CROCODILE

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

Cryogenic Return Of Cometary Organics, Dust, and Ice for Laboratory Exploration

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Contents

APPENDICES

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Executive Summary

The Cryogenic Return Of Cometary Organics, Dust, and Ice for Laboratory Exploration, or CROCODILE mission concept study was to determine the feasibility of cryogenic comet sample return within the scope of a NASA New Frontiers (NF) class mission within the next decade. This study was conducted by Goddard Space Flight Center.

The overall science goal of the CROCODILE mission is to assess the elemental, isotopic and structural composition of the volatile, organic and inorganic components of a comet nucleus to: understand the compositional reservoirs present in the early Solar System; address the role of comets in the delivery of water and organic molecules to the early Earth, terrestrial planets and satellites; and gain insight into the evolutionary processes spanning from the protoplanetary disk to current cometary activity., In order to accomplish this goal, CROCODILE will orbit a Jupiter family comet, map the comet nucleus and select an optimal sampling site, sample the nucleus below the surface, and return cryogenically preserved samples to Earth for laboratory analysis.

To meet the science requirements of this concept, the CROCODILE payload is comprised primarily of the harpoon sampling system, the long-term storage cryogenic sample holder, a suite of OSIRIS-REx (heritage) infra-red imaging spectrometers and cameras, and the Earth Entry Vehicle. A number of trades were considered for this study to ensure the cryogenic sample is returned unaltered for sample analysis in laboratories around the world for generations. The key decision was that a cryogenic sample return temperature of 120 °K was selected for preservation of amorphous water ice and entrained volatiles, as well as water-soluble organics and salt. This temperature choice was also guided by results from the ESA/Rosetta mission that suggested surface materials within the top 20 cm never fall below 120 °K during an orbital cycle of comet 67P/Churyumov–Gerasimenko (hereafter 67P). Comet 67P was the target optimized for this study both due to the range from Earth and its previous characterization by Rosetta. Other trades considered for this study included the sample depth, number of samples, long term sample storage and number of coolers and sample collection mechanism.

The CROCODILE mission baseline launch readiness date is May 2036 on a Falcon Heavy Recoverable. The spacecraft will cruise for ~5.7 years to rendezvous with 67P in Jan 2042 where it will spend its 4-year survey and sample collection campaign at the target. Return to Earth will occur after another ~5.7 year cruise for intercept and Earth entry/recovery in Nov 2051. The EEV and sample recovery phase have been optimized with lessons from previous sample return missions and anticipate a maximum 3 hour post-landing recovery effort where the EEV will be opened and samples transferred to a cryo-fridge for final transport to the curation facility.

The total costs estimated for Phase A-D, including the full instrument payload, is $1.9B (FY25). These estimated costs include 50% reserves with high heritage camera/spectrometer suite as well as analogous orbiter, mission operations, and ground systems (primarily from OSIRIS-REx). While this study's cost exceeds guidelines provided of $1.1B (FY25) for the New Frontiers level, considerations for technology development would enable cryogenic sample return in a more timely manner. Such technologies that were identified from this study include: cryocooler reliability; solar array efficiency; sample mass confirmation; and cryogenic sample and storage systems. Cryogenic sample return offers an unprecedented scientific return and has been identified as a priority in small body science for multiple decades. Focused technology development investments would provide essential support to enable this high-priority mission to be accomplished at the New Frontiers class in the future.

1.0 SCIENCE OBJECTIVES

Science Justification

Jupiter family comets are the best repositories of materials that record the history of the earliest stages of our solar system formation and evolution that are accessible for sample return. With the right collection of cometary samples, planetary scientists can address fundamental questions including: the nature of the initial compositions of solids, organics and gases in the solar system; the inventory of inherited materials from the proto-solar molecular cloud that retain signatures of their extrasolar origin; the extent of nebular transport of gas and dust during the initial stages of planetesimal formation; how comets themselves formed and what drives their activity; and the degree to which comets contributed to the delivery of water and pre-biotic organic matter to the early Earth. Laboratory analyses, in combination with remote sensing and *in situ* measurements at the comet, are highly desirable for connecting the cometary materials properties to their formation and processing histories. While remote sensing and *in situ* measurements provide macro to global scale constraints on comet properties, laboratory analysis of returned samples enables a much greater range of potential measurements, and in many cases, orders of magnitude greater precision for key quantities (Figure 1-1).

