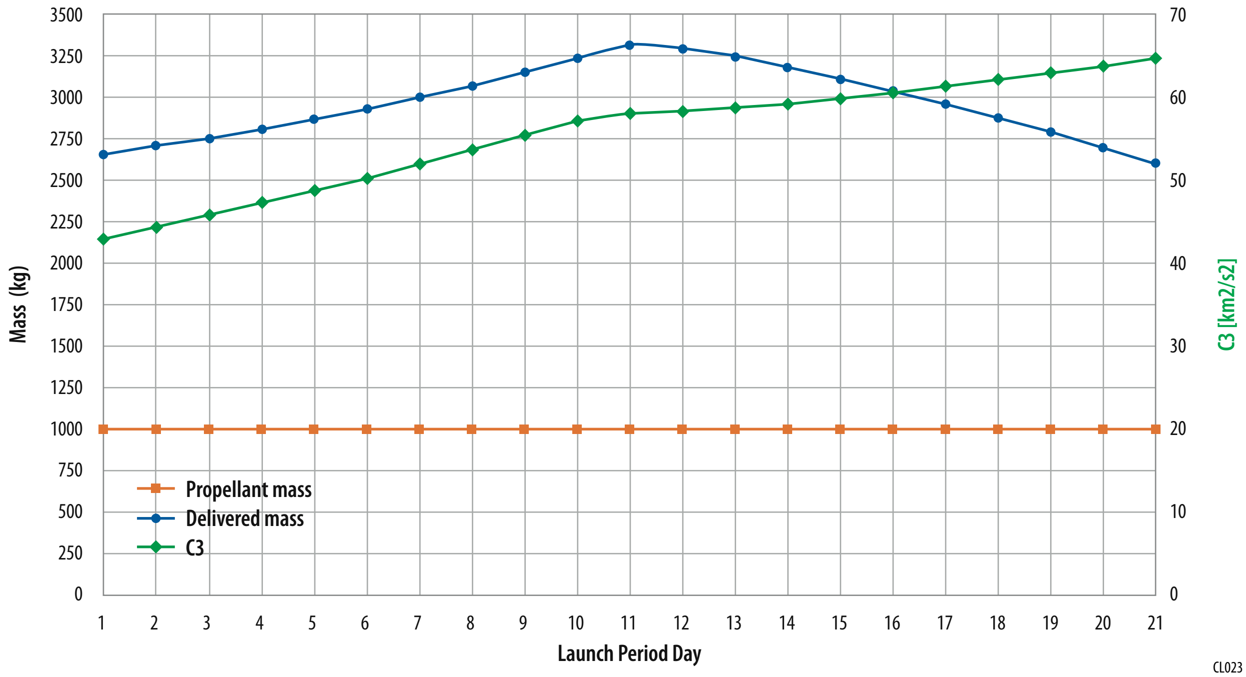

Figure 2-3: 21-day launch period performance of required propellant mass, delivered mass at Centaur arrival, and launch C3 for the 2015 BQ311 DRM (1/13/2040 launch period open date)

Table 2-1: 2015 BQ311 Design Reference Mission 21-Day Launch Period Parameters

Launch period open date	01/13/2040
Time of Flight	9 years
Min delivered mass	2599 kg
Max propellant required	1000kg
Min launch mass	3599 kg
Max C3 required	64.8 km^2/s^2
Max interplanetary ΔV	914 m/s

Given the DRM's 9-year interplanetary cruise and 1,000 kg maximum propellant for the interplanetary transfer, additional launch opportunities are also available as shown in Figure 2-4. To be considered a viable launch opportunity the delivered mass performance minus the 400 kg proximity operations propellant must allow for 30% dry mass margin over the maximum expected spacecraft dry mass. Additional launch opportunities exist in late 2038 and early 2039, late 2039, and again in April 2040.

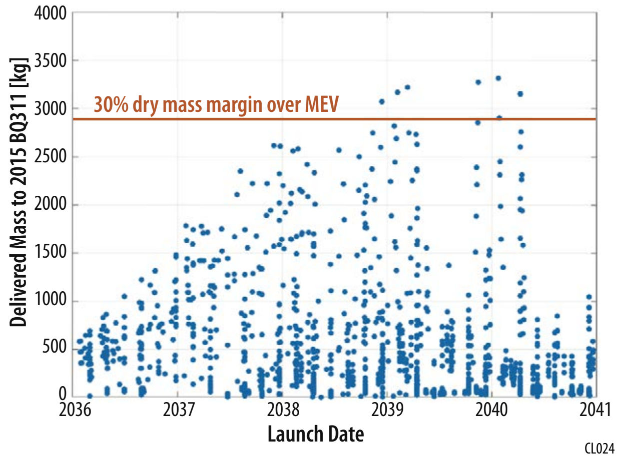

Figure 2-4: 2015 BQ311 launch opportunities with interplanetary cruise duration < 9 years and interplanetary transfer propellant < 1000 kg

The ΔV budget for the DRM is shown in Table 2-2. The maximum impulsive ΔV over the 21-day launch period is 914 m/s, which is margined by 10% to account for finite burns, maneuver execution errors, and statistical maneuvers correcting dispersions. The proximity operations ΔV is 243 m/s given the worst-case Centaur GM and a 30% margin. The ΔV budget allots 108 m/s and 30 kg of ACS propellant for contingencies and extended mission options.

Table 2-2: delta-V and Propellant Budget

Subphase	Nominal Impulsive ΔV (m/s)	Margin %	Total ΔV (m/s)	ACS Prop (kg)	Thrust Isp (s)	Pre-mnvr mass (kg)	Prop Consumed (kg)	Post-mnvr mass (kg)	Notes
Interplanetary transfer & rendezvous	914	10	1005	13	320	3599	1000	2599	High ΔV case & min. launch mass in 21-day LP
Proximity operations	187	30	243	20	305	2599	223	2376	High Centaur GM ΔV scenario
Landing	15	30	20	50	305	2376	65	2311	
Contingency	108	0	108	30	305	2311	112	2199	For prox ops contingencies & extended mission
Totals	**1224**		**1376**	**113**			**1400**		

After launch the CORAL spacecraft deploys its HGA, performs initial instrument checkout and communicates with Earth. The communication subsystem will be in receive mode at all times and will communicate with Earth upon command roughly once a month during its 9 year interplanetary cruise. Table 2-3 shows the nominal events, available power and the maximum expected value (MEV) of the average power used during the interplanetary cruise. The MEV includes a 30% contingency on the current best estimate (CBE) of the average power. The available power from the RTGs decreases by 1.9% per year. The power profile is further discussed in Section 3.6.

Table 2-3: CORAL Available Power

Date	Event	Available RTGs Power (W)	Average Power Required (W)	Power Margin (W)
1/23/2040	Launch	800.0	594.6	205.4
1/23/2040	Cruise	793.7	560.0	233.7
12/17/2040	Propulsion Burn	786.4	579.4	206.9
2/2/2042	Earth Flyby	769.5	560.0	209.5
4/12/2044	Propulsion Burn	737.8	579.4	158.4
5/16/2044	Jupiter Flyby	768.5	560.0	208.6
11/28/2045	Propulsion	746.3	560.0	186.3
1/23/2046	Cruise	744.1	560.0	184.1
1/20/2049	Rendezvous and Approach	711.1	579.4	131.7

2.2 Rendezvous and Proximity Operations

The proximity operations concept for 2015 BQ311 closely follows past small-body missions such as OSIRIS-REx. The interplanetary cruise ends and proximity operations begins with series of rendezvous maneuvers totaling roughly 350 m/s. The rendezvous is followed by an approach sequence, reducing the range to the Centaur to 250 km, and ensuring the Sun-Centaur-spacecraft phase angle is sufficiently low at the end of near-field approach. After a preliminary survey to map the Centaur gravitational parameter (GM), a mapping campaign allows for construction of digital terrain maps for science and navigation based on optical images. Additionally, the mapping subphases enables broad coverage of the Centaur with the UV and IR spectrometers. Following mapping, the spacecraft then conducts close flyby examinations of potential launch sites to collect high-resolution images before performing a series of landing dry runs. The various subphases for proximity operations and their nominal durations and ranges are outlined in Table 2-4. Four years is allocated for proximity operations and one landing with ample time margin. Given the RTG lifetime of 14 years and a nine-year interplanetary transfer, an additional one year is available for contingencies or extended mission options. This provides the option for an extended mission with a takeoff and landing at a second landing site after completion of the primary mission. The duration of the subphases is driven by the number and size of images taken during each subphase and the communication subsystem data rates to Earth.

Data Budget

Table 2-5 shows a notional data budget obtained by applying lessons learned to what was done for OSIRIS-Rex and tailoring them for CORAL needs. During all but two subphases there is a single 4 hour DSN contact needed per day. For the Mid-altitude mapping and low-altitude mapping phases a single 8 hour DSN contact per day or two 4 hour DSN contacts per day are needed. More detail on the data budget is provided in Section 2.2.

Table 2-4: Proximity Operations Subphase Descriptions

Subphase	Phase Duration (weeks)	Range to Centaur	Objective
Rendezvous maneuvers	5	2,500,000 km to 100,000 km	Reduce speed relative to Centaur, initial detection, refine approach trajectory with Centaur OD updates, initial rotation characterization
Far-field approach	8	100,000 k m to 1,000 km	Far-field approach, low-res topography map to support landmark navigation, identify Centaur activity, identify satellites, rotation characterization
Near-field approach	6	1,000 km to 200 km	Finalize rotation characterization, initial map delivery, update mapping campaign based on activity/satellites
Preliminary Survey	5	15 km flybys	Gravity mapping (obtain GM), high resolution images

High-altitude mapping	20	50 km orbits	High altitude mapping with NFOV camera with flybys, initial shape model, refine gravity mapping
Mid-altitude mapping	85	25 km orbits	Mid altitude mapping with NFOV camera to generate hazard map
Low-altitude mapping	40	10 km orbits	Low altitude mapping with NOFV and WFOV camera at different nodes, finalize gravity mapping for landing
Landing site survey	10	~2 km close passes, 10 km home orbit	Close inspection of target landing sites, determine primary landing site
Landing dry runs	10	~200 m, 100 m, 50 m waypoints	Series of rehearsals to test navigation through final checkpoint maneuver prior to descent
Landing	1	~200 m, 100 m, 50 m, surface	Safely descent to surface
Landed operations	8	Surface	Landed science operations
Total Duration	**198**		

Table 2-5: Proximity Operations Data Budget

Subphase	Phase Duration (weeks)	% of Centaur mapped	Total Required Mbytes	Required Transmit Duration (hours)	Required number of 4-hr DSN passes	Required number of 8-hr DSN passes	Required weeks assuming 1 4-hour pass per day	Required weeks assuming 1 8-hour pass per day
Rendezvous maneuvers	5	N/A	602	17	4	2	1	0
Far-field approach	8	N/A	1,355	39	10	5	1	1
Near-field approach	6	90	2,861	82	21	10	3	1
Preliminary Survey	5	N/A	1,915	55	14	7	2	1
High-altitude mapping	20	80	17,688	507	127	63	18	9
Mid-altitude mapping	85	25	81,439	2,335	584	292	83	42
Low-altitude mapping	40	5	37,691	1,081	270	42	39	19
Landing site survey	10	N/A	6,674	191	48	24	7	3
Landing dry runs	10	N/A	5,684	163	41	20	6	3
Landed operations	8	N/A	355	10	3	1	0	0
Total Duration	**198**							

Several key assumptions dictate the proximity operations design for 2015 BQ311. There are uncertainties associated with the albedo and bulk density of 2015 BQ311, which in turn drive uncertainties on the diameter and gravitational parameter (GM) as shown in Table 2-6. The bounding minimum and maximum albedo are 0.02 and 0.353, given the lowest albedo listed for a Centaur in the JPL Small-Body Database and the highest albedo for a Centaur listed in the literature (2005 UJ438 in (Duffard et al. 2014). Bulk densities between 0.5 and 2.05 g/cm^3 are considered to conservatively encompass expected variations (Duffard et al 2014, Grundy et al 2007). The resulting bounding diameter ranges from 7.9 to 33.2 km and the bounding GM ranges from 8.61E-6 to 2.62E-3 km3/s2. The associated sphere of influence, Hill's sphere, and maximum stable orbit given a solar distance between 5.06 and 6.18 AU are also listed in Table 2-6. Given the distance from the sun and resulting low solar radiation pressure, stable orbits far from 2015 BQ311 are feasible.

A bi-propellant system is used for maneuvers during proximity operations with an Isp of 305 s, and the propellant allocated for this phase of the mission is 400 kg of the 1400 kg carried for the mission. The four-year duration for proximity operations does not include the Rendezvous maneuver sequence, which is detailed in the mission design section.

Table 2-6: 2015 BQ311 Characteristics

	Low Bounding Mass	Nominal Mass	High Bounding Mass
Albedo	0.353 (max)	0.0764	0.02 (min)
Bulk density (g/cm^3)	0.5 (min)	1.0	2.05 (max)
Mass (kg)	1.29E14	2.56E15	3.92E16
GM (km^3/s^2)	8.616E-6	1.711E-4	2.619E-3
Diameter (km)	7.9	17.0	33.2
Gravitational Sphere of Influence Radius (km)	[251, 611]	[832, 2021]	[2479, 6019]
Hill Sphere Radius (km)	[2095, 5087]	[5675, 13778]	[14091, 34209]
Max. Stable Orbit Semi-major Axis wrt SRP (km)	[1850, 4493]	[8249, 20026]	[32272, 78349]

Naigation Instruments

A Narrow Field of View (NFOV), Medium Field of View (MFOV) and Wide Field of View (WFOV) camera are baselined for navigation purposes and provide imaging capabilities for several types of optical navigation across a wide range of flight phases ranging from millions of kilometers from the target, down to a few meters from the surface. These cameras also support non-navigation goals, such as collecting science images and supporting shape modeling and mapping efforts. The primary properties of these cameras are listed in Table 2-7; the specific values are derived from the camera suite onboard OSIRIS-REx for similar purposes. Additionally, a LIDAR and laser altimeter provide range measurements for navigation during the landing dry runs and landing.

Table 2-7: Camera Properties derived from OSIRIS-REx camera suite

	NFOV	MFOV	WFOV
Field of View (Deg)	0.8	4	44
Detector Size (pixels)	1024 x 1024	1024 x 1024	2592 x 2592
Focal Length (mm)	630	125	7.6
Pixel Size (um)	8.5 x 8.5	8.5 x 8.5	2.2 x 2.2
Data Size/Image (MB)	1 MB	1 MB	10 MB

The practical properties of each camera are illustrated in Figures 2-5 –- 2-7. These plots demonstrate how the suite of three cameras with varying fields of view results in a wide array of possible resolutions as a function of distance from the target.

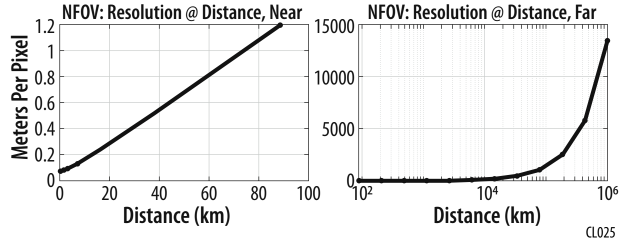

Figure 2-5: NFOV Camera Resolution for Various Distances

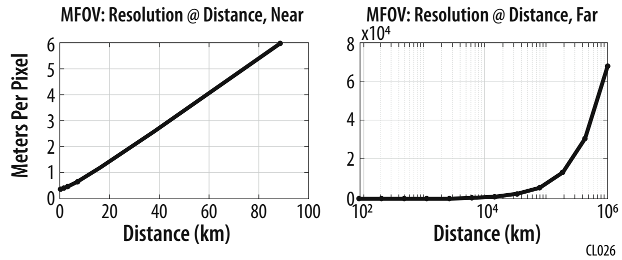

Figure 2-6: MFOV Camera Resolution for Various Distances

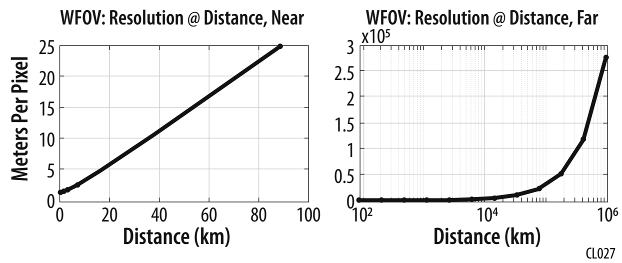

Figure 2-7: WFOV Camera Resolution for Various Distances

Navigation Baseline

The navigation details of each flight phase are summarized in Figure 2-8. The NFOV camera captures high resolution images throughout proximity operations but has a narrow footprint and many images are required to build global maps at the closer ranges. The WFOV camera's broad footprint enables more efficient mapping but at a lower resolution. All three cameras are applied for terrain relative navigation. The MFOV camera provides navigation images at distances from the Centaur surface that are suboptimal for either of the NFOV or WFOV cameras.

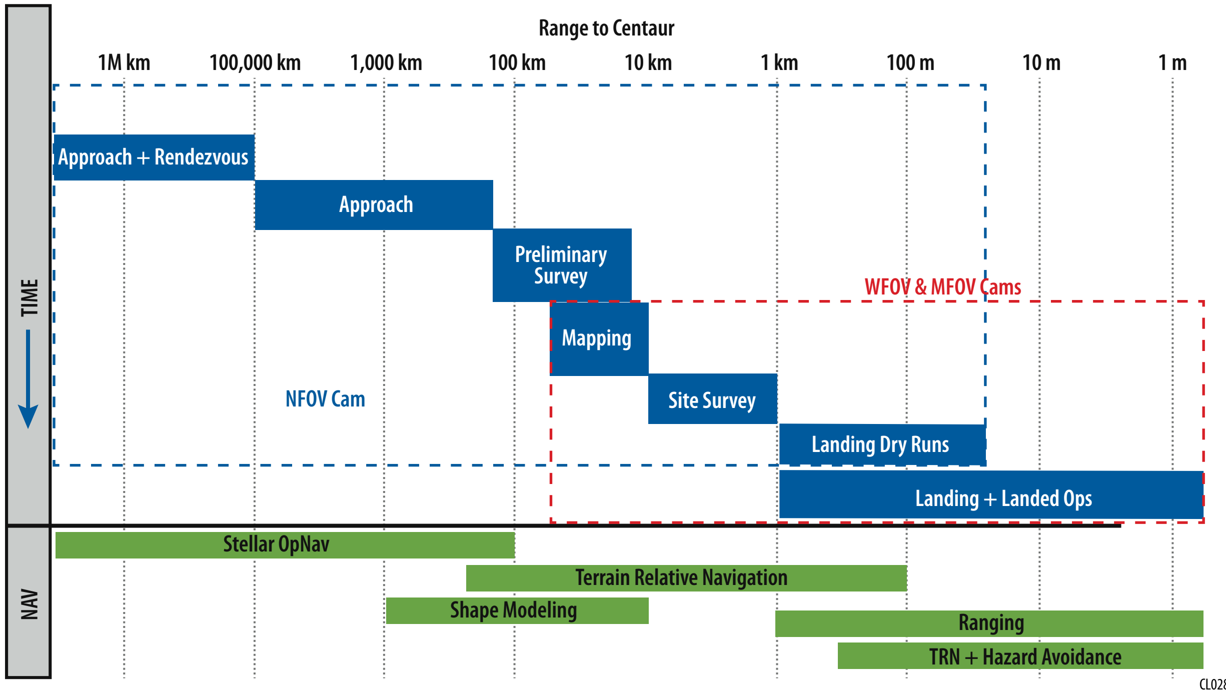

Figure 2-8: Navigation timeline

Table 2-8 provides an estimate of the number of daily images required specifically for optical navigation in order to achieve navigation performance that can support precision rendezvous and landing for sample collection. These values are dependent on the specific subphase and are largely based on those used for the OSIRIS-REx mission, which was able to achieve groundbreaking levels of navigation accuracy using a similar concept of operations. These images are downlinked daily and used to compute navigation solutions throughout the mission.