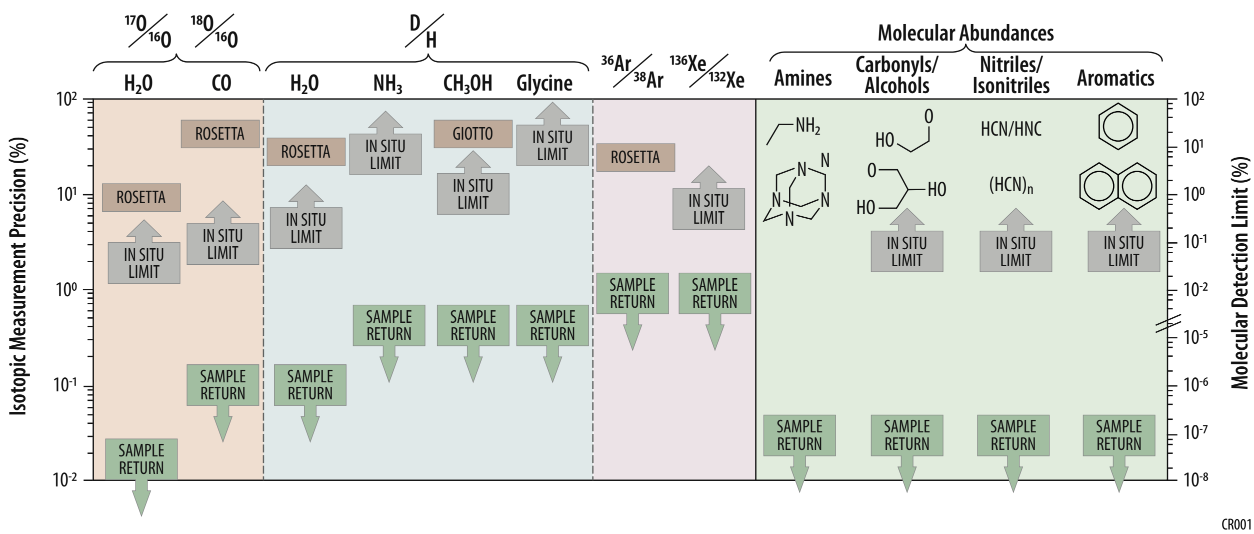

Figure 1-1: Volatile measurement capabilities for returned samples are superior to those for in situ spacecraft. Here examples for a comet are shown Left: isotopic measurement precision for selected volatiles. Right: molecular detections for selected classes of organics. Reprinted from Milam et al., (2020).

The range of questions that can be addressed with a comet sample return mission depends largely on the depth from which samples are retrieved, and the temperature history of the samples on the comet through sampling, delivery to Earth and subsequent curation. Two regimes of such return missions have been studied previously: comet surface sample return (CSSR) and cryogenic comet nucleus sample return (CCNSR). Surface sample return that includes dust grains, and refractory organic matter, but no intact ices, has been extensively validated through a preliminary mission concept study for the 2013-2022 Visions and Voyages Decadal Survey (V&V 2013), and three independent mission concepts competed in New Frontiers IV, from which the CAESAR mission concept was selected for a Phase A development. Cryogenic return, at temperatures sufficient to return H_2O ice, and potentially CO_2 and HCN ice (see Table 1-1) was studied for V&V 2013 as a potential Flagship mission, but ultimately deemed too technologically risky for immediate recommendation. With the return of intact ices along with dust and organic matter, a cryogenic mission could provide researchers unprecedented opportunities to address the questions of the comet formation modes, origins of comet activity, and the extent of the exogenous delivery of water and prebiotic chemistry on Earth by comets. Either a CSSR or CCNSR mission would represent a dramatic improvement over the cometary sample collections available for laboratory study today. These collections comprise only the most physically and thermally robust fractions of cometary coma dust are altered to varying degrees by atmospheric entry, terrestrial contamination or collection artifacts. These samples do not permit study of volatile species, labile organics, or the spatial relationships among these components that reflect the physical and chemical interactions among the different cometary materials.

Identification of Temperature Requirements for Specific Cometary Ices

Table 1-1: Ice retention rates in vacuum after two years, from V&V 2013 Comet Nucleus Sample Return Mission Concept Study

Temp (K)	H_2O (bulk)	CO_2 (bulk)	CO (bulk)	HCN (bulk)	H_2O (granular)	CO_2 (granular)	CO (granular)	HCN (granular)
150	0.969	0.000	0.000	0.000	0.000	0.000	0.000	0.000
125	1.000	0.000	0.000	0.000	0.999	0.000	0.000	0.000
100	1.000	0.003	0.000	0.978	1.000	0.000	0.000	0.168
90	1.000	0.987	0.000	1.000	1.000	0.404	0.000	0.984

In order to return substantial fractions of a sample containing granular water ice, collection and storage temperatures of 125 °K or below are required. Significant amounts of HCN and CO_2 ice species would be retained only if the collection and storage temperatures stayed below 100 °K and 90 °K, respectively. If collection temperatures of 60 °K could be achieved, all four expected major ice species, including CO, could be retained (Westphal et al., 2020). For this study, which sought to address the question of whether cryogenic return of organics, dust and water ice is now viable within a mid-level (New Frontiers) cost cap, we focused on the temperature range of 100 °K to 120 °K. This range retains the petrographic arrangement of granular H_2O ice, refractory and labile organic matter, and also potentially some volatile species, if held in vesicles with the water ice grains. In addition, this temperature range is amenable to the use of commercially available, or readily customizable, laboratory instrumentation for sample processing and measurement, such as liquid N_2 stages. Thus, such temperatures do not present a major cost or technology barrier for the sample curation and laboratory analyses, whereas handling of samples at 60 °K would require liquid He or other cryocooling technology not currently easily integrated into the planetary materials analysis facilities.

Thermal Modeling

The relative amounts of the various ice species that would be present in a given collected sample depend on the depth of collection, and thermal depth profile of the target site on the specific comet through its diurnal and orbital cycles. In order to calculate how the temperature and ice abundance change with latitude, depth under the surface, and time, it is necessary to apply a thermophysical model. We used the "Numerical Icy Minor Body evolUtion Simulator", or NIMBUS, developed by Davidsson (2021). NIMBUS considers solar heating, thermal cooling, heat conduction, sub-surface sublimation, gas diffusion, advection, recondensation, and outgassing of H_2O and CO_2, dust mantle formation, and erosion. This model can be applied to any JFC; for this study we restricted the modeling to the most well studied JFC, Comet 67P/Churyumov-Gerasimenko (Davidsson et al., 2021b).

Focusing on the northern hemisphere (available for CROCODILE sampling), Davidsson et al. (2021b) find that CO_2 typically is located ~4 meters below the surface. Below that depth, the material has never been heated above 120 °K. Therefore, the retrieval of $T < 120$ °K material that contains super-volatiles like CO_2 requires drilling to substantial depths, that can only be afforded under a Flagship budget. Davidsson et al. (2021b) show that the dust mantle thickness is typically ~0.02 m, which means there is abundant H_2O ice in the ≤ 0.3 m region. The peak temperature experienced at 0.2-0.3 m during an orbit depends largely on the exact thickness of the dust mantle and its diffusivity. Using parameters that best reproduce the inbound water production rate curve measured by Rosetta/ROSINA, Davidsson et al. (2021b) find that the temperature at a depth of 0.2-

0.3 m may reach as high as 190 °K. However, if the mantle locally has higher diffusivity, or if it is unusually thin, alternative modeling by Davidsson et al. (2021a) shows that lower peak temperatures of 160 – 180 °K may be applicable. If the material near the very surface contains H_2O ice, the peak temperature is near 200 °K, while a dry dust mantle becomes substantially warmer, up to 260 °K (Davidsson et al. 2021a,b). In summary, the ice-rich material available for near-sub surface sampling (≤ 0.3 m depth) has been cycled up to temperatures in the 160 – 190 °K range, while a small fraction of the material that constituted the dust mantle may have reached 200 – 260 °K. Sample collection operations should be performed at a point in the orbit where the entire sample will be cold (as low as 140 – 160 °K if made near aphelion). Keeping the retrieved sample in the 100 – 120 °K range during return therefore guarantees that it is being preserved below the sampling temperature, and far below the peak temperatures experienced by the material in its natural environment. Because this material has been mostly devolatilized by exposure to temperatures above 150 °K, with only minor amounts of volatiles potentially retained in vesicles in the H_2O ice, hermetic sealing of a cryogenic sample container is possible without risk of excessive degassing and over-pressurization.

Implications of Comet Regolith Strength Estimates on Viable Sampling Mechanisms

CROCODILE addressed the state of the knowledge of cometary surface regolith strength and the implications of this information on the viable choices for cryogenic sample collection. Since the V&V CCSNR study, new data from the Rosetta Philae lander has become available (O'Rourke et al. 2020). These data indicate an upper bound of 12 Pa for the compressive strength of the surface material on 67P at touch down site 2 (TD2), determined from the depth of the lander impression on the 67P surface. Estimates from a combination of measurements and modeling of the collapse of cliff overhangs and scratches produced by Philae, indicate internal strengths of < 150 Pa. These estimates are significantly lower than the prior values, 1 kPa (TD1) and 2 MPa (TD3), from the Rosetta MUPUS instrument, which experience some deployment uncertainties (O'Rourke et al. 2020). Additional confidence for the lower compressive strengths comes from the OSIRIS-REx (O-REx) spacecraft touch-and-go interaction with the surface of Bennu. Whereas regolith strengths of 100s to 1000s of kPa used for the prior CCNSR mission concepts would restrict the sampling strategy to a lander and drill architecture, strengths of < 1 kPa are amenable to ballistic, "shoot and go" sampling strategies, such as the "harpoon" chosen for CROCODILE.