Table 2-8: Navigation images required per flight phase

Flight Phase & Range to Target	Minimum # of Images Required per Day for Navigation
Rendezvous (2.5 x 10^6 km to 1 x 10^5 km)	2
Far Field Approach (1 x 10^5 to 1 x 10^3 km)	16
Near Field Approach (1 x 10^3 to 200 km)	16
Preliminary Survey (15 km flybys)	16
High Altitude Mapping (50 km)	16
Mid Altitude Mapping (25 km)	12
Low Altitude Mapping (10 km)	12
Landing Site Survey (2 km)	12

Rendezvous Subphase

The spacecraft interplanetary trajectory will bring it to rendezvous with 2015 BQ311 in January 2049. The final two years of flight leading up to the rendezvous are shown in Figures 2-9a (perspective view) and 2-9b (ecliptic plane projection).

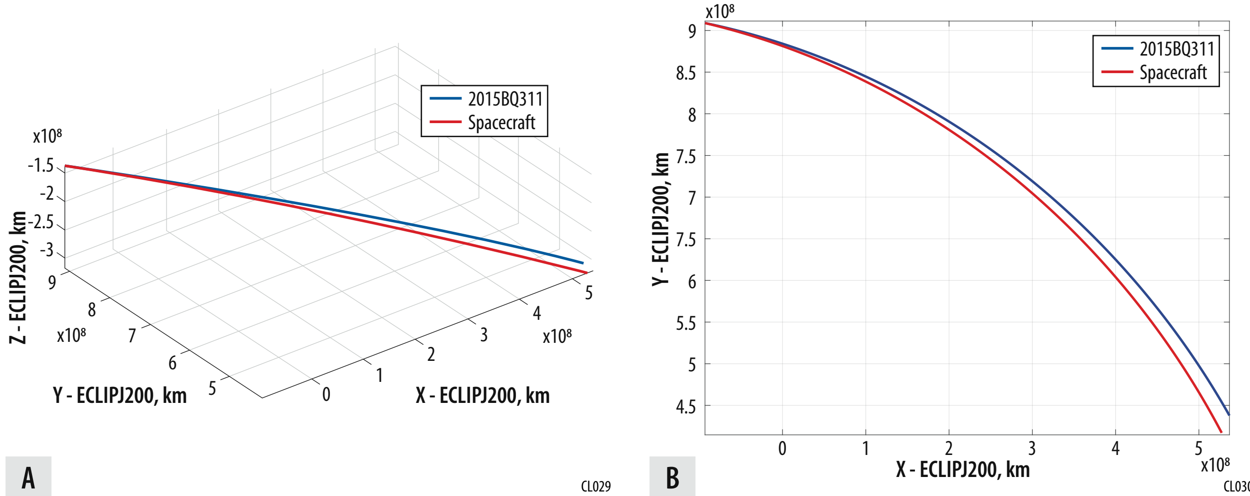

Figure 2-9: Final two years of flight leading to rendezvous. (a) Perspective view. (b) Ecliptic plane projection.

At two years before rendezvous, the spacecraft is approximately 24 million km away from 2015 BQ311, with a relative speed of approximately 490 m/s. In the absence of rendezvous maneuvers, the relative speed decreases to approximately 350 m/s by the time of rendezvous. That relative speed will be propulsively reduced to zero, thereby effecting rendezvous (matching of 2015 BQ311's position and velocity). Ultimately, this will be done with a series of three rendezvous maneuvers over time.

The solar phase angle of 2015 BQ311 from the spacecraft's perspective is approximately 43 degrees two years prior to rendezvous, and decreases to about 33 degrees by the time of rendezvous. For reference, a phase angle of 90 degrees would mean 2015 BQ311 would be approximately half lit by the sun, so a phase angle of less than 90 degrees means that more than half of 2015 BQ311 is sunlit. Thus, a phase angle between 33 and 43 degrees during approach suggests that 2015 BQ311 should be readily visible by the spacecraft's instruments during approach.

Using the notional criterion that the spacecraft camera can detect a 13th magnitude object with signal-to-noise-ratio (SNR) >=10 within a 5-second exposure (Barbee, et al 2015), 2015 BQ311 should be detectable by the spacecraft camera (i.e., reaches 13th magnitude brightness) approximately 640 days (~1.74 years) before rendezvous, when the spacecraft is still about 19.5 million km away from 2015 BQ311. This indicates that the spacecraft should have more than enough time to detect, acquire, and track 2015 BQ311 while performing terminal guidance to complete rendezvous and approach.

Approach

The approach to 2015 BQ311 is composed of Far-Field and Near-Field Approach subphases as illustrated in Figure 2-10. Far-field Approach begins after the final rendezvous maneuver and reduces the range to the Centaur to 1000 km. During Far-Field Approach a series of trajectory waypoints are traversed, decreasing the solar phase angle, and allowing for correction of trajectory dispersions. An imaging campaign with the NFOV camera is conducted to identify any satellites, detect Centaur activity, characterize 2015 BQ311's rotation. The handoff to Near-Field Approach begins at a solar phase angle near zero. Near-Field Approach takes the spacecraft from 1000 km to 250 km via waypoints that allow for observation of a variety of Centaur latitudes. Any corrections to the proximity operations plan given Centaur activity are completed during this subphase. The first global map at a low resolution is produced during approach with images from the NFOV camera and allows for terrain relative navigation in later subphases.

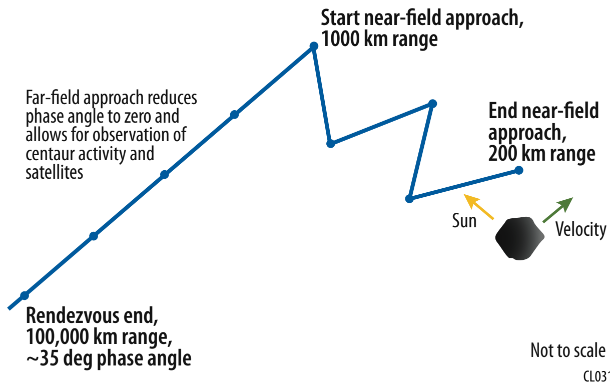

Figure 2-10: Far-Field Approach and Near-Field Approach trajectory concepts

Preliminary Survey Subphase

The primary goal of the Preliminary Survey subphase is the estimation of 2015 BQ311's GM via close, slow hyperbolic flybys of 2015 BQ311 by the spacecraft. These flybys are structured such that the incoming asymptote relative to 2015 BQ311 is perpendicular to the Earth line-of-sight (LOS), making the deflection of the spacecraft's flight path due to 2015 BQ311's gravity detectable in the Deep Space Network (DSN) radiometric tracking data for the spacecraft as illustrated in Figure 2-11. Any angle off that would result in a cosine loss in the post-flyby Doppler signature.

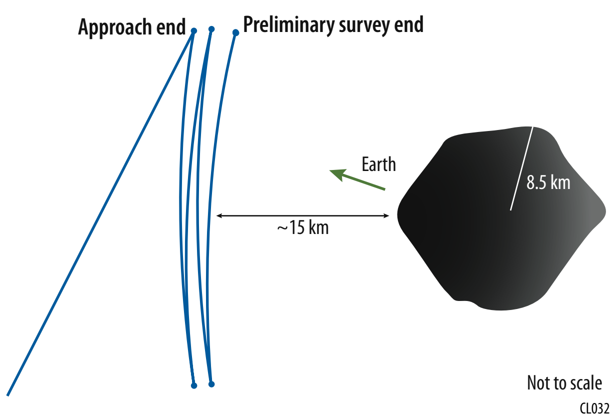

Figure 2-11: Preliminary Survey subphase trajectory concepts

In forward work, a detailed assessment of 2015 BQ311 GM performance estimation via a covariance analysis is recommended. As a preliminary assessment, calculations for the close approach distance needed relative to 2015 BQ311 during the Preliminary Survey slow hyperbolic flybys in order to achieve various levels of GM estimation accuracy are conducted. A nominal DSN Doppler uncertainty of 0.2 mHz, and perfect radial knowledge is assumed, which should be nearly attainable with adequate optical tracking. The results of these calculations for the lowest 2015 BQ311 mass estimate are shown in Figure 2-12, which indicates that a slow hyperbolic flyby within 15 km of 2015 BQ311 should allow its GM to be estimated to within at least 0.1% accuracy.

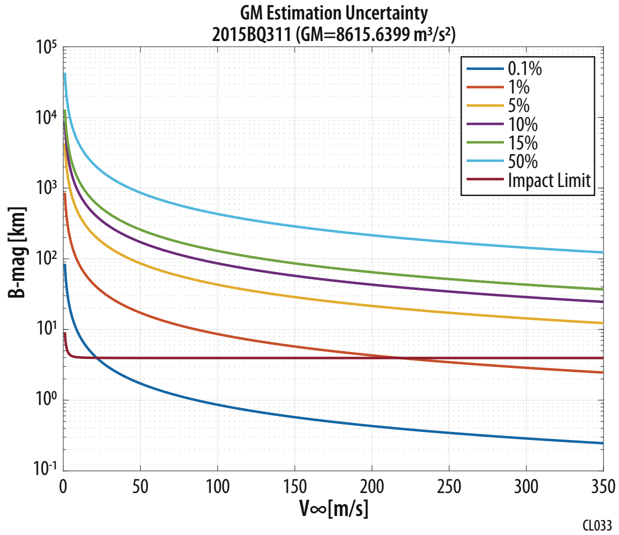

Figure 2-12: GM estimation uncertainty curves as a function of hyperbolic flyby speed and approximate periapsis radius with respect to 2015 BQ311's center of mass.

The Preliminary Survey flybys begin with the spacecraft at a distance of approximately 250 km from 2015 BQ311. The flyby speeds examined range from 1.5 to 20 m/s for the low, medium, and high Centaur mass estimates, as shown in Table 2-9. The Centaur periapsis altitude for these flybys is 15 km.

Table 2-9: Preliminary Survey hyperbolic flyby characteristics for low, medium, and high 2015 BQ311 mass cases.

GM (km^3/s^2)	V_inf (m/s)	Turning Angle (deg)	Hyp. Flight Time (hrs)	Periapsis Speed (m/s)
8.62E-06	1.5	19.35	89.23	1.78
1.71E-04	6	19.38	22.17	7.11
2.62E-03	20	19.77	6.58	23.79

Global Mapping Subphase

Global mapping is divided into three subphases that achieve increasingly higher resolution maps and science observations with orbits at decreasing ranges as depicted in Figure 2-13. The objective of High-Altitude Mapping is to allow for large-footprint science observations with the UV and IR instruments as well as the WFOV camera. Additionally, a global DTM will be constructed with NFOV camera images. This map will be used for science and navigation in the subsequent Mid-Altitude Mapping subphase. To enable images from a diversity of lighting and viewing geometries for sufficient mapping accuracy, the spacecraft will transfer between 50-kilometer altitude, near-circular orbits, passing over the Centaur's poles with varying orbital nodes to collect images at 1 m/pixel resolution. With strategic variation in illumination, emission, and azimuth angles, a roughly 2 m/pixel map of up to 80 percent of the Centaur can be generated. The selection of the orbital nodes can be adjust based on the Centaur's spin rate for optimal imaging efficiency.

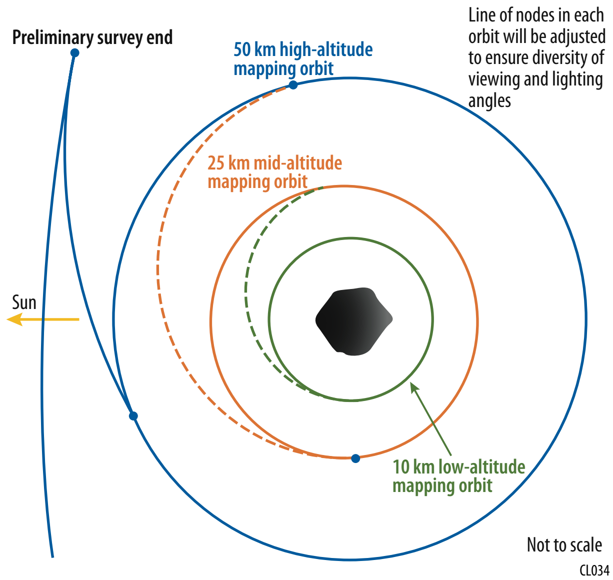

Figure 2-13: High-Altitude Mapping subphase orbit concept

Similar to High-Altitude Mapping, the orbits in Mid-Altitude Mapping have varying orbital nodes to enable viewing and lighting variation, but at a 25 km circular altitude. During Mid-Altitude Mapping, the NFOV collects images with a resolution of roughly 0.5 m/pixel and generates a 1-meter resolution map of 25% of the Centaur with particular focus on potential landing sites. Preliminary landing sites are identified during Mid-Altitude Mapping for subsequent high-resolution observation during Low-Altitude Mapping.

The altitude is reduced to 10 km for the Low-Altitude Mapping, and again the orbit plane is adjusted regularly to allow different viewing and light conditions sufficient for map construction. The low-altitude imaging campaign focuses on the potential landing sites with objective of building the high-resolution map for eventual descent and landing navigation. MFOV images are transmitted regularly to the ground for terrain relative navigation based on the DTM built in the earlier subphases.

All orbits during global mapping are stable for long periods based on long-term simulations. The orbit period and orbit speed for the different altitudes are listed in Table 2-10 for the low, nominal, and high GM values.

Table 2-10: Global Mapping orbit periods and speeds for different 2015 BQ311 GMs

Orbit Period (days); Orbit Speed (m/s) For Each Combination of Centaur GM and Orbit Altitude			
	Orbit Altitude (km)		
GM (km^3/s^2)	10	25	50
8.62E-06	1.29; 0.79	3.86; 0.55	9.82; 0.40
1.71E-04	0.44; 3.04	1.08; 2.26	2.49, 1.71
2.62E-03	0.19;9.92	0.38; 7.94	0.77; 6.27

Captured orbit simulations for the high-altitude, mid-altitude, low-altitude orbits are run for 4 years for each of the three GM cases considered for 2015 BQ311 (low, medium, high), as shown in Table 2-10. The dynamics model included solar radiation pressure (SRP) acting on the spacecraft (with the spacecraft mass set to 1500 kg, for conservatism) and point mass gravities of 2015 BQ311, the Sun, Uranus, and Jupiter. All of the those simulated orbits remained stable without any orbit maintenance maneuvers through the 4 years. Figure 2-14 presents an exemplar four-year simulation plot for the least dynamically stable of the cases, which is HAMO for the lowest GM realization of 2015 BQ311. Note that while there is some visible deviation from circularity, the orbit is stable and bounded throughout.

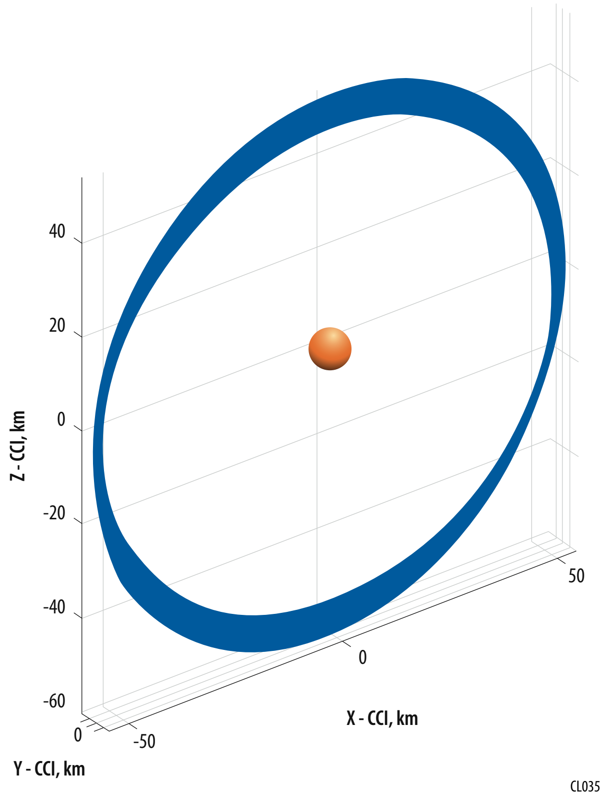

Figure 2-14: Four-year simulation of HAMO around the lowest GM realization of 2015 BQ311 considered in this work.

Landing Site Survey

Following the three mapping phases, the Landing Site Survey enables close inspection of at least three landing sites. Sufficient time margin exists for more than three site flybys. A two-kilometer altitude is baselined to enable high resolution image collection for final landing site selection as depicted in Figure 2-15. The 10-km altitude low-altitude mapping orbit serves as the home orbit between survey passes. The line of nodes of the home orbit can be adjusted for additional imaging at the 10 km altitude. Following the flybys, two final landing sites are selected.

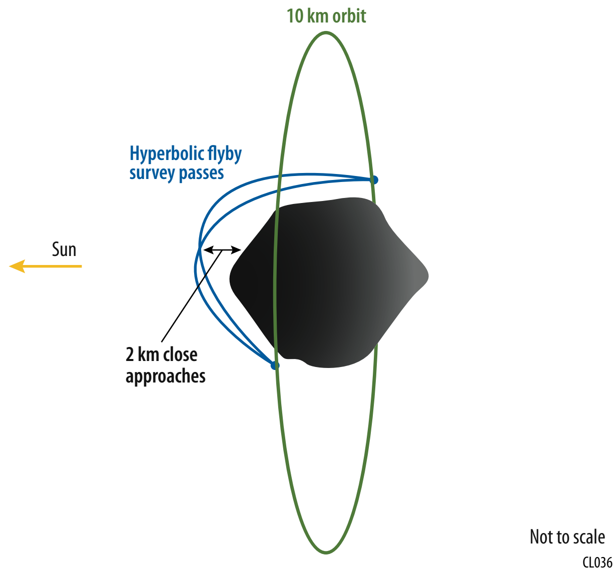

Figure 2-15: Four-year simulation of HAMO around the lowest GM realization of 2015 BQ311 considered in this work.

Proximity Operations ΔV Analysis

The spacecraft wet mass at arrival, i.e., the beginning of Preliminary Survey, will be between 2650 and 3405 kg, of which 400 kg is propellant available for proximity operations. The proximity operations thrusters have a nominal specific impulse of 305 seconds and a thrust of 22 N. The worst-case proximity operations ΔV capability for the spacecraft is, therefore 374 m/s, assuming the maximum wet mass at arrival, but without accounting for ACS propellant and any thruster canting. A maximum of 489 m/s for thrusting is available given the lowest wet mass on arrival from the 21-day launch period and without accounting for ACS propellant and thruster canting. The total ΔV required for all proximity operations maneuvers described in the preceding sections is computed for the low, medium, and high mass cases for 2015 BQ311 and found to range from 20 to 243 m/s with a 30% margin to account for finite burns and dispersions. Thus, even the worst-case ΔV capability of 375 m/s is more than sufficient for handling the worst-case ΔV budget and leaves sufficient margin for ACS propellant usage. For reference, the ΔV budget table for the medium mass 2015 BQ311 case is shown in Table 2-11.

Table 2-11: Proximity operations ΔV budget

	Low GM	Nominal GM	High GM
Initiate Slow Hyperbolic Flybys for Preliminary Survey (x3) (m/s)	4.5	18.0	60.0
Loiter in Between Preliminary Survey hyperbolic Flybys (x3) (m/s)	1.5	6.0	20.0
Capture into High-Altitude Mapping Orbit (m/s)	2.2	7.0	20.5
RAAN Shifts During High-Altitude Mapping (9 x 20 deg shifts) (m/s)	1.3	5.3	19.6
Transfer to Mid-Altitude Mapping Orbit (m/s)	0.1	0.5	1.6
RAAN Shifts During Mid-Altitude Mapping (9 x 20 deg shifts) (m/s)	1.7	7.1	24.8
Transfer to Low-Altitude Mapping Orbit (m/s)	0.2	0.8	2.0
RAAN Shifts During Low-Altitude Mapping (9 x 20 deg shifts) (m/s)	2.5	9.5	31.0
Landing Site Survey Passes @ 2 km from 10 km Home Orbit (x4) (m/s)	1.4	3.6	7.4
Total ΔV (m/s)	15.4	57.9	186.9
Total ΔV with 30% margin (m/s)	20.0	75.2	242.9

Summary

2015 BQ311 should be readily detectable by the spacecraft's onboard cameras during the last couple of years before arriving at close proximity to 2015 BQ311. 2015 BQ311's diameter, mass, and gravitational field strength are not well constrained, for lack of observational data, and this introduces uncertainty into the proximity operations design, performance, assessments, etc. However, even the lowest-mass realization of 2015 BQ311 considered in our analysis is capable of keeping a spacecraft in a stable captured orbit for at least 4 years. 2015 BQ311's GM should be able to be estimated with high accuracy (e.g., <0.1% uncertainty) via a series of slow hyperbolic flybys near 2015 BQ311 during the Preliminary Survey phase at the beginning of proximity operations. This result holds for the range of 2015 BQ311 diameter & mass considered in our work. The worst-case spacecraft ΔV capability is more than sufficient for the worst-case proximity operations ΔV budget, which corresponds to the highest-mass realization of 2015 BQ311 considered. The dynamical environment in proximity to 2015 BQ311 is not highly perturbed, i.e., solar radiation pressure and gravity from bodies other than 2015 BQ311 are much weaker than 2015 BQ311's gravity within its sphere of influence. The spacecraft should be able to navigate and maneuver efficiently and effectively throughout proximity operations.

Forward Work

The rendezvous and approach phase analyses should be updated after the rendezvous maneuver is divided up into multiple maneuvers over time. The predictions for 2015 BQ311's detectability by the spacecraft cameras should be updated based on actual spacecraft camera performance specifications. 2015 BQ311 GM estimation performance should be updated with a covariance analysis. All proximity operations phases, including captured orbits, should be re-simulated using expanded models that include 2015 BQ311 rotation, non-spherical shape model, and corresponding irregular gravity field. Detailed plans for collecting optical navigation imagery and constructing the 2015 BQ311 shape model should be developed, including the simulation of surface coverage via spacecraft instruments. Guidance, navigation, and control performance throughout proximity operations should be simulated in detail and studied with Monte Carlo analyses, including dispersions due to anticipated sources of error and uncertainty. This should include DSN radiometric tracking and optical navigation using landmark tracking. Those studies should be incorporated into parameter space studies to quantify the effects of possible variations in 2015 BQ311 size, rotation, density, shape, etc. Long round-trip light time delay with Earth (between ~1.5 and ~2.3 hours) should be properly accounted for in the proximity operations plans, which may require advances in onboard autonomy capabilities.

References

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2.3 Descent and Landing

After the Landing Site Survey has completed and initial maps are available the ground determines the primary landing site using the initial maps and plans several landing dry runs from the 10 km home orbit to refine the maps that are needed for TRN and Hazard Avoidance (HA) as shown in Figure 2-16.

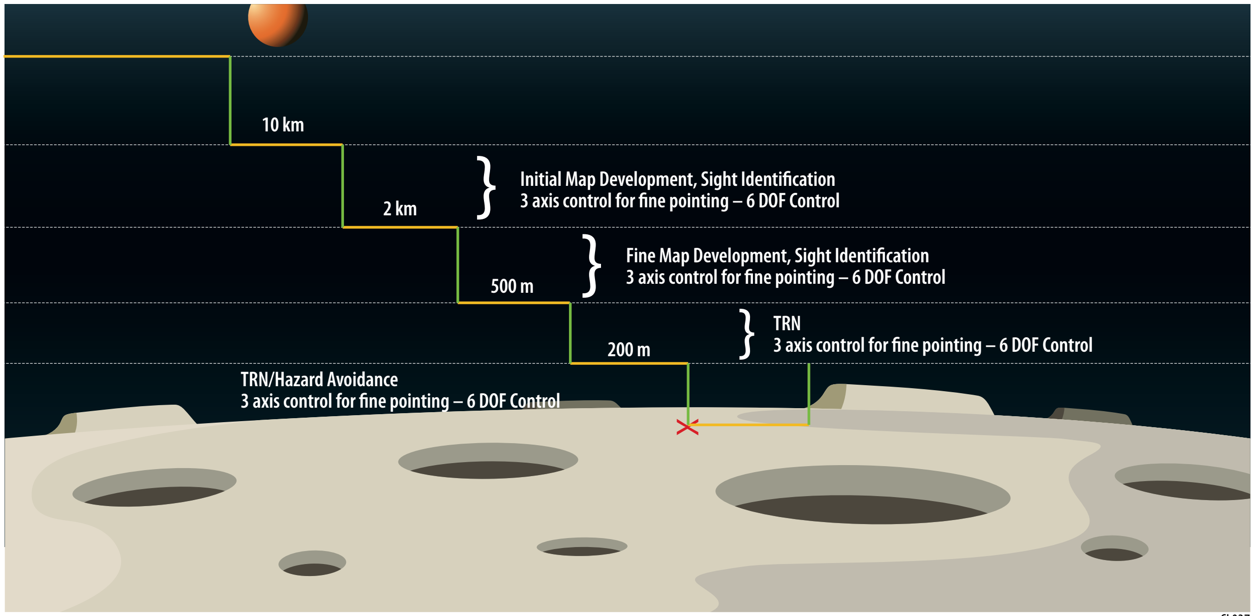

Figure 2-16: CORAL Descent and Landing Operations

Primary navigation is done by a combination of dead reckoning using star trackers, the inertial measurement unit (IMU) and Terrain Relative Navigation (Figure 2-17) utilizing optical cameras, a laser range finder to achieve a precise attitude and relative position knowledge.

The spacecraft uses reaction wheels for fine pointing along with 8 ACS thrusters to provide sufficient torque to steer the 1,600 N main engine. In order to provide the ability to horizontally translate the spacecraft for HA 16 additional ACS thrusters are used. The design is Single Fault Tolerant with 4 additional thrusters to maintain full 6-DOF control authority

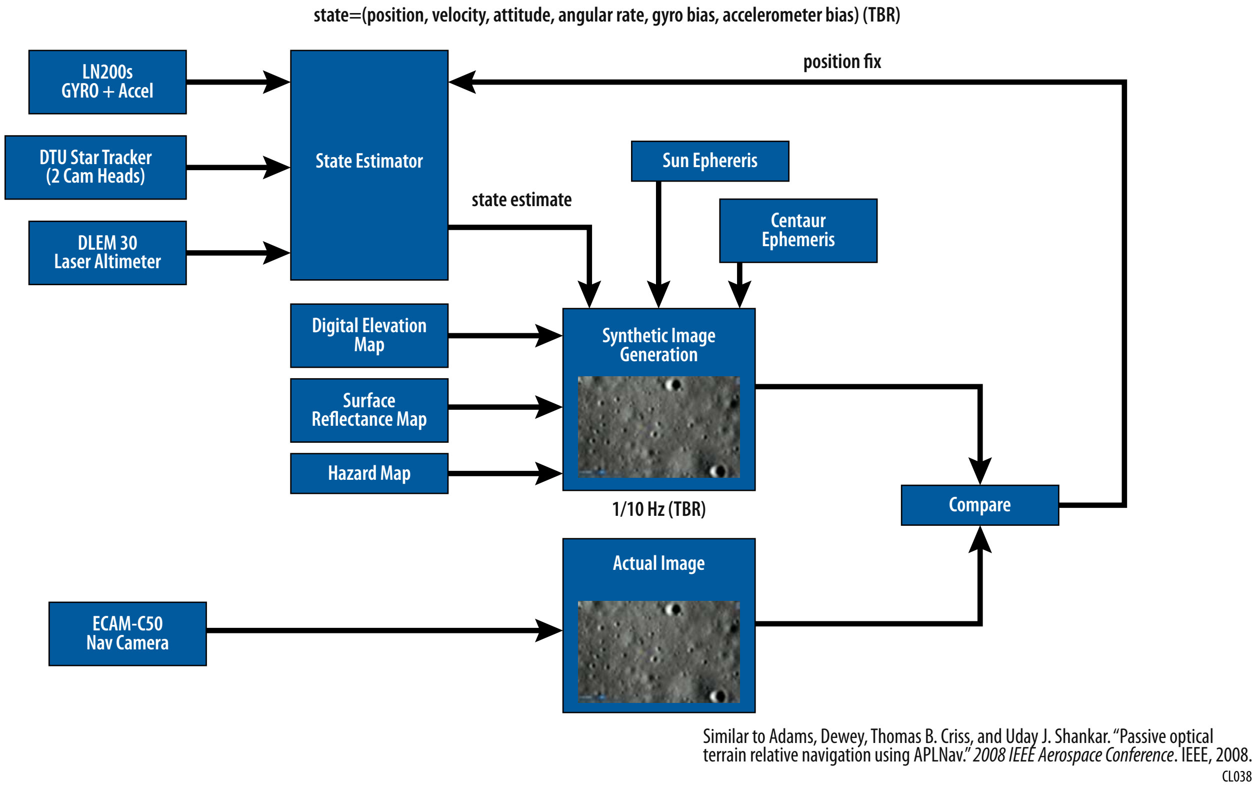

Figure 2-17: Terrain Relative Navigation Process

During landing, an optimal guidance law is used to assess the landing performance. This law is based on lunar module guidance. Figure 2-18 shows the position, velocity and force required during the landing assuming the landing sequence is initiated right above target site at three different trajectories beginning at 200, 100 and 50 meters. Each trajectory ensures full LIDAR visibility for use with the active TRN.

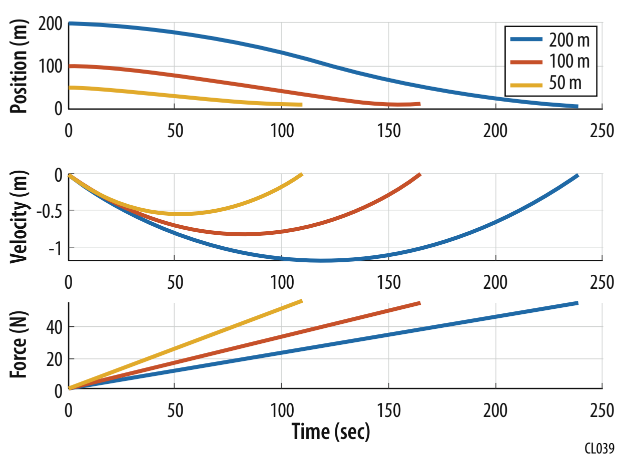

Figure 2-18: Decent Trajectory (Top) Position vs. time, (Middle) Velocity vs. time, (Bottom) Force vs. time

In additional to evaluating the nominal landing, a divert capability at 200, 100 and 50 meters is needed to ensure that CORAL can travel 50 meters downrange to avoid any hazard. Similar to Figure 2-18, Figure 2-19 shows trajectory diversion with adequate amount of control authority.

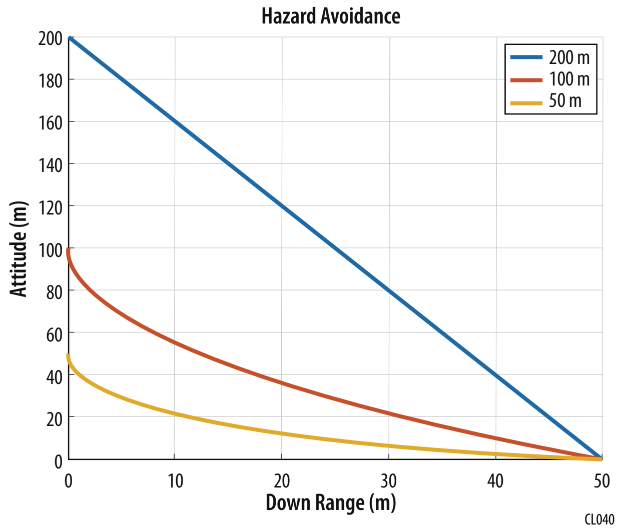

Figure 2-19: Trajectory diversion at various points along trajectory if Hazard is detected

Descent and landing begins at 2 km with a main engine burn to slow the spacecraft for its descent. Using fine maps built during the landing dry runs the spacecraft descends to 500 and the matches Centaur body rates. The spacecraft continues to descend to 200 m using TRN. TRN is designed to position the Spacecraft within 10 m of the target landing site. At 200 m HA begins and guides the spacecraft down to 40 m. During HA the spacecraft can accommodate unknown hazards with the capability to adjust its landing site by upto 50m. At 40 m above the landing site the horizontal velocity is reduced to < 0.1 m/s, the main engine cuts off to prevent contamination of the landing site and the spacecraft continues to the surface with a vertical velocity of < 1 m/s.

When one of the spacecraft legs senses contact with the surface the anchoring system fires and secures the spacecraft to the surface. In addition, the ACS thrusters on the top deck of the Spacecraft fire for 2 seconds as a backup to ensure the spacecraft is on the surface. Once the spacecraft is securely on the surface the lander uses its position knowledge to point its HGA at Earth and begins its landed operations.

2.4 Landed Operations

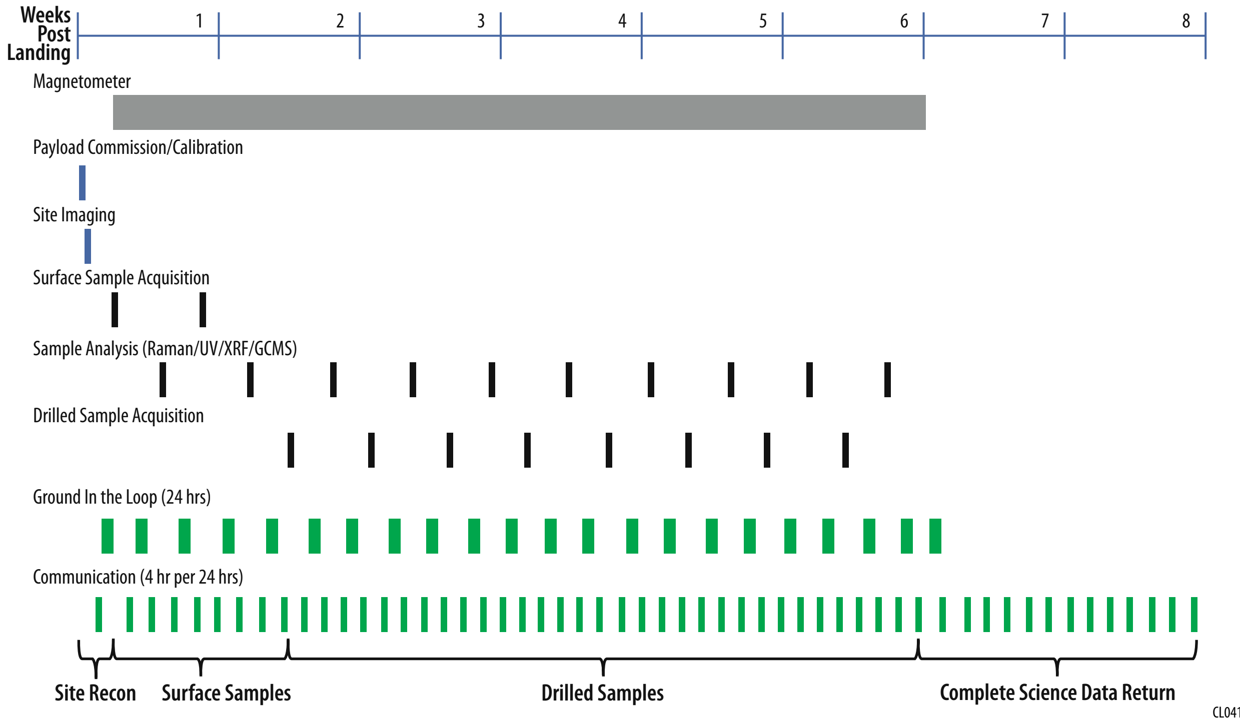

Figure 2-20: Landed Operations

The landed operations (Figure 2-20) begin with post landing instrument commissioning/calibration and end after the science data collected on the surface has been downlinked. 8 weeks are needed to complete landed science operations at a single landing site. 8.8 Gbit of payload data are generated with 1.1 Gbit designated decisional data requiring downlink to the ground during landed operations. This allows for go / no go decisions and payload parameter adjustments for nominal operations at designated decisions points.

The pace of landed operations is set by the assumption of 4 hrs of DSN time per 24 hrs and the need for Ground in the Loop (GITL) operations at key decision points in the landed operations timeline. A minimum of 8 hours after science data is received on the ground is designated for data analysis, decisional meetings, and command sequence selection for uplink. Off nominal operations requiring command sequence generation and validation would be handled by using time on the surface currently allocated to margin and may require 36 – 48 hours per GITL operation.

Instrument commissioning/calibration consists of processing sample blanks after landing.

The GCMS initial post landing sample will be in 'sniff' mode, bypassing the sample handling system. Images from each sample handling system foot camera are taken after instrument commissioning. A site panorama is taken consisting of images spaced at 15 degrees with a gimbal rate of 0.5 deg/sec.