Sample Mass Requirements

Although laboratory instrumentation is ever improving, there are some fundamental and practical limits on the minimum required mass of the returned sample. For example, in order to place robust constraints on models of the effects of self-shielding mechanisms on oxygen gas in the outer solar nebula, a macroscopic sample of H_2O ice is required, of order 1 g or more. To enable adequate sample mass for multiple types of measurements, including abundances of entrained noble gasses, radiogenic dating of minerals, and isotope-specific analysis of labile organics, in addition to meeting long-term sample curation guidelines, sample masses of order 100 g or more are desirable. For returned masses above 100 g, the greatest science returns would come from sampling multiple target sites to address regolith heterogeneity.

Science Payload Instruments

In order to achieve the cryogenic return of organics, H_2O ice, and dust for laboratory exploration at less than Flagship mission cost-levels, CROCODILE restricted the science instruments to the minimum required to map the comet surface distribution of organics, dust and ice components, collect and cryogenically store the sample, and provide pre- and post- collection characterization of the sampling site. Two infra-red imaging spectrometers, based on the OSIRIS-REx mission OVIRS and OTES instruments were specified with imaging camera suite OCAMs and lidar, along with a sampling system consisting of two tethered harpoons, two hermetically sealed, passively cooled sample canisters, and a two-compartment actively cryo-cooled sample storage refrigerator. The full camera suite would be used to optimize ROIs on the surface for site selection and characterization.

The lidar will provide a measure of surface roughness as well as inform on site selection by helping to quantify potential hazards in a more detailed fashion down to cm level with respect to elevation resolution. The harpoon sheaths would be instrumented with accelerometers and temperature sensors to record the sampling depth and temperature profile. A sample camera will also be used for sample confirmation. Science questions beyond spectroscopic mapping of the comet surface would be addressed through laboratory analysis of the returned samples for multiple generations.

Table 1-2: CROCODILE Science Traceability

Science Objective	Measurement	Implementation	Functional Requirement
What are the sub-surface components of Jupiter Family Comets (JFCs)?			
Determination of the nature of pristine ice, gas, and dust grain components in JFCs	Return cryogenically preserved sub-surface comet nucleus samples from a depth that ensures minimal thermal and radiological processing	Sampling system & Sample Dewar, IR camera/spectrometer suite, lidar	Optimize site selection, including characterization of surface roughness and geologic context for sample collection with presence of ice/volatiles organic material. Retrieve nominal 100 g (> 1 g) sample from depth of 20 cm to 30 cm. Maintain sample temperature < 120 °K from sampling event to delivery to curation facility
What inherited components from the pre-solar interstellar medium are preserved in JFCs?			
Evidence for photochemical self-shielding	C,H,O,N Isotopic compositions of H_2O ices and entrained volatiles	Earth-based laboratory analyses	Sample return temperature 120 °K
Evidence for cold molecular cloud chemistry	C,H,O,N Isotopic compositions of labile organics		Sample return temperature 120 °K
Relationship of pre-solar silicates, and GEMS to H_2O ice and organic matter	O isotope composition of silicates in situ in H_2O ice		Sample return temperature 120 °K
What determines the activity of JFCs?			
Dust/ice interaction driven activity	mass ratio of dust grains to H_2O ice and spatial distribution on surface and within sample	IR Camera/spectrometer suite; Earth-based laboratory analyses	Sample return temperature 120 °K, map surface to determine ice/volatile presence, mineral composition
Ice sublimation driven activity	physical characteristics (grain size, porosity) of H_2O ice, surface composition	IR Camera/spectrometer suite	Sample return temperature 120 °K, map surface to determine ice/volatile presence, mineral composition
Noble gas driven comet activity	elemental abundance of noble gases (NGs) entrained in H_2O ice	Earth-based laboratory analyses	Sample return temperature 120 °KK
What reservoirs of nebular materials / signatures of protoplanetary disc processes are preserved in JFCs?			
Transport, chronology in early solar system; preservation of early stages of accretion	spatial relationships of refractory nebular components (chondrules / CAIs, silicates, other minerals) and H20 ice	Earth-based laboratory analyses	Sample return temperature 120 °K
Differences between accreted grain components in comets and asteroids	spatial relationships of refractory nebular components to presolar / interstellar grains and water ice		Sample return temperature 120 °K