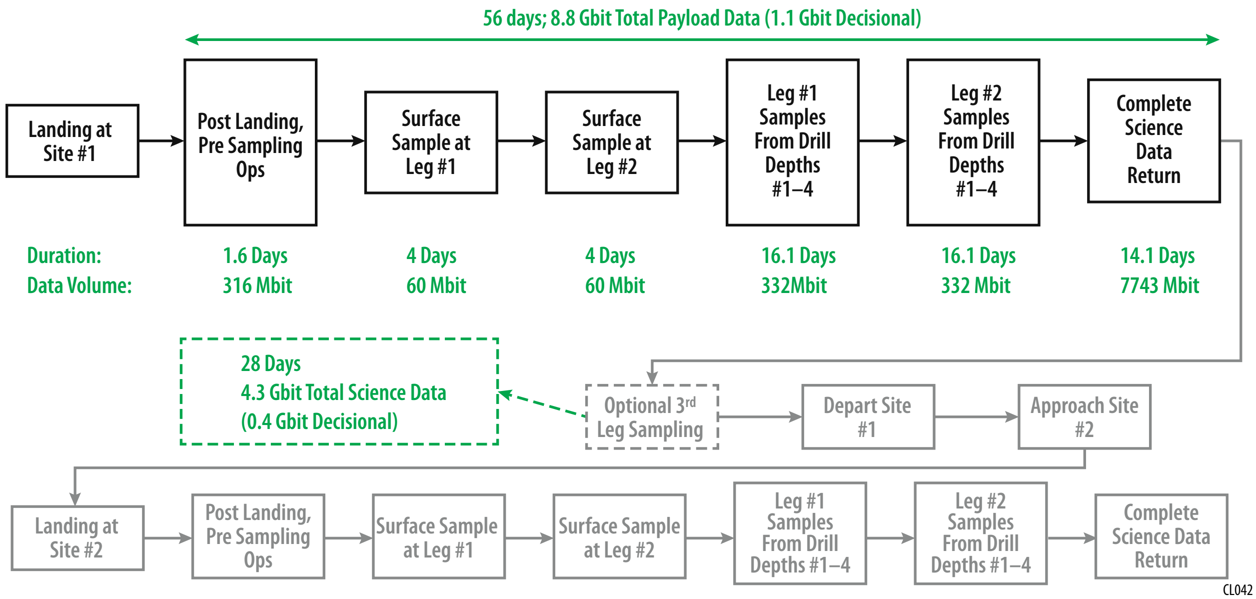

Figure 2-21: Landed Nominal Sampling Sequence

After the site panorama is taken, image and instrument commissioning data is returned for the initial post landing GITL operation. Instrument commissioning, site reconnaissance and GITL operations are completed approximately 2 days after landing. The magnetometer is turned on after the initial GITL operation and collects data through the end of the sample collection and analysis campaign for a duration of approximately 40 days. Magnetometer data is considered non decisional and is downlinked at the conclusion of landed science operations.

A nominal sampling sequence (Figure 2-21) consists of 1 surface sample from each of 2 legs and 4 samples at depth from each of 2 legs for a total of 10 samples at the landing site. GITL operations occur after sample collection and before sample analysis. All sample handling system data is considered decisional. After the decision to proceed with sample analysis is made, the sample analysis is completed and decisional instrument data, assumed to be 25% of the total sample analysis data volume, is sent to the ground.

After collection and analysis of 10 samples is completed, 2 weeks are needed to downlink the remaining science data. Upon completion of the nominal landed operations sequence, samples could be taken from the third landing leg equipped with a sample acquisition system. A nominal sampling and data return sequence consisting of 1 surface sample and 4 samples at depth lasts 28 days and generates 4.3 Gbit of additional science data.

The option exists to take a 3rd set of samples from the backup sample acquisition system as shown in Figure 2-21. In addition, the mission design supports a take off from site 1 and a 2nd landing at site 2. The nominal sequence would be repeated at the second site and would include the option to also sample from the backup sample acquisiton system.

3.0 SPACECRAFT

The CORAL Spacecraft is shown in Figure 3-1. The CORAL Spacecraft in the Falcon Heavy payload fairing is shown in Figure 3-2.

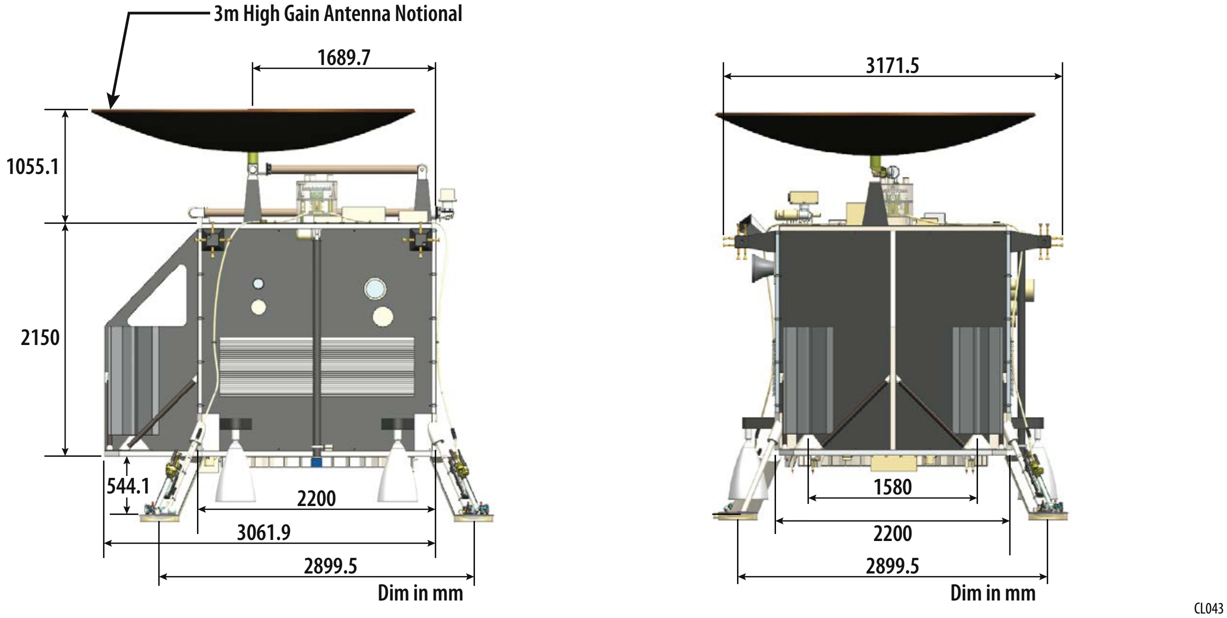

Figure 3-1: CORAL Spacecraft

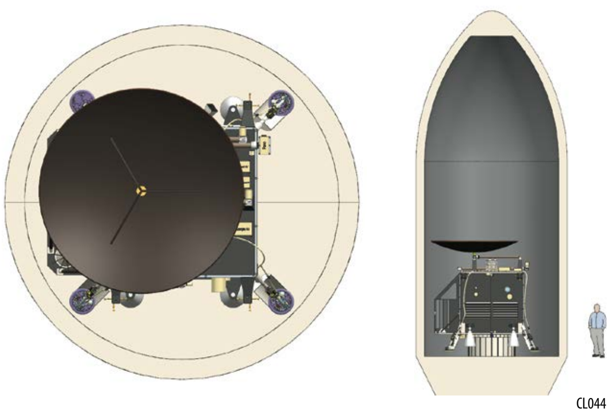

Figure 3-2: Figure CORAL Spacecraft in the Falcon Heavy Expendable 5m Payload Fairing

Table 3-1: Flight System Element Characteristics Table

Flight System Element Parameters (as appropriate)	Value/ Summary, units
General	
Design Life, months	168
Structure	
Structures material (aluminum, exotic, composite, etc.)	Composite
Number of articulated structures	3
Number of deployed structures	3
Aeroshell diameter, m	N/A
Thermal Control	
Type of thermal control used	Passive, heaters and louvers
Propulsion	

Estimated delta-V budget, m/s	1,376 m/s (21-day LP mission total)
Propulsion type(s) and associated propellant(s)/oxidizer(s)	Regulated bipropellant
Number of thrusters and tanks	20 ACS Thrusters, 4 Main Engines, 2 MMH Tanks, 2 NTO Tanks, 2 Pressurant Tanks
Specific impulse of each propulsion mode, seconds	Primary, ME Mode: 315s (299.7s at -3σ) Secondary, ACS Mode: 300s (285s at -3σ)
Attitude Control	
Control method (3-axis, spinner, grav-gradient, etc.).	3-axis, spinner
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial, Venus-Nadir, Solar
Attitude control capability, degrees	< 0.1 degrees
Attitude knowledge limit, degrees	< 30 arcsec
Agility requirements (maneuvers, scanning, etc.)	DSM, Landing, TRN, HA
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	HGA, PanCam Arm
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	CSS: 2π stradian ST: 30 arcsec boresight IMU: ARW = 0.07 deg/root-hour, Bias: 1 deg/hr RCS: 5 lb Wheel: 0.2 Nm, 250 NMS
Command & Data Handling	
Flight Element housekeeping data rate	1 kbps
Data storage capacity	3.5 Tbits
Maximum storage record rate	2,000 kbps
Maximum storage playback rate	2,000 kbps
Power	
Type of array structure (rigid, flexible, body mounted, deployed, articulated)	N/A
Array size, meters x meters	N/A
MRTG	Two 16-GPHS STEM-RTGs
Solar cell type (Si, GaAs, Multi-junction GaAs, concentrators)	N/A
Expected power generation at Beginning of Life (BOL) and End of Life (EOL)	800 W (BOL), 580 W (EOL)
On-orbit average power consumption	450 W
Battery type	Li-ion
Battery storage capacity	15.25 amp-hours

3.1 Science Instruments

The complementary suite of science instruments were selected to focus on meeting the first-priority science requirements in the STM (see Table 1-1 in the main report). The science instruments described herein are meant to be a representative payload and not the actual instruments that will be fly on this mission. This payload will be used in two different phases of the mission – orbital and landed science phases.

During the orbital phase, the heritage infrared spectrometer (OVIRS/OSIRIS REx), ultraviolet spectrometer (Alice/Rosetta) and high-resolution imagers (WAC and NAC/LRO) are used for characterizing the environment and target.

The imagers and imaging spectrometers are mounted on the same side of spacecraft and will face nadir during most of the orbital phase.

The landed phase has the Gas-chromatograph mass spectrometer which is modeled on flown mass spectrometers including Ptolemy on Rosetta/Philae, X-ray fluorescence instrument (PIXL/Mars 2020 Perseverance), and a combined Raman and UV spectrometer system (SHERLOC/Mars 2020 Perseverance) to perform in-situ elemental, isotopic, and organic analyses of the samples on the Centaur surface. The in-situ instruments are mounted on the top deck encircling the sample carousel (see Figure 3-3).

The standard fluxgate magnetometer (Magnetometer/MAVEN) is mounted on a deployable boom to measure the magnetic properties from orbit and near the surface. The spacecraft uses a deployable panoramic camera for contextual imaging of the landing site and local region sampled for chemical analysis.

The sample handling system with drill is a pneumatic sample acquisition system responsible for collecting samples from surface and sub surface depths and distributing samples via a carousel to the GCMS, XRF and UV and Raman combined instruments. Three sample handling systems will be implemented in this design, each attached near the lander footpads. One of the three sample handling system will be used as a backup system. See more details in Section 2-4.

Instrument Payload Description

The CORAL instruments are shown in Figure 3-3 and 3-4. The instrument mass and power values are summarized in Table 3-2 and the instrument characteristics are given in Table 3-3.

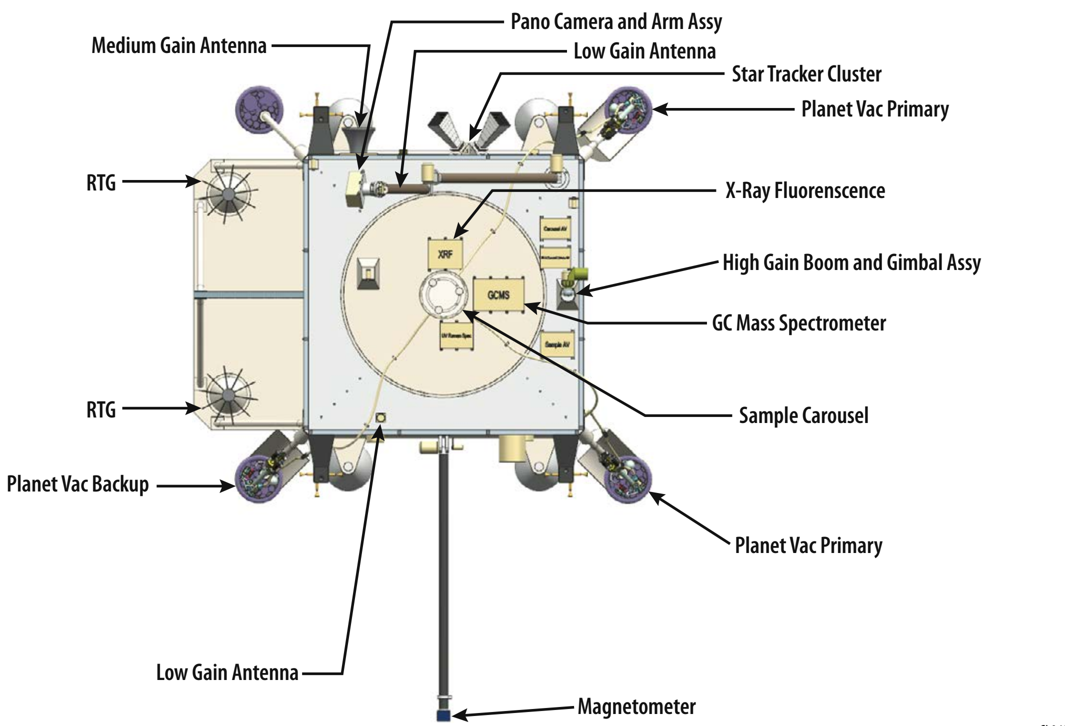

Figure 3-3: CORAL Lander with Instruments (Top View)

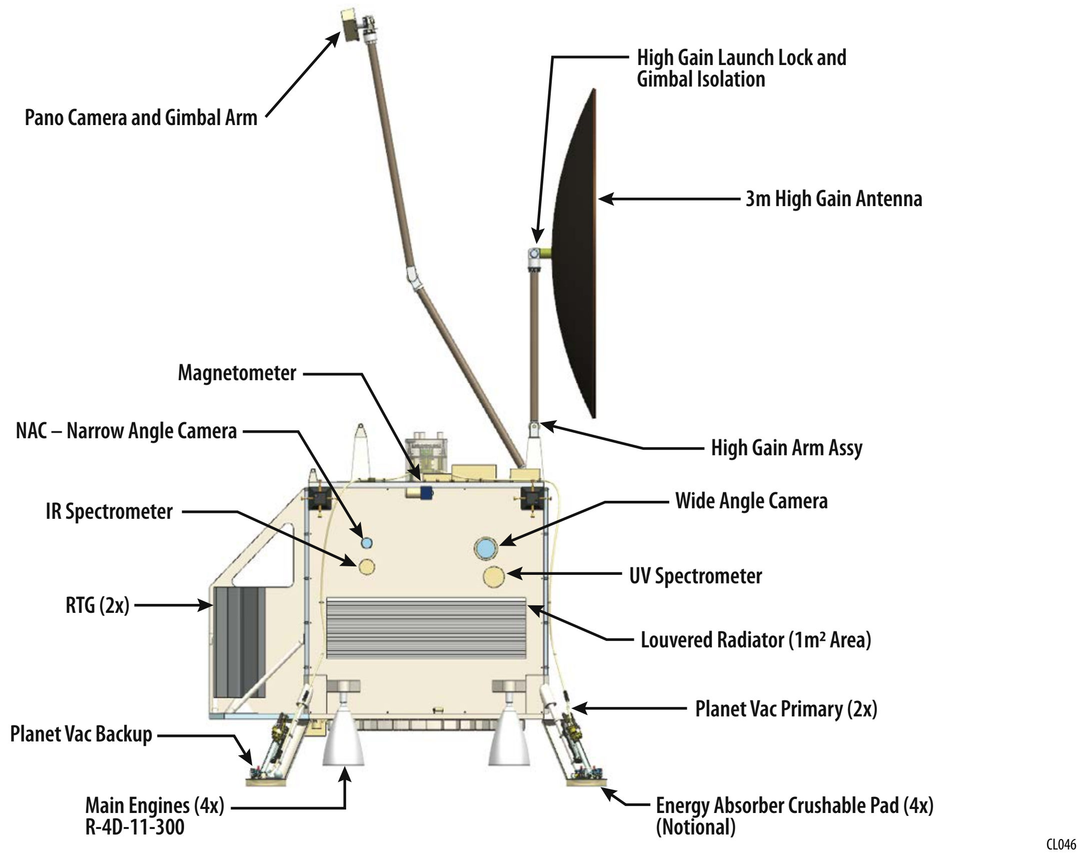

Figure 3-4: CORAL Lander with Instruments (Side View)

Table 3-2: Payload Mass and Power Summary

Instrument Name	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
GCMS	5.2	30	6.8	48.0	30	62.4
XRF	6.9	30	9.0	25.0	30	32.5
NAC (2)	16.4	30	21.3	18.6	30	24.2
WAC	0.9	30	1.2	2.7	30	3.5
UV and Raman Spectrometer	4.7	30	6.1	48.8	30	63.4
IR Spectrometer	17.7	30	23.0	13.5	30	17.6
UV Spectrometer	4.5	30	5.9	4.5	30	5.9
Magnetometer	1.5	30	2.0	1.0	30	1.3
Total Payload Mass	**57.3**	**30**	**75.3**			

Gas Chromatograph Mass Spectrometer:

This mission carries a Gas Chromatograph Isotope Ratio Mass Spectrometer (GC-IR-MS) modeled on elements with high heritage.

The instrument is responsive to the applicable needs of the Science Traceability Matrix (STM), such as isotope ratios. It also has capability for objectives beyond those levied by the STM, such as other organics. Capabilities beyond those levied by the STM were not prioritized for design or analyzed for capability, so they will only be mentioned in rare cases.

The sample can enter the instrument via the paths indicated in the diagram (See Figure 3-5, GCMS Block Diagram). Solid sample enters via loading into the oven. Freely volatilizing sample directly from the surface of the Centaur can also enter via a gas inlet tube.

Minimum m/z capability is suitable to measure D/H ratios in water and ammonia, and maximum capability can go well beyond carbon dioxide. Again, capability for measurements beyond satisfying the STM were not prioritized for design or analyzed for capability.

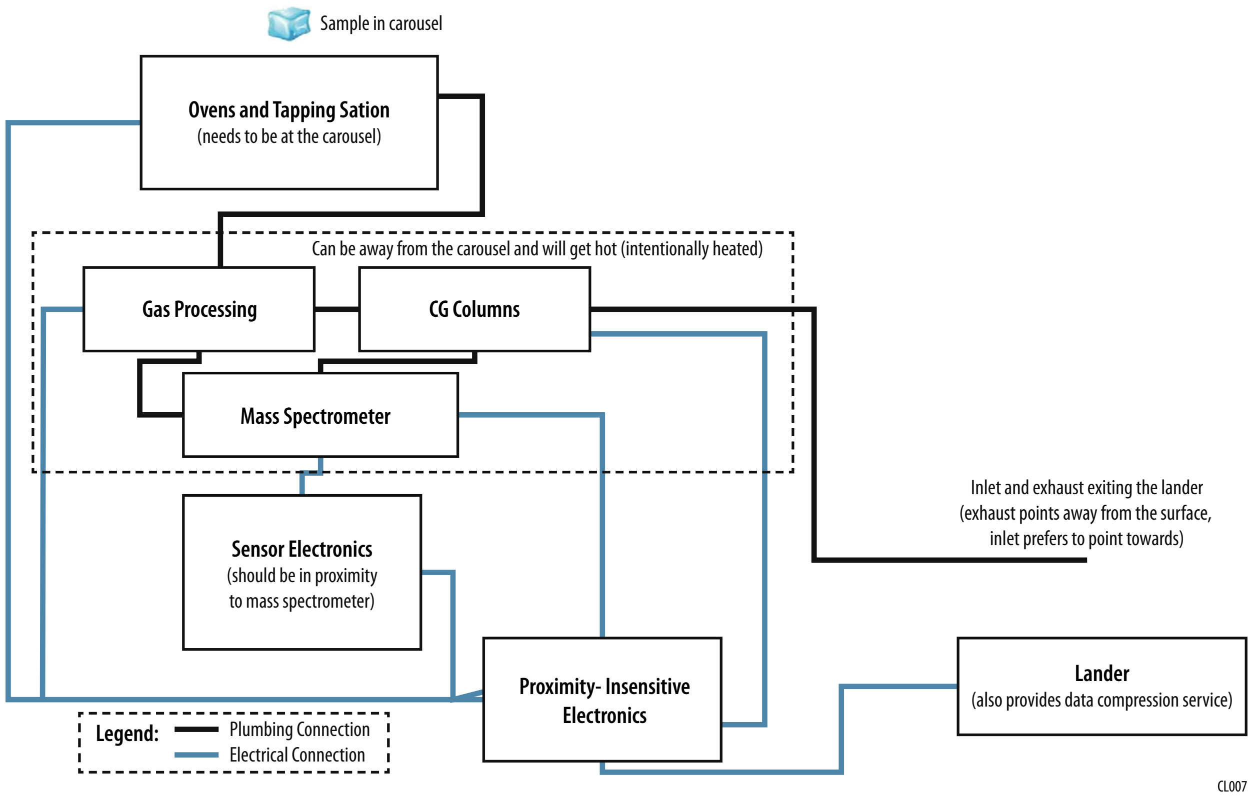

Figure 40: GCMS Block Diagram

XRF:

The X-Ray Fluorescence spectrometer is based on the Planetary Instrument for X-ray Lithochemistry (PIXL) flown on the NASA Mars 2020 Perseverance rover. This instrument measures the fine-scale chemical makeup of rocks using an X-ray spectrometer and camera. It is a microfocus instrument that has a high sensitivity for detection of trace chemical elements at sub-millimeter spatial resolution levels. The PIXL design incorporates a high-resolution camera for 2D fast mapping of rock samples. Its rapid spectral acquisition system can measure major and minor elements within seconds. This operationally efficient XRF instrument meets the required science measurement of mineralogy at the surface.

Combined Raman and UV Spectrometer:

The combined Raman and UV spectrometer is based on the Scanning Habitable Environments with Raman & Luminescence for Organics & Chemicals (SHERLOC) instrument. It has also flown on NASA Mars 2020 Perseverance rover which combines Raman and Deep UV-induced native fluorescence for fine scale detection. This 2D spectral mapper use: cameras, spectrometers and a UV laser to detect and classify organics and minerals present in rocks and help understand the environment in which the rock sample formed. SHERLOC's mapping mode is used for nondestructive, sub-picogram sensitive organic detection. SHERLOC spectra can be complementary with measurements made by other payload elements, including elemental abundances measured by PIXL.

Infrared spectrometer:

The IR spectrometer is modeled after the NASA OSIRIS-REx visible and infrared spectrometer (OVIRS). This is a point spectrometer covering wavelengths from 0.4 µm to 4.3 µm. It will provide spectral mapping of the surface composition and global context for the sampling site.

UV spectrometer:

The UV spectrometer is based on the Alice ultraviolet imaging spectrometer on board Rosetta focuses on spectral features in the far-ultraviolet wavelength range from 70 nm to 205 nm. It will be used to characterize the target and study the surface properties.

Magnetometer:

The NASA/GSFC standard fluxgate magnetometer has direct heritage with the MAVEN mission. It will provide continuous, high resolution coverage of the magnetic field about the Centaur.

Cameras:

The NASA Lunar Reconnaissance Orbiter Camera (LROC) Wide Angle Camera (WAC) and the Narrow Angle Cameras (NACs) were representatives imagers for this study with two positions that are used for collecting high resolution images suitable for crater counting and global imaging of the body. The WAC is a 7-color push-frame camera (100 m/pixel and 400 m/pixel visible and UV, respectively), while the two NACs are monochrome narrow-angle line-scan imagers (0.5 m/pixel).

The NAC includes a sequence and compression system for data processing prior to data transfer to the spacecraft command and data handling.

Table 3-3: CORAL Instrument Payload Characteristics

Item	GCMS	XRF	NAC	WAC	Spectometers: UV and Raman	Spectometers: Visible and Infrared	Spectometers: Ultraviolet	Magneto-meter
Type of instrument	Mass Spectrometer	X-ray fluorescence	Monochrome Camera	Color Filter Camera	Imaging Spectrometer	Imaging Spectrometer	Imaging Spectrometer	Magnetometer
Number of channels	TBD	TBD	1	1	TBD	TBD	TBD	TBD
Size/dimensions (m x m x m)	0.25 x 0.40 x 0.15	0.21 x 0.27 x 0.23	0.70 x 0.27 diameter	0.16 x 0.23 x 0.32 (incl. radiator)	0.26 x 0.20 x 0.06	0.49 x 0.41 x 0.29	0.2 x 0.41 x 0.14	0.08 x 0.10 x 0.12
Instrument mass without contingency (Kg CBE*)	5.2	6.9	8.2	0.9	4.7	17.7	4.5	1.5
Instrument mass contingency (%)	30	30	30	30	30	30	30	30

Instrument mass with contingency (Kg CBE+Reserve)	6.8	9.0	10.7	1.2	6.1	23	5.9	2.0
Instrument average payload power without contingency (W)	48	25	9.3	2.7	48.8	13.5	4.5	1
Instrument average payload power contingency (%)	30	30	30	30	30	30	30	30
Instrument average payload power with contingency (W)	62.4	32.5	12.1	3.5	63.4	17.6	5.9	1.3
Instrument average science data rate^ without contingency (kbps)	1.4	2.22	20,000	20,000	11.1	183	0.69	2
Instrument average science data^ rate contingency (%)	30	30	30	30	30	30	30	30
Instrument average science data^ rate with contingency (kbps)	1.82	2.89	26,000	26,000	14.4	238	0.89	2.6
Instrument Fields of View (degrees)			2.85	92 (mono-chrome) 61 (visible) 59 (UV)				
Pointing require-ments, knowledge (degrees)			30	30				
Pointing require-ments, control (degrees)			60	60				
Pointing require-ments, stability (degrees/sec)			0.1	0.1				

*CBE = Current Best Estimate.

^Instrument data rate defined as science data rate prior to on-board processing

NOTE: The WAC and NAC pointing requirements in the table are based on LRO mission. They are applicable for the CORAL mission and are typical capabilities for flight systems.

*CBE = Current Best Estimate.

^Instrument data rate defined as science data rate prior to on-board processing

3.2 Avionics

Figure 3-6 shows the CORAL spacecraft block diagram. The CORAL avionics consists of a Power System Electronics unit (PSE), Command and Data Handling unit (C&DH), Propulsion / Mechanism Electronics unit (PME), Pan Camera Gimbal Unit, Aperture Covers Unit, Antenna Tower Deploy Unit, Antenna Gimbal Unit, and RTG Pump Unit. The PSE, C&DH, and PME are implemented with Goddard's Mustang Avionics. The Mustang Avionics will fly on the PACE mission in 2023. The rest of the units are implemented with Moog Gimbal Control Electronics (GCE). The Moog Gimbal Control Electronics flew on the NICER Mission.

The PSE, C&DH, and PME are block redundant. The PSE and C&DH backup units fly as warm backups, and the PME backup flies as a cold backup. The GCE are internally redundant. Their backup cards fly as cold backups.

The PSE consists of 6 types of cards. Its PSE Monitor Card provides PSE control and telemetry acquisition functions. The RTG Interface Module performs battery charging and distribution functions, and handles 16 Amps of current. A Segment Module handles 40 Amps of RTG and battery current. Altogether, the RTG Module and Segment Module can handle 56 Amps of power. The requirement is 42 Amps, leaving a margin of 25%. Three High Current Power Output Modules provide three 15 Amp outputs, six 7.5 Amp outputs, and twelve 3 Amp outputs. One 15 Amp output is required for the Drill Electronics Unit, and five 7.5 Amp outputs are required for the reaction wheels and Ka-Band TWTA. Three Low Current Power Output Modules have sixteen 3 Amp outputs each. Those outputs along with twelve 3 Amp outputs on the High Current Power Output Modules provide a total of sixty 3 Amp outputs. Fifty 3 Amp outputs are required, leaving a margin of 20%. Lastly, a Low Voltage Power Converter provides the secondary voltage power required by the PSE cards.

The C&DH consists of 5 types of cards. Its Processor Card is based on a GR712RC Dual-Core LEON3FT SPARC V8 Processor ASIC (200 MIPS). It has 32 MB SRAM and 128 Gits of Flash memory. Two Housekeeping Cards provide a total of 20 course sun sensor inputs (15 are required), 8 pressure sensor inputs (8 are required), and 138 temperature sensor inputs. A Communication Card provides the RF Communication system interfaces. A Data Storage card provides 3.5 Tbits of data storage. 1 Tbit of data storage is required. Lastly, a Low Voltage Power Converter provides the secondary voltage power required by the C&DH cards.

The PME consists of 4 types of cards. Two Main Engine Valve Drive Card provides the actuation outputs for the four Main Engines. Three ACS Valve Drive Cards provide a total of twenty four thruster outputs (20 are required), and 12 latch valve outputs (12 are required). A Deployment Module provides 8 actuation outputs. Two deployment outputs are required: One for the high gain antenna launch lock, and one for the Magnetometer boom deployment. Lastly, a Low Voltage Power Converter provides the secondary voltage power required by the PME cards.

The GCE units are identical. They consist of 3 types of cards: Two Controller Cards, two Gimbal Drive Cards, and two Low Voltage Power Converters. Each Gimbal Drive Card controls the dual coil of one gimbal.

The total mass of the avionics system 71.5 kg MEV. The peak power of the avionics system, including the warm backups with the gimbals in full torque, is 195.8 W MEV.

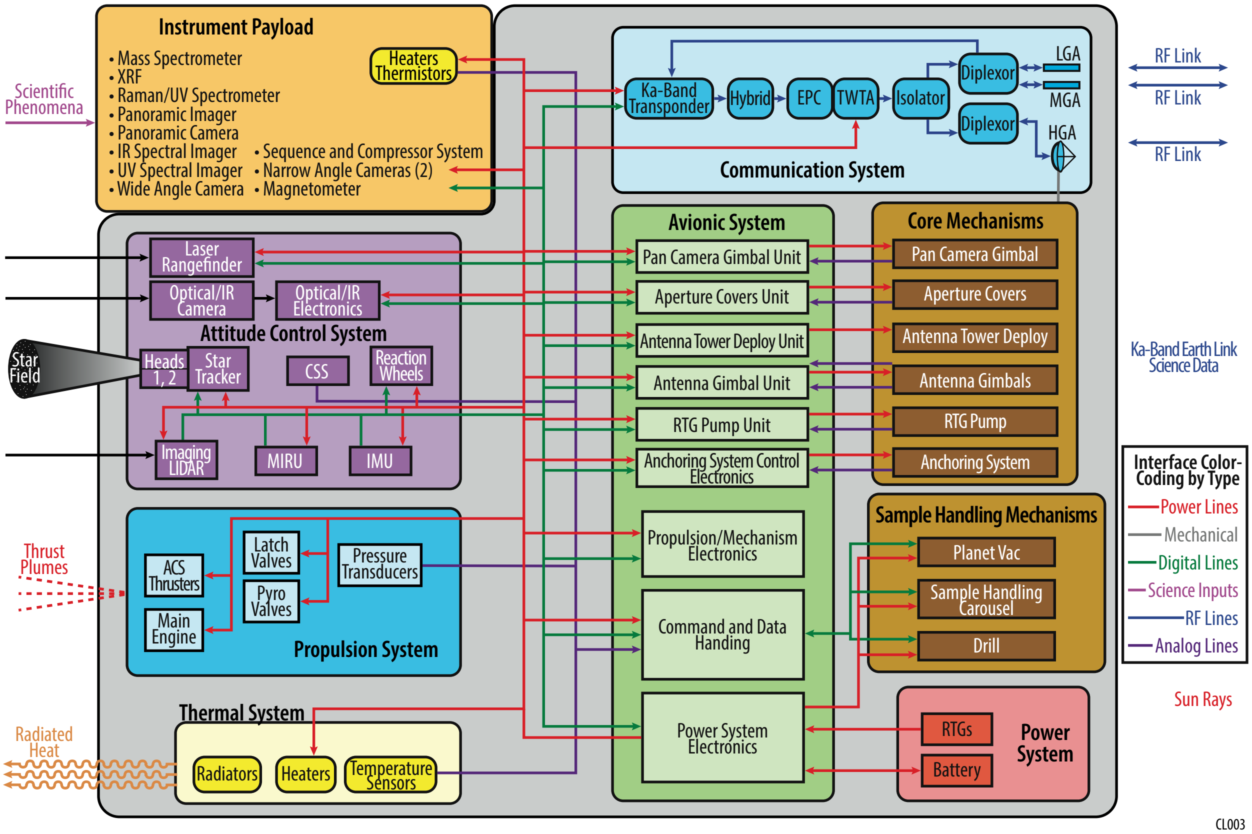

Figure 3-6: Spacecraft Block Diagram

3.3 Attitude Control

Mission objective of Centaur Orbiter & Land Mission Concept Study (CORAL) is to measure the chemical and physical properties of a Centaur to understand the accretion and evolution of icy planetesimals. Goal of this mission to understand early solar system compositional reservoir of planetesimals, understand the accretion and dynamical evolution of primordial icy planetesimals, geological and evolutionary process that influenced these icy bodies and biological potential. Top science priority is to determine global mineralogical composition, impact history and relative ages, physical characteristics, and landforms and any evidence for changes over the mission of icy planetesimals.

The objective of the Attitude Control System (ACS) is to perform pointing control via means of onboard actuators (Reaction wheel & Thrusters) to accomplish phasing burns, Decent Orbit Injection, and final landing. In addition to providing adequate controllability, ACS also provides the knowledge of final position and orientation. Accurate knowledge will be used for directional antenna and instrument deployment.

Rendezvous and Proximity Operations (RPO) and final decent landing are considered critical phases of flight. The mission requirements imposed on ACS are as follows:

1. Lander final actual position within 10 meters of target site as the target sites are identified within 1 Km clear region of hazard.
2. Lander final position knowledge within 1 meter. This is needed immediately after landing to point High Gain Antenna (HGA)

3. Lander velocity at touchdown less than 1 m/s to prevent any damage to its support legs.

On CORAL, the following set of sensors were selected to provide good knowledge estimation

- Coarse Sun Sensor Assembly (x16)
  - » Provides estimate of direction of the sun. Each coarse sensor comes as in grouping of four which provides 2pi steradian angle. Each of these assemblies are mounted on each side of spacecraft to get full 360 degrees of coverage
- Novatel IMU-LN200
  - » LN200 is Inertial Measurement Unit which provides low noise estimate of angular velocity
- DTU Micro Advanced Steller Compass
  - » Since, no knowledge or pointing requirement is explicitly specified, in addition to digital processing unit, 3 camera heads are selected that are able to provide attitude knowledge on the order of few arcseconds. Each of the camera heads include baffle and a MEMS gyro which is able to provide rate estimates.

The following set of ACS actuators are utilized on CORAL. Selection of these ACS actuators was based on their ability to perform fine pointing, momentum management and delta-v capability.

- Honeywell HR16-50
  - » 4 wheels each with 50 Nms. of momentum capacity was selected primary based on RPO and landing phase of mission. 4-wheel pyramid configuration provide redundancy. Each provide 0.2 Nm around their spin axis. Figure 3-7 shows the wheel mounting configuration. Primary reason for these wheels is to provide precise pointing as they are not primary source for slew maneuvers.
- Moog DST-13 ACS thrusters
  - » A net total of 20 5-lb thrusters were selected. 16 thrusters provide full 6-DOF attitude control and positional control. 4 thrusters are used for redundancy. of them provide secondary ACS control capability and 4 additional thrusters are used as contingency for main engine failure. These thrusters are used primary for slew and maneuver to different attitude and provide wheel momentum management. Table 3-4 shows the ACS control authority from both translation and rotational maneuvers.

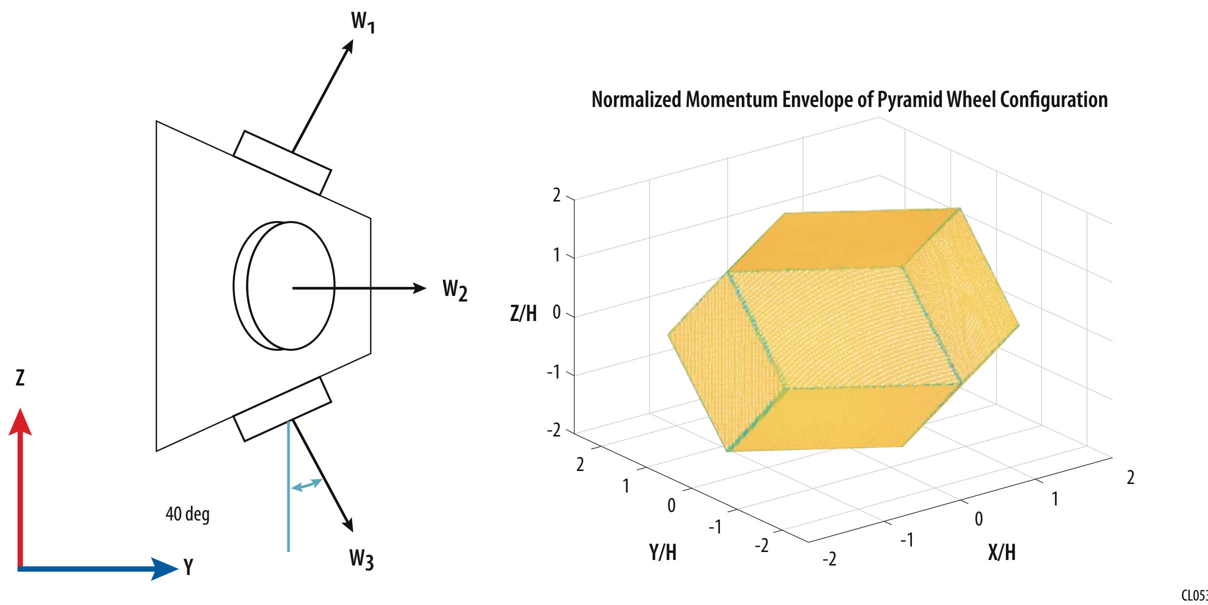

Figure 3-7: 4-Wheel 40-degrees wheel Pyramid configuration (Left), Effective Momentum Capacity on Each axis (Right)

Table 3-4: ACS Thruster Capability

Force (N)			Torque (Nm)		
X	Y	Z	X	Y	Z
80	80	80	88	88	88

The FSW control loop Ensures that ACS meets mission constraints. The ACS block diagram is shown in Figure 3-8. The baseline Concept of Operation assumes a ground command that is received by onboard GNC system. The navigation system, utilizing an onboard attitude sensor, provides CBE CORAL attitude to the control system. The controller uses desired attitude and CBE attitude to provide commands to the thruster system. The resulting change in attitude is sensed by the sensors to provide a closed loop feedback to GNC FSW. Furthermore this diagram is used during landing phase to provide full 6-DOF control

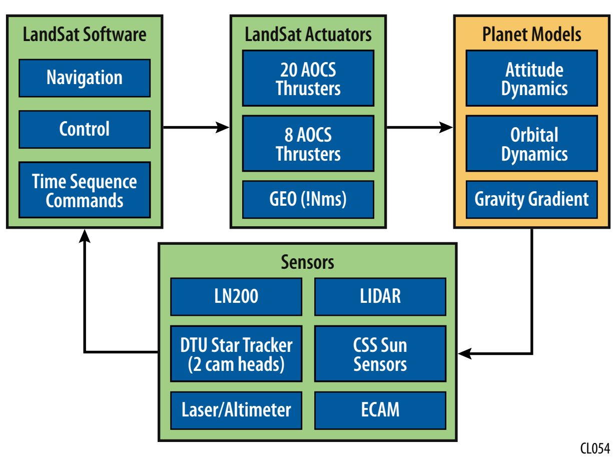

Figure 3-8: GNC Block Diagram

There are total of 7 spacecraft control modes as following:

- Launch
  - » ACS in inhibited in this mode until CORA has separated from Launch Vehicle and required amount of time has passed to avoid potential collision
- Rate Null
  - » This mode utilizes RCS thrusters on board to null rates. This mode is used once ACS becomes active after separation and removes the tip-off rates
- Mission
  - » Inertial Ponting Mode
    - ▪ This mode is used most of the time during transfer trajectory. Spacecraft is held to a desired attitude.
  - » Delta-V Mode
    - ▪ This mode is used during main engine to place the spacecraft in transfer trajectory and in between Trajectory correction maneuvers.
- Delta H
  - » This mode is used to dump momentum from the wheels. RCS are fired occasionally to desaturate the wheels. This mode is specifically used during RPO mode when fine pointing is desired for TRN and LIDAR mapping
- RPO
  - » In this mode, spacecraft is princely pointed and utilizes combination of 16 RCS thruster and reaction wheel. Lander can purely be translated or rotated. TRN as well as fine maps are generated for final decent phase
- Land
    - ▪ In this phase, on orbit landing guidance is used to land the spacecraft. In this mode spacecraft is also performing occasional Hazard avoidance to ensure vehicle lands in obstacle free area.

All hardware, with the exception of the GSFC inhouse LIDAR, is at TRL 6 or higher. The GSFC LIDAR is in development and has TRL of 4-5. This development version is expected to fly on OSAM-1 before CORAL mission and would expected to have achieved TRL-9

3.4 Communication

The communications subsystem must maintain uplink and downlink with the Deep Space Network (DSN) with high enough data rates to ensure timely delivery of the mission science, telemetry, and command data volumes.

Figure 2-2: 2015 BQ311 DRM distance to Sun and Earth during interplanetary transfer in the mission design section shows that the maximum range is 6.7 AU. This worst case distance is used to size all communication links. Link performance is shown in Table 3-5. Mission Operations and Ground Data system data is provided in Table 3-6.

For this mission a single frequency system was determined to meet the needs and constraints. This system consists of a General Dynamics (GD) Small Deep Space Transponder (SDST), Traveling Wave Tube (TWT) with electronic power conditioner (EPC), high power isolator, a parabolic high gain antenna (HGA), a medium gain antenna (MGA), and two low gain antennas (LGAs). These components have redundancy where necessary and are connected with RF cabling, waveguides, switches, and diplexers to allow for options in the communications path. This system is designed to only use Ka-Band as it provided the best option for high data rates within the power consumption constraint.

Table 3-5: CORAL Links

From	To	Operational Frequency (GHz)	Tx Antenna	Tx Antenna Gain (dBi)	Rx Antenna	Range (km)	Au	Information Rate (kbps)
Downlink								
Spacecraft	Earth	32	HGA – 3m	58.67	DSN-34m	1346380836	6.7	77.5
Spacecraft	Earth	32	MGA	22	DSN-34m	1346380836	6.7	0.021
Spacecraft	Earth	32	LGA	7.4	DSN-34m	1346380836	6.7	0.0007
Uplink								
Earth	Spacecraft	34.45	DSN-34m	79	3m HGA	1346380836	6.7	137
Earth	Spacecraft	34.45	DSN-34m	79	MGA	1346380836	6.7	0.613
Earth	Spacecraft	34.45	DSN-34m	79	LGA	1346380836	6.7	0.036

All these components have flight heritage the only component that needs modification will be the MGA as the original component was designed for X-Band and will therefore need to be scaled appropriately to function at Ka-Band frequencies.

Table 3-6 Mission Operations and Ground Data Systems

Communications	Launch and Cruise	Rendezvous, Proximity and Mapping	Landed Science
Number of Contacts	1 per month	1 per day	1 per day
Number of Weeks for Mission Phase, weeks	432	190	8
Downlink Frequency Band, GHz	32	32	32
Telemetry Data Rate(s), kbps	HGA > 77.5 MGA > 0.021 LGA > 0.0007	HGA > 77.5 MGA > 0.021 LGA > 0.0007	HGA > 77.5 MGA > 0.021 LGA > 0.0007
Transmitting Antenna Type(s) and Gain(s), DBi	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4
Transmitter peak power, Watts	200	200	200
Downlink Receiving Antenna Gain, DBi	79	79	79
Transmitting Power Amplifier Output, Watts	100	100	100
Total Daily Data Volume, (MB/day)	>139.5	>139.5	>139.5
Uplink Information			
Number of Uplinks per Day			
Uplink Frequency Band, GHz	34.45	34.45	34.45
Telecommand Data Rate, kbps	HGA > 137 MGA > 0.613 LGA > 0.036	HGA > 137 MGA > 0.613 LGA > 0.036	HGA > 137 MGA > 0.613 LGA > 0.036
Receiving Antenna Type(s) and Gain(s), DBi	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4	HGA 58.67 MGA 22 LGAs 7.4

The system will always be in receive mode but to save power the transmitting components will be turned off for long durations when they are unnecessary. The transmit will need to be turned on periodically for the purposes of ranging with the DSN and for telemetry updates. The landed portion of the operation drove the need for a gimballed HGA as the orientation, rotation rate, and landing location will all determine the pointing needs for the antenna and those factors will be unknown until a landing site is chosen during the mission. The gimballed dish will need to be stowed for landing operations as it will not be useful at that time and the strength of the gimbals and waveguides cannot be guaranteed under high accelerations from the attitude control system (ACS) while deployed.

All of the technologies in this subsystem are well developed and there is little concern that they won't be available for this mission. The scaling adjustment to the MGA will take some NRE but theoretically should be a very straightforward modification. The LGAs are derived from a feed used on other flight hardware and should only need to be tested as independent horn antennas. If the MGA modifications are unsuccessful, a suitable alternative Ka-Band antenna can be found or manufactured as Ka-Band technology is reaching a high level of maturity.

3.5 Contamination

Isotopic and organic analyses that the spacecraft performs while landed can be affected by trace gases from chemical propulsion. To mitigate against this the landing operation cuts off the main engine at 40m above the landing site and free falls to the surface.

A plume analysis should be done to confirm that the 40m cutoff altitude is sufficient to avoid deposition of contaminates on the landing site. Additional mitigation steps would include performing an analysis to determine the composition of chemical propulsion in detail in order to identify and correct for any potential contamination of the samples. This would include an analysis of below the surface and application of evolved gas analyses for isotopes . For instance, water from the propulsion would be outgassed first and those from the sample/minerals/organics would be outgassed as you continue to heat the sample so then the D/H of the sample could be distinguished from that of the propulsion contamination.

3.6 Power

CORAL uses two 16-GPHS STEM-RTGs to power the spacecraft. At the beginning of the life (BOL) the output power from the RTGs is 800 W. This accounts for 3 years storage after fueling before launch. The power output is 580 W (EOL) at the End of Design Life (14 years after launch, 17 years after fueling). The CORAL baseline mission is 12 years. The RTGs have a mass of 62 kg. Each RTG is 0.47 m in diameter (fin tip to tip) with a 1.07 m length.

Because the RTG puts out a constant power (there is a 1.9 % degradation per year) it was necessary to create a power profile to ensure that the MEV value of all spacecraft power demands is below the power output. The MEV power demands include 30% contingency above the current best estimate. Tables 3-7 and 3-8 show the power profile.

There is a small 15.25 AH Li-ion battery on board that provides 500 W during peak power usage to cover negative power margins. These negative power margins occur during the mapping orbit because all instruments are assumed on at the same time which is not going to be true in reality. Landing does need all the navigation hardware on, but the duration of landing is very short and the battery provides the extra W for that short duration. During drilling phases the drill is used for ~3 minutes at the peak power. The battery provides the extra power needed during that short time.

Table 3-7: Power Profile for Cruise

Cruise Phase													
			Instruments										
Date	Duration (Years)	Event	Orbit	Surface	ACS	Avionics and RTG Pump	Communication	Thermal and Harness losses	Sample Acquisition	Total Power Required	Total Power Available	Margin
1/23/2040	0	Launch Power	36.6	22.2	17.5	155.5	284.7	82.5	0	594.6	800	205.4
6/23/2040	0.4	Cruise Power	36.6	22.2	17.5	120.9	284.7	82.5	0	560	793.7	233.7
12/17/2040	0.5	Propulsion Burn	36.6	22.2	17.5	140.4	284.7	82.5	0	579.4	786.4	206.9
1/23/2042	1.1	Cruise Power	36.6	22.2	17.5	120.9	284.7	82.5	0	560	769.9	209.9
2/2/2042	0	Earth Flyby	36.6	22.2	17.5	120.9	284.7	82.5	0	560	769.5	209.5
4/12/2044	2.2	Propulsion Burn	36.6	22.2	17.5	140.4	284.7	82.5	0	579.4	737.8	158.4
5/16/2044	0.1	Jupiter Flyby	36.6	22.2	17.5	120.9	284.7	82.5	0	560	768.5	208.6
1/23/2045	0.7	Cruise Power	36.6	22.2	17.5	120.9	284.7	82.5	0	560	758.5	198.5
11/28/2045	0.8	Propulsion Burn	36.6	22.2	17.5	120.9	284.7	82.5	0	560	746.3	186.3
1/23/2046	0.2	Cruise Power	36.6	22.2	17.5	120.9	284.7	82.5	0	560	744.1	184.1
1/23/2047	1	Cruise Power	36.6	22.2	17.5	120.9	284.7	82.5	0	560	729.9	170
1/23/2048	1	Cruise Power	36.6	22.2	17.5	120.9	284.7	82.5	0	560	716.1	156.1
1/20/2049	0.4	Rendezvous and Approach	36.6	22.2	17.5	140.4	284.7	82.5	0	579.4	711.1	131.7
6/2/2049	3.7	Proximity and Mapping	48.3	22.2	96.7	140.4	284.7	82.5	0	670.2	660.7	-9.5
2/12/2053	0.2	Landed Operations										

Table 3-8: Landed Power Profile

Landed Phase										
Total Elapsed Time (Days)	Event	Orbit Instruments (Avg MEV W)	Surface Instruments (Avg MEV W)	Sample Acquisition (Avg MEV W)	Spacecraft Subsystems (Avg MEV W)	Total Power Required (Peak MEV W)	Total RTG Power Available (W)	RTG Peak Margin (W)	Battery (W)	Total Peak Margin (W)
0	Landing Power	30.0	22.2	0.0	634.0	686.1	660.7	-25.4	500.0	474.6
0	Deployments	30.0	22.2	0.0	522.7	574.9	660.7	85.8	500.0	585.8
0	Post Landing Commissionng and Pre-Sampling	0.0	80.6	0.0	464.7	545.3	660.7	115.4	500.0	615.4
1.6	Surface Sample Leg #1	0.0	22.2	68.9	464.7	555.8	660.4	104.6	500.0	604.6
5.6	Surface Sample Leg #2	0.0	22.2	68.9	464.7	555.8	657.7	101.9	500.0	601.9

9.6	Leg #1 Sample From Drill Depth #1	0.0	22.2	518.7	464.7	1005.6	653.1	-352.5	500.0	147.5
13.6	Leg #1 Sample From Drill Depth #2	0.0	22.2	518.7	464.7	1005.6	645.3	-360.3	500.0	139.7
17.6	Leg #1 Sample From Drill Depth #3	0.0	22.2	518.7	464.7	1005.6	634.8	-370.8	500.0	129.2
21.6	Leg #1 Sample From Drill Depth #4	0.0	22.2	518.7	464.7	1005.6	621.9	-383.7	500.0	116.3
25.7	Leg #2 Sample From Drill Depth #1	0.0	22.2	518.7	464.7	1005.6	606.6	-399.0	500.0	101.0
29.7	Leg #2 Sample From Drill Depth #2	0.0	22.2	518.7	464.7	1005.6	589.2	-416.4	500.0	83.6
33.7	Leg #2 Sample From Drill Depth #3	0.0	22.2	518.7	464.7	1005.6	569.9	-435.7	500.0	64.3
37.7	Leg #2 Sample From Drill Depth #4	0.0	22.2	518.7	464.7	1005.6	548.9	-456.7	500.0	43.3
41.8	Complete Science Data	0.0	22.2	0.0	464.7	486.9	520.9	34.0	500.0	534.0
55.9	End landed Operations									

3.7 Propulsion

The CORAL propulsion subsystem (Figure 3-9) is a large regulated bipropellant system. The propellant is minimized by using a high C3 launch. It carries 1,400 kg of propellant to perform three principal functions: Two deep space maneuvers, a rendezvous maneuver, and proximity operations. The propellant is stored in COTS tanks. A set of 4 x 450 N engines (AJ PN R-4D-15) are used for the main maneuvers, with twenty (20) smaller 22N engines for ACS and proximity operations. Separate pressurization manifolds are used to provide regulated pressure to both the fuel and oxidizer tanks. All of the components are COTS.

The system is single fault tolerant. Each pressurization string is fully redundant.

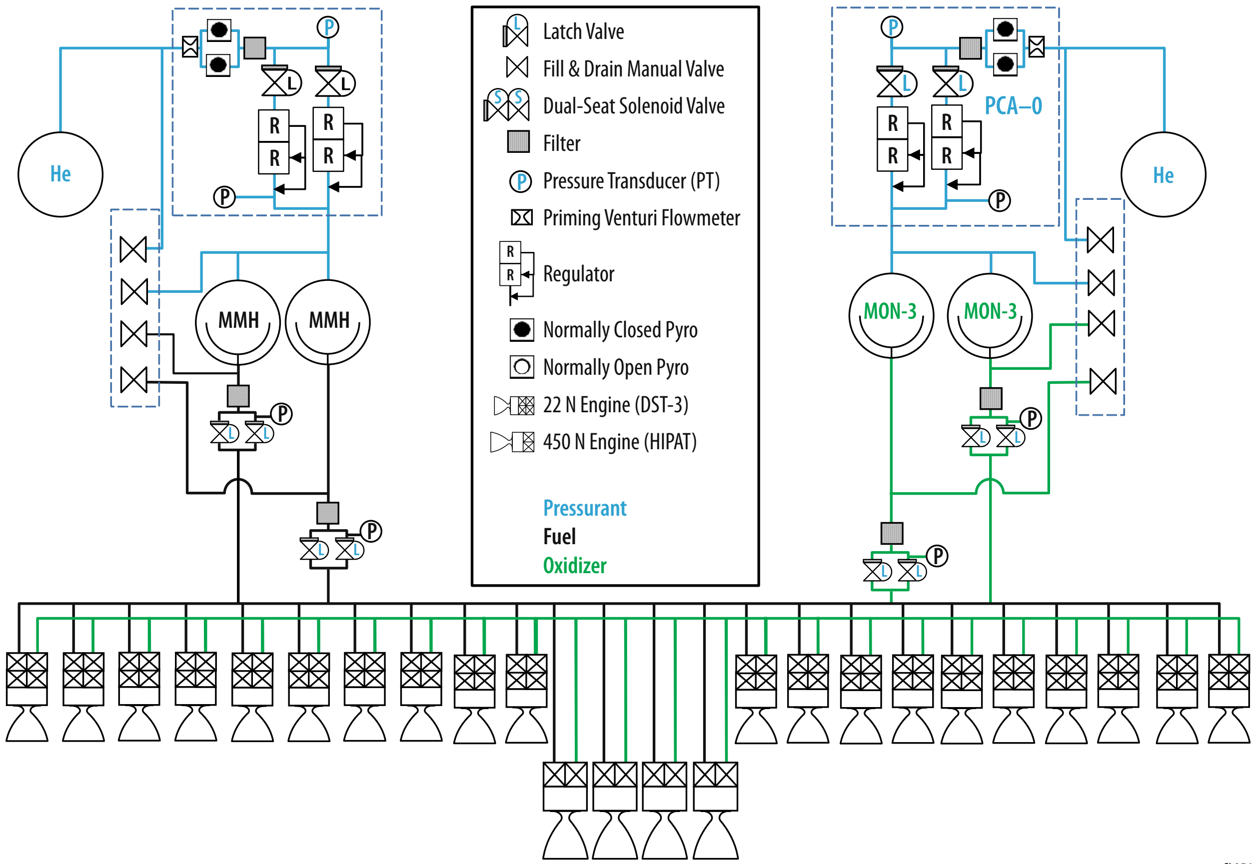

Figure 3-9: Schematic of the Propulsion Subsystem

The pressurant tanks are isolated by redundant pyro valves during launch. The system is pressurized during the transfer to lunar orbit insertion and a calibration maneuver is performed. All maneuvers are performed with the main engine, except for smaller orbit maintenance maneuvers.

All of the components in the system are TRL-9.

3.8 Key Mechanisms

CORAL has several key mechanisms that enable it to conduct its operations. Three of these: Panoramic Camera and Arm. Drill and the Sample Acquisition System are part of the Sample Acquisition and Handling System. In addition, CORAL uses an anchor system to hold it to the Centaur during landed operations. Mass and Power for each of these is shown in Table 3-9. There are additional mechanisms, including but not limited to, reaction wheels, HGA gimbals, launch locks, aperture covers, deployment mechanisms and a RTG heat pump that are not discussed here but are included in the spacecraft mass and power Table 3-9.

Table 3-9: Key Mechanisms Mass and Power Summary

Mechanism Name	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Panoramic Camera and Arm						
Panoramic Camera	1.0	30	1.3	12.0	30	15.6
Arm	12.4	30	16.2	30.0	30	39.0
Sample Acquisition and Handling System						
Drill	18.0	30	23.4	375	30	487.5
SAS (Sample Acquisition System)	7.5	30	9.8	30.0	30	39.0
Carousel	5.5	30	7.2	30.0	30	39.0
Anchor System	2.6	30	3.4	48.0	100	96.0
Total Key Mechanisms Mass	**47.0**	**30**	**61.3**	**555**	**41.7**	**716.1**

3.8.1 Panoramic Camera and Arm

The panoramic camera arm (PanCam Arm, Figure 3-10) was designed to provide a view of all the sample areas and is critical for science operations. The PanCam Arm is a 3m mechanical arm with two 1.5m sections five-axis robotic arm with a single axis shoulder mounted to a single axis turntable, a single axis elbow, and three-axis gimballed head to provide maximum flexibility for the Panorama Camera to obtain context imagery around the landed spacecraft.

The PanCam is launch locked until after landing.

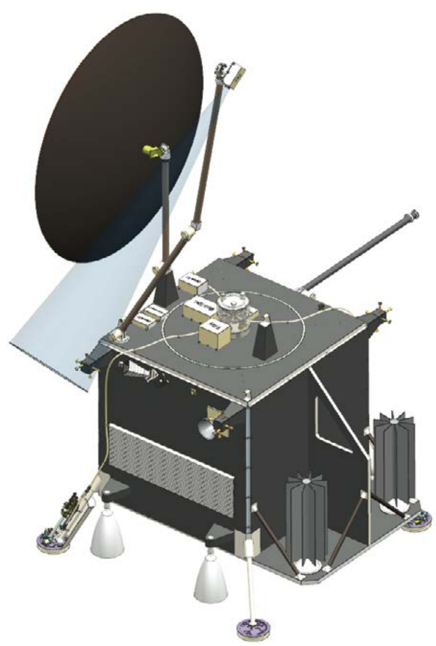

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Figure 3-10: PanCam Arm Field of View around HGA

3.8.2 Sample Acquisition and Handling System

In order to complete CORAL science objectives, a sampling system must be created to retrieve both surface and sub-surface materials and deliver them to the science instruments mounted on the lander. For the purposes of the study a representative sampling system was selected. The representative system is the PlanetVac sampling system, a rotary percussive drill, and a sample carousel. PlanetVac is the system being used on the Martian Moons Exploration (MMX) mission launching in 2024 and NASA's Commercial Lunar Payload Services (CLPS) program will fly a PlanetVac sampler in 2023 to the surface of the Moon. The drill is based off drills being used on Dragonfly and VIPER, which are going to Titan in 2026 and the Moon in 2023, respectively. While the sample carousel will be based on the Dragonfly carousel. Charactristics of the sample acquisition and handling system are shown in Table 3-10.

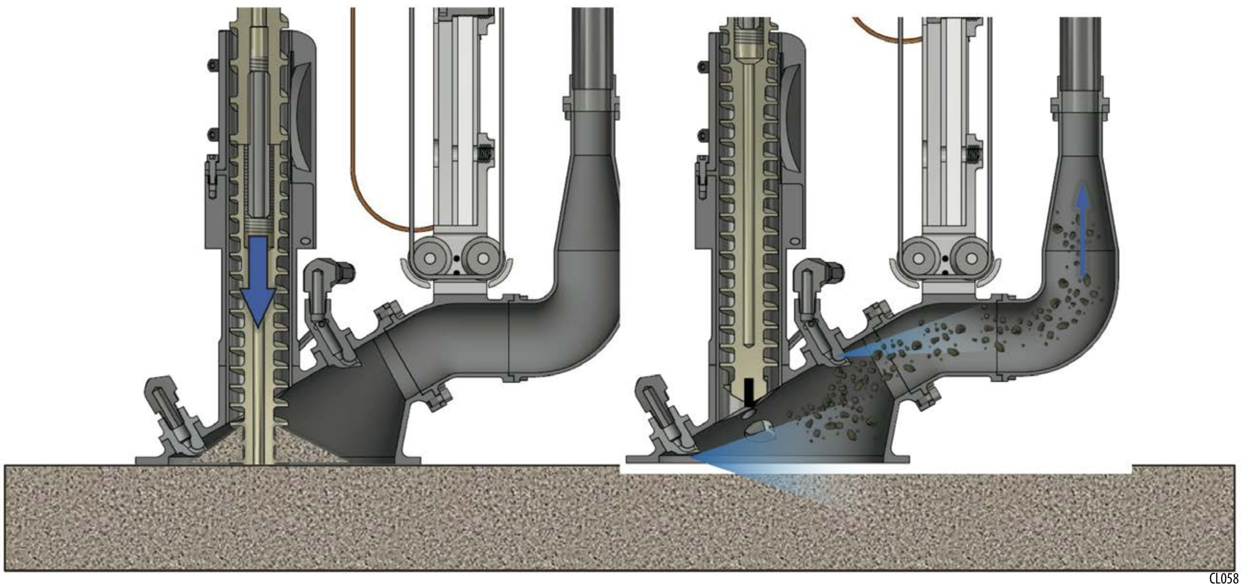

Figure 3-11: PlanetVac sprays compressed gas to loft material

The PlanetVac system sprays compressed gas at the surface to loft loose material (Figure 3-11) and uses the pressure differential to transport the sample through pneumatic lines to the sample carousel. To collect sub-surface material (Figure 3-12), the rotary percussive drill pulls material to the surface, where PlanetVac can then collect a sample using the same method.

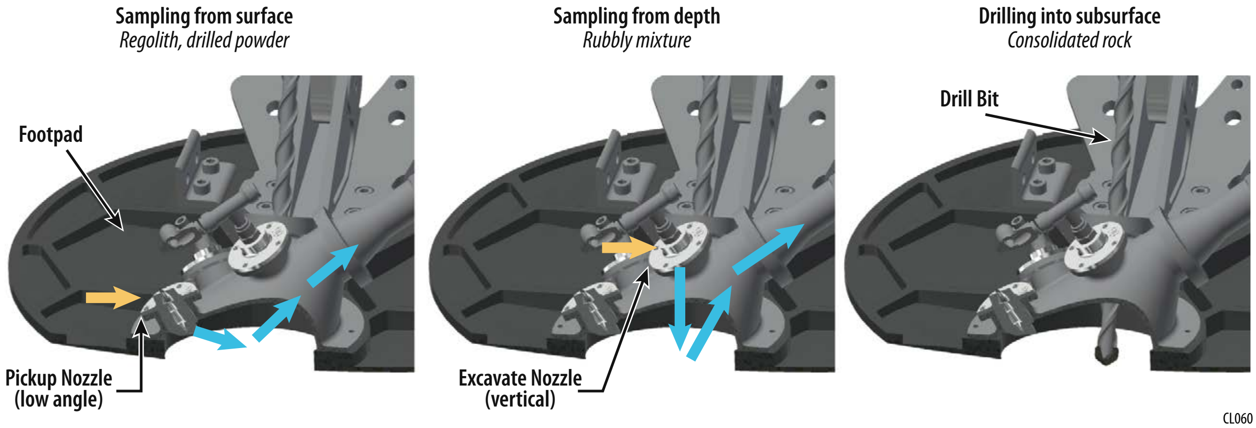

Figure 3-12: PlanetVac Sampling from surface and at depth

3.8.3 Carousel

Once collected, the sample moves to the carousel (Figure 3-13), which consists of a rotary mechanism that controls the position of the sample cups, and linear elevators that lift the samples into position where they can interact with the science instruments and the pneumatic system.

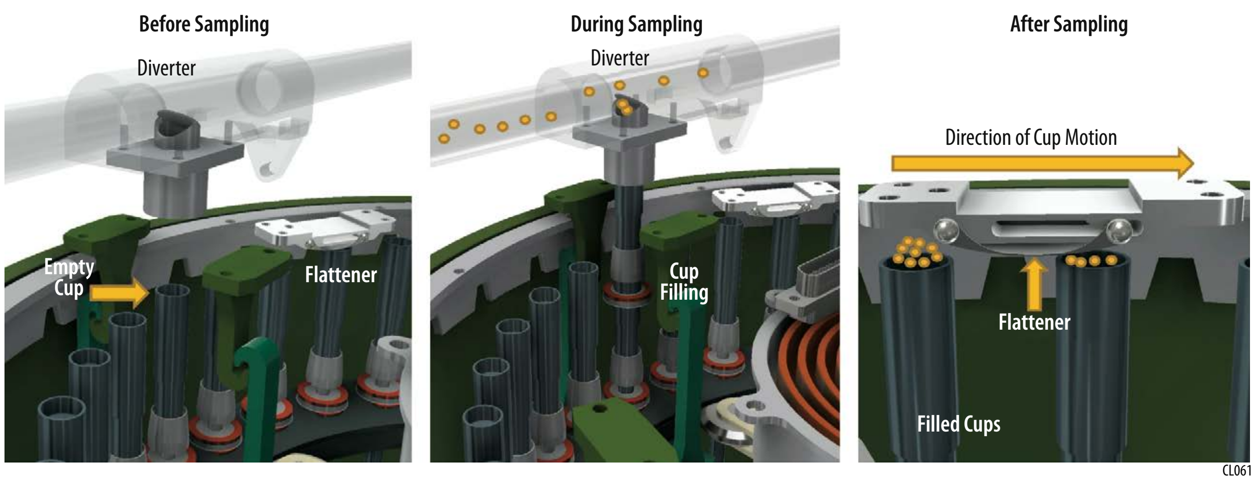

Figure 3-13: Carousel concept of operation

An important aspect of the mission is to maintain sample temperature below 200 K. In order to achieve this the heat from sampling operations and lander sources are restricted. During pneumatic sampling, there is no concern of heating due to the cooling effect associated with decompression of the carrier gas. For drilling, the two sources of heating are conductive heat from the motors travelling to the drill bit and friction heating generated between the drill bit and surface material. Both heat sources can be controlled by adjusting drilling parameters and duty cycling as needed; this will increase overall sampling time but should not impact active time. While in the carousel, the heat sources are the actuators and conductive heating from the spacecraft body, these are addressed by creating an enclosure within the carousel mechanism that isolates samples from the actuators and insulative stand-offs between the carousel structure and the spacecraft structure.

Table 3-10: Sample handling system characteristics

Item	Drill	PlanetVac	PV/Drill Avionics	Carousel	Carousel Avionics	Total	Units
Type of instrument							
Number of channels	3	1		5			
Size/dimensions (for each instrument)	0.18 x 0.18 x 0.61	0.3 x 0.20 x 0.18	0.125 x 0.26 x 0.192	Cylinder 0.36 dia x 0.31	0.0635 x 0.16 x 0.24		m x m x m
Instrument mass without contingency (CBE*)	6	4.7	6.5	12.2	4.5	33.9	Kg
Instrument mass contingency						0	%
Instrument mass with contingency (CBE+Reserve)	6	4.7	6.5	12.2	4.5	33.9	Kg
Instrument average payload power without contingency	375	15	30	23	24	467	W
Instrument average payload power contingency						0	%
Instrument average payload power with contingency	375	15	30	23	24	467	W
Instrument average science data rate^ without contingency	22	2400 per op 3000 per image		8000 per op 3000 per image		22	kbps
Instrument average science data^ rate contingency						0	%
Instrument average science data^ rate with contingency	22	2400 per op 3000 per image		8000 per op 3000 per image		22	kbps
Instrument Fields of View (if appropriate)	n/a	n/a	n/a	n/a	n/a	n/a	degrees
Pointing requirements (knowledge)	n/a	n/a	n/a	n/a	n/a	n/a	degrees
Pointing requirements (control)	n/a	n/a	n/a	n/a	n/a	n/a	degrees
Pointing requirements (stability)	n/a	n/a	n/a	n/a	n/a	n/a	deg/sec

3.8.4 Anchor System

Four Anchoring systems are mounted to the underside of CORAL (Figure 3-14). The Anchoring system ensures that CORAL remains on the surface during the final step of landing and during landed operations subsurface drilling. The system is a customized version of the Harpoon Anchors used on Rosetta's Philae lander (Figure 3-15).

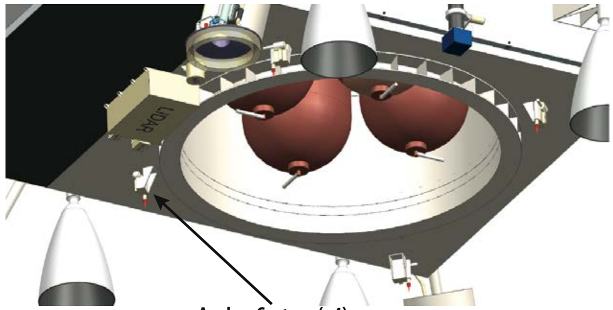

Figure 3-14: CORAL Anchor System Locations

Each Anchoring system consists of four main components: a copper beryllium projectile, a pyrotechnical expansion system, a cable magazine and a rewind system driven by a brushless gearmotor.

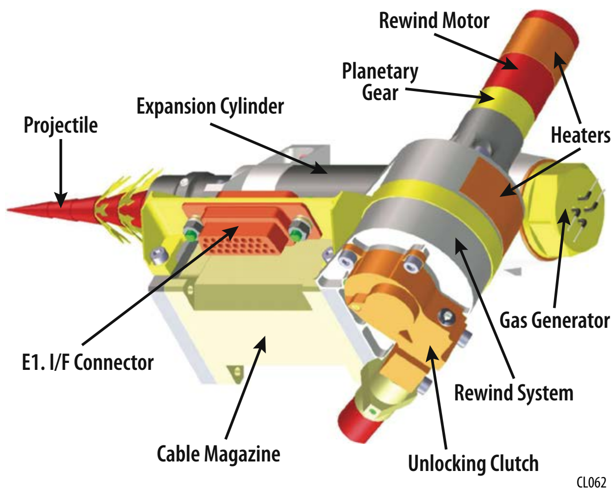

Figure 3-15: CORAL Anchor system

From 2003 Thiel et all:

“The anchoring projectile is designed to anchor safely in a wide range of different comet materials. Sharp notches and stainless steel barb rings at the anchor tip provide good anchoring capability in case of a strong, high density comet material whereas spring hinged shovel flaps with hard stops ensure safe anchoring in case of lower strength, lower density material. The anchor tip, the shaft and the shovel flaps are made from hardened copper-beryllium alloy ($CuBe_2$).

In case of an unexpectedly weak comet surface the projectile may penetrate deeper than the amount of anchor cable stored in the magazine (2.5 m) would al- low. To dissipate the remaining kinetic energy 25 cm of cable are stored in the shaft of the projectile and led through a cable brake, which is designed for a braking force of 60 N.”

Reference

Thiel, Markus & Stöcker, Jakob & Rohe, Christian & Kömle, Norbert & Kargl, Günter & Hillenmaier, Olaf & Lell, Peter. (2003). The Rosetta Lander anchoring system. Harris, R.A.: 10th European Space Mechanisms and Tribology Symposium, ESA Publications Division, 239-254 (2003).

3.9 Radiation

The trapped electron environment at Jupiter is severe. The spacecraft trajectory was obtained from the flight dynamics group. The JPL model (Divine and Garrett), updated after the Galileo mission, was used to evaluate the trapped particle environment for Jupiter. It includes trapped electrons, bremsstrahlung radiation and trapped protons (Figure 3-16). The JPL model was implemented through SPENVIS. The JPL group recommends using a margin of x2 with this model. The expected dose behind 100 mils of aluminum shield for this segment of the mission is 1.004 krad(Si), including a margin of x2.

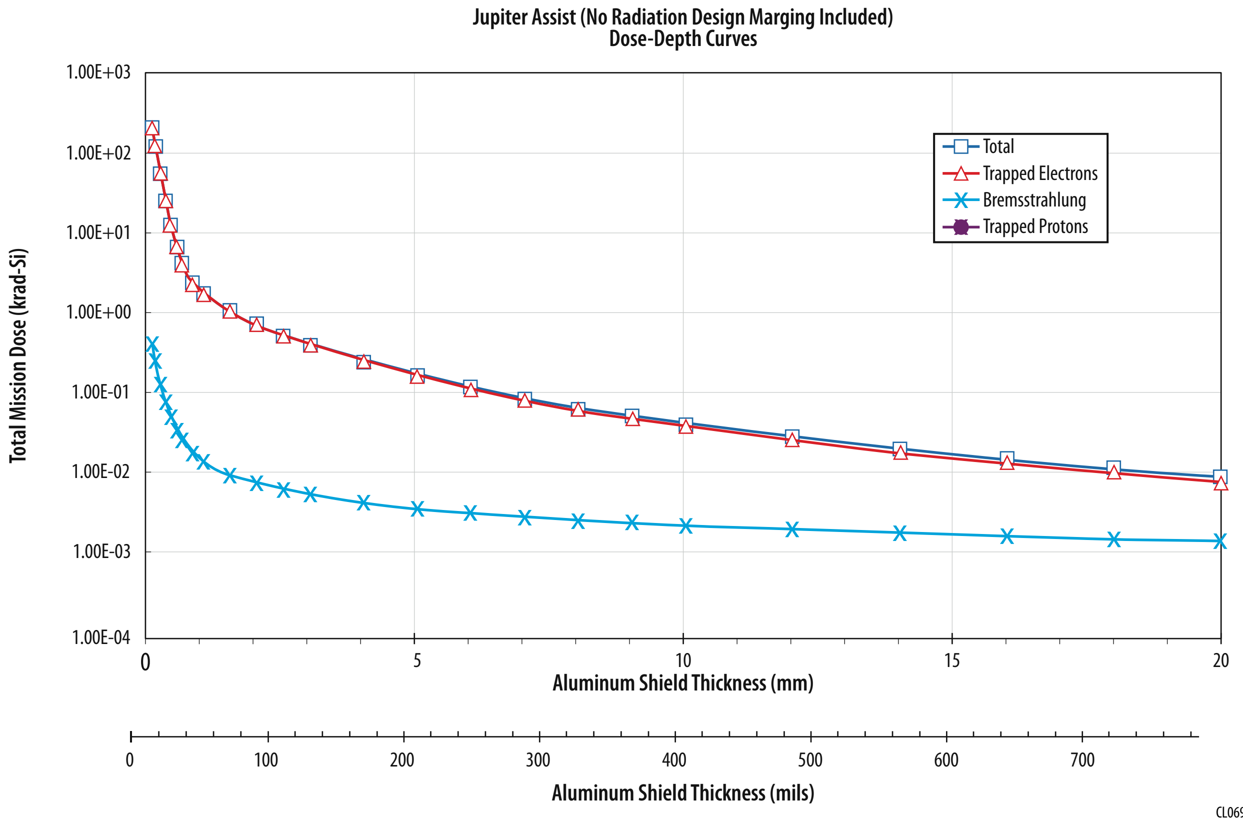

Figure 3-16: CORAL Dose Depth Curves

There is also exposure due to solar particle events during the mission. This depends on the phase of the solar cycle and the spacecraft distance from the sun (Figure 3-17). It has been conservatively assumed that 9 years of the mission occurs during the solar maximum period and 4 years in solar minimum. It is also assumed that the solar particle event flux falls off as 1/r2, where r is the distance of the spacecraft from the sun. The GSFC ESP model was used to evaluate this environment. The result is 4.25 krad(Si) for 100 mils of aluminum shield. These results are calculated for the 95% confidence level so no additional margin is required.

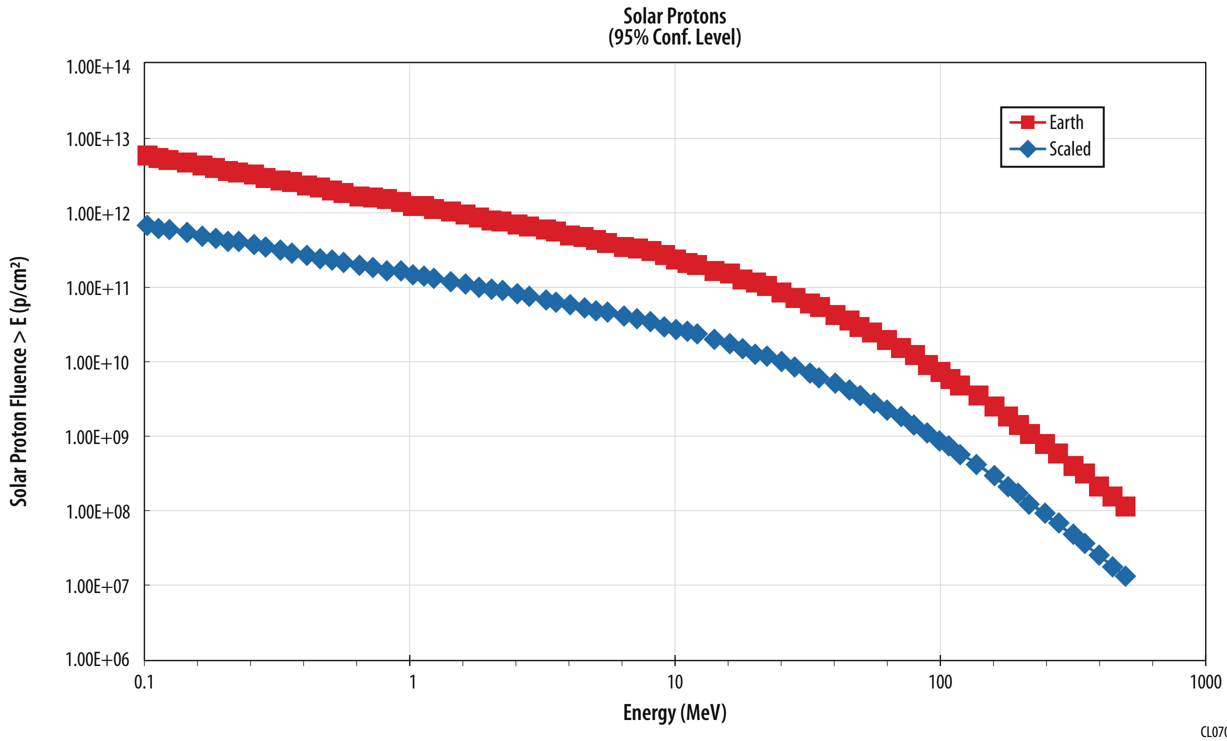

Figure 3-17: Solar particle events

Adding up the two contributions, the total ionizing dose requirement for the CORAL mission including margin is 5.254 krad(Si) for 100 mils of aluminum shield. As the spacecraft design matures a 3-D ray trace, Monte Carlo simulation using the CAD file can be done. This will relax these requirements.

3.10 Structure

Although the primary role of the CORAL lander is to accommodate the suite of science instruments, the resultant structural design was heavily driven by the propulsion system tanks and the Next Generation Radioisotope Thermoelectric Generators (RTG). The suite of instruments are described elsewhere in detail but, from a structural/packaging perspective, accommodating the sample handling system and providing a stable method for the feet to engage properly with the surface had the most influence on instrument packaging. The sample handling system has two primary and one backup system mounted to three of the four legs with a centrally located sample handling carousel. The primary systems are mounted on the legs furthest from the RTGs and the backup system is mounted on a leg near the RTGs. This was done to avoid the heat of the RTGs from compromising the sample site for the primary sample handling system. Another instrument with an influence on the structural design was the magnetometer since it required a deployable boom. The Wide and Narrow Angle Imagers, the IR Imaging Spectrometer and an UV Imaging Spectrometer are simply mounted on one of the side panels with openings for the Fields of View (FOV). It should be noted that details about these instruments were not known during this study so representative volumes were used along with notional FOVs.

The tall aspect ratio of the two fuel tanks and two oxidizer tanks was chosen to allow all the tanks to fit within the central cylinder whose diameter is based on a standard 1575 launch vehicle adapter. This allows carrying all the propulsion load directly into the launch vehicle. Since the propellant/oxidizer volume design drivers were late inputs into the structural design, trading tank shape, launch vehicle adaptor, tank mounting, and structural mass was left for future work. It seems likely that there is a more efficient solution. Tanks were packaged about the dry Center of Mass (CM) to reduce CM shift as the propellant is used. Small helium tanks were also mounted to the tank mounting structure to create a notional propulsion module to simplify integration. The details of packaging the components into a propulsion module were also left to future work.

The RTGs are expected to run around 400C. The thermal design preferred them separated from the main lander assembly as much as possible, to have a clear view of space, and to avoid a direct line of sight between the units and between the units and the instruments on the top deck to reduce radiant heating. It was also important that they do not have a view of the sample surface area where the radiant heat might compromise the sample. As a result, they were packaged on a platform with a dividing wall that acts both as structural support and thermal isolation. The location provides a clear view of space while isolating temperature sensitive regions. Avoiding a view of the top deck instruments was helped by the tall shape of the lander. It should be noted that if lower profile tanks are used in future iterations allowing the structure height to be reduced, the height reduction will likely be limited by the height of the RTGs.

The lander must also accommodate a 3-meter diameter gimballed High Gain Antenna system and a panoramic camera mounted on an articulated arm. The panoramic camera arm was designed to provide a view of all the sample areas and is critical for science operations. The High Gain Antenna needs to have the flexibility to point in almost any direction in the hemisphere and is critical for communication. The two large, articulated systems will need to be controlled to avoid each other. It was assumed that could be done with the control system algorithms. Limiting movement of the two systems will have an effect on operations and will need to be studied.

All structural elements have been sized using hand-calculations or rules-of-thumb (which is typically conservative) as time did not permit a full structural analysis. The design of the structure is a typical "cylinder-in-a-box" with composite and titanium bracketry and honeycomb panels with composite face sheets, aluminum honeycomb cores and titanium inserts. The panel components are assembled using the clip and post method employed on other composite structures such as LRO. The basic structure of a central cylinder, upper and lower deck, radials and equipment panels is very common and well understood. The dry mass efficiency of the structure (Primary Structure Mass/Dry Mass) is approximately 28% which is reasonable but hints at room for improvement.

Load Path and Mechanisms

The load path for the tanks is directly through the central cylinder providing an efficient, low mass, and commonly used design. The tall central cylinder is stiffened by structural radials. The mechanisms include the High Gain Antenna deployable boom with a two axis gimbal, the Magnetometer deployment boom with a simple drive/hinge/latch mechanism and the panoramic camera arm which is essentially a five-axis robotic arm with a single axis shoulder mounted to a single axis turntable, a single axis elbow, and three-axis gimballed head. All deployables will require launch locks. The release mechanism between the lander and the launch vehicle is expected to be a commercial-off-the-shelf (COTS) separation systems such as a Lightband or RUAG system. The mechanisms and launch locks are notional but are all expected to be COTS or modified COTS, well understood, and with extensive flight heritage.

Center of Mass and Landing

For the purposes of a baseline landing leg design, the Centaur was assumed to have a 17km diameter and a 2 gram per cubic centimeter density. This resulted in a very weak gravitational acceleration of 4.75mm/sec2. With a release height of 20m and a release velocity of 1 m/s the impact velocity was predicted to be 1.091m/s (very little acceleration from the Centaur gravity). Little was known of the max loads the instruments could endure so 6g deceleration, which is the maximum design load for the Falcon Heavy rocket, was used. That load was chosen because the instruments were going to have to survive that load as a minimum. At this velocity the stroke of the legs during impact must be at least 1cm to keep the landing g-loads at or below chosen design limit of 6g. Assuming a simple crush pad on the bottom of each leg, the stroke distance is the same as the crush distance. The worst case scenario for keeping landing loads below 6gs is if all four legs were to hit at the same time with evenly distributed loads. Since all 4 legs would be absorbing the Kinetic Energy (KE) at the same time, the crushing force will be at the minimum and, as a result, the stroke will also be at its minimum. The smaller the stroke, the greater the deceleration. The crush force of the energy absorbing material must be low enough to maintain the minimum vertical stroke distance in that situation. Assuming an impact lander mass of 1368kg, the expected mass kinetic energy (KE) in the system 814J. This energy must be absorbed upon impact to obviate any unwanted landing dynamics. To achieve this goal, the crushable material would need to be tuned to a crush strength of .552 MPa for the current leg geometry. This can be achieved with material selection and crush pad geometry. The crushable material will need to be characterized for the surface temperatures. The other extreme landing scenario is landing with only one leg absorbing the impact KE. The crush strength of the energy absorbing material is set by the four leg case. The total height of the crush pad is set by the one leg case. The resulting compression of the crushable material in the one leg case results in a vertical stroke distance of 4.1cm. Typically the crushable material is linear through 70% crush distance so the crush pad should be approximately 5.8cm thick to account for both extreme situations. These calculations are very simplistic for the purpose of demonstrating theoretical viability of the concept. Actual implementation will need to be much more rigorous.

The CM location changes very little in the height direction as the fuel is used up due to the CM of the propellant being near the CM of the dry system. The basic tip-over geometry was briefly studied and the current leg configuration works well. The CM location and the minimum stance width of the landing legs show the lander is statically stable to 49 degrees. This is an idealized, simple turn-over calculation that assumes the kinetic energy at impact can be fully absorbed by the legs and crushable material. A full and detailed landing dynamics analysis will need to be performed.

The structural elements developed for this study are well within the current state of the art and have extensive heritage. All manufactured structural elements are expected to be TRL6

3.11 Thermal

The thermal environment of the CORAL mission varies from near Earth, following launch and again during the Cruise Phase flyby, to the coldest part in the later Cruise phase and the approach/landed operations where there is negligible environmental heating from the sun (Figure 3-18). Minimizing electrical heater power is key to operations in this environment as the 2 RTGs power reduces significantly through the early phases of the mission's lifetime (Figure 3-19).

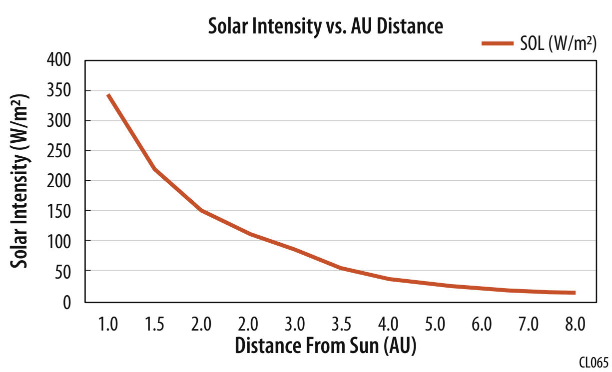

Figure 3-18: Solar Intensity vs AU Distance

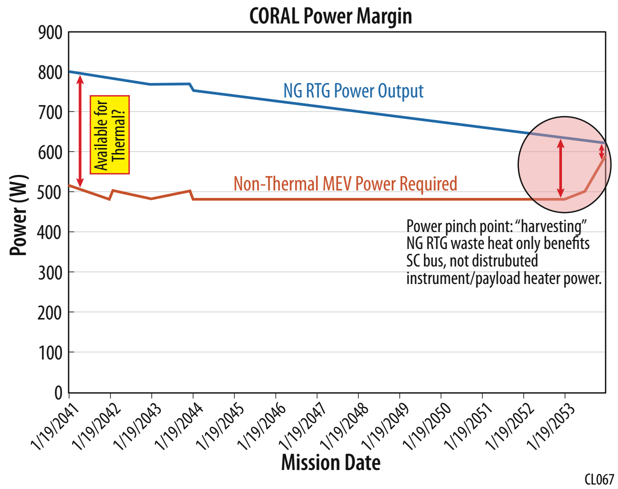

Figure 3-19: CORAL Power Margin over time

The baseline thermal control approach (Figure 3-20) is to passively control the bus to minimize the need for heater power by a) packaging as much of the temperature sensitive equipment as possible, within the SC bus to allow the dissipative heat generated to be distributed and shared throughout, maintain 0°C to 30° within this Warm Electronics Module (WEM), b) minimizing radiator area by using louvers to "close" during colder environments (most of mission).

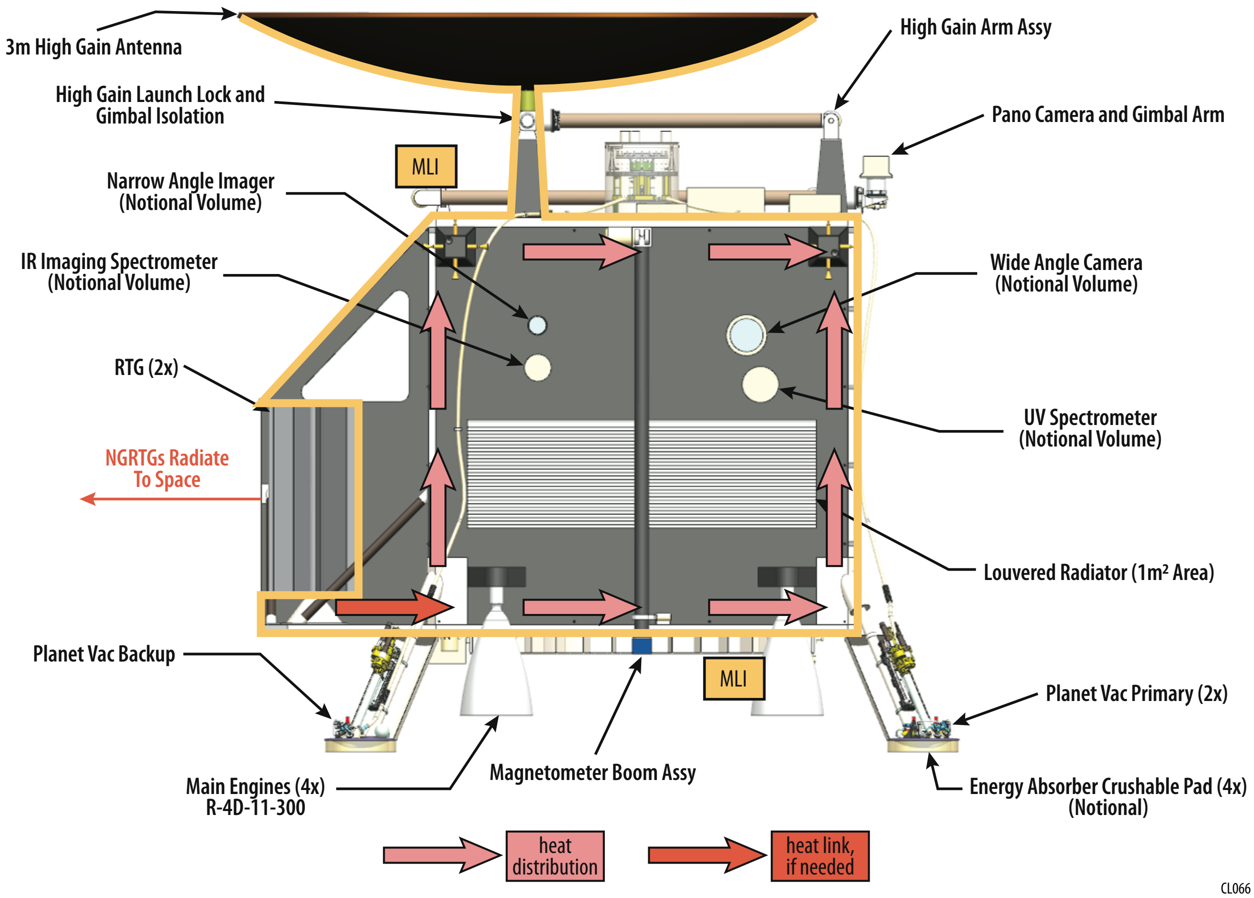

Figure 3-20: Bus thermal design

The propulsion subsystem, having the least tolerance to cold temperatures, benefits from this approach with the tanks located in the center of the bus and requiring direct heating only for the lines and valves near the thrusters. This approach does require a conductive panel design with inter-panel heat pipe connections to iso-thermalize the bus. Minimizing the size of radiator needed and using louvers to "close" in colder environments greatly reduces bus heater power needed.

The external honeycomb panels (aluminum facesheets and core), with embedded Constant Conductance Heat Pipes (CCHP) will allow the dissipative heat loads to be spread efficiently within and between the 6 external panels. A thermal joint will be designed to provide structural and heat pipe connections to be made during assembly as shown in Figure 3-21.

This mission has the advantage of two 16-GPHS STEM-RTGs, located on the external "patio" of the SC bus, that generate large amounts of waste heat. In order to substantially reduce the bus heater power required, the RTGs are linked using a heat pump to transfer heat. This heat load will need to be modulated between hot and cold conditions, using Variable Conductance Heat Pipes (VCHP) from the source to the bottom deck and its CCHP heat distribution network. The heater power needed during transit is thus 0 W with heater powered needed in the landed configuration being 70 W for the exterior Landed Payload and HGA Gimbal and PanCam & gimbal.

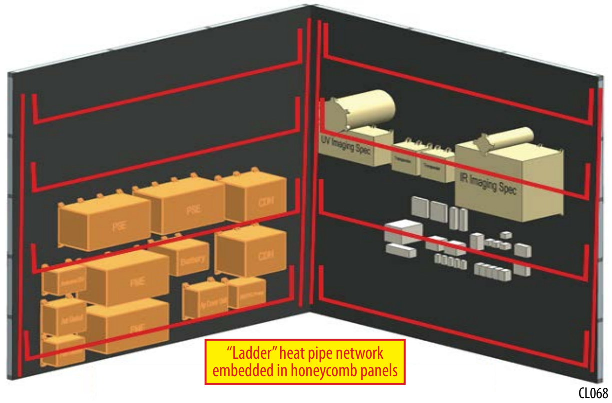

Figure 3-21: Thermal Concept (typical) for Heat Pipe Distribution Network