How did JFCs contribute to the water, organic molecules and volatiles delivered to the early Earth?			
Cometary water to early Earth	H_2O isotopic compositions of water ice	Earth-based laboratory analyses	Sample return temperature 120 °K
Prebiotic compounds	Abundance of "semivolatiles", e.g., ammonia salts, soluble organics (sugars, alcohols, amino acids)		Sample return temperature 120 °K
Fraction of cometary water delivered to early Earth in mineral/ dust grains	Abundance of hydrated minerals		Sample return temperature 120 °K
Volatiles on surface of early Earth	Isotopic compositions of NGs entrained in ices: water		Sample return temperature 120 °K
Maintain the returned samples in a curation facility without degradation			
Curation of pristine comet nucleus samples for community analysis using advanced laboratory instrumentation	Isotopic compositions of NGs entrained in H_2O ices	Curation facility environmental monitoring	(Non-S/C) Augmented JS/C curation facility to include cryogenic storage and processing for 120 °K samples.

2.0 HIGH- LEVEL MISSION CONCEPT

Overview

CROCODILE will orbit Comet 67P/ Churyumov-Gerasimenko,a Jupiter family comet last visited by ROSETTA in 2014, perform a survey to map its nucleus, extract two, ~100g, core ice samples ~25 centimeters below the surface, cryogenically preserve the samples and return them to Earth for laboratory analysis. An overview of the Concept of Operations (ConOps) by mission phase is depicted in Figure 2-1. Key driving requirements for design implementation to meet the science objectives involved the mission design trajectory, sample collection method, sample cryogenic temperature storage and returning the samples to Earth. Leveraging off previous similar mission studies, a series of trades were conducted to close the mission design, given spacecraft mass, propulsion (Δ-V) and power constraints, resulting in selecting the NEXT-C Ion Solar Electric Propulsion (SEP) system with a 50kW Roll-Out Solar Array (ROSA). A tethered “harpoon” sample collection system was chosen to eliminate the complexities and risks associated with physically landing on the comet surface. This method, known as “Shoot and Go” or SAG, will allow the spacecraft to safely hover 5-10 m above the comet’s surface and extract a viable core ice sample. The samples will be stored in Cryogenic Sample Dewars, which are annular containers containing solid argon. Two-stage Stirling cryocoolers will maintain the sample Dewars at 60 – 80 °K in a Sample Storage Refrigerator during the long term cruise. The sample Dewars will be returned to Earth for curation via an Earth Entry Vehicle (EEV), selected for its design heritage with the Stardust and OSIRIS-REx sample return capsules.

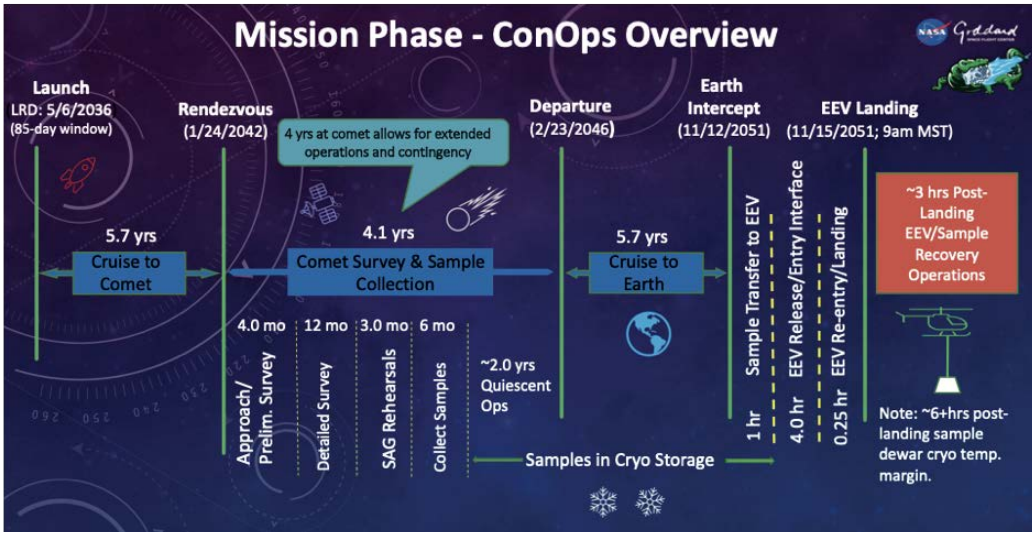

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Figure 2-1: CROCODILE Mission Overview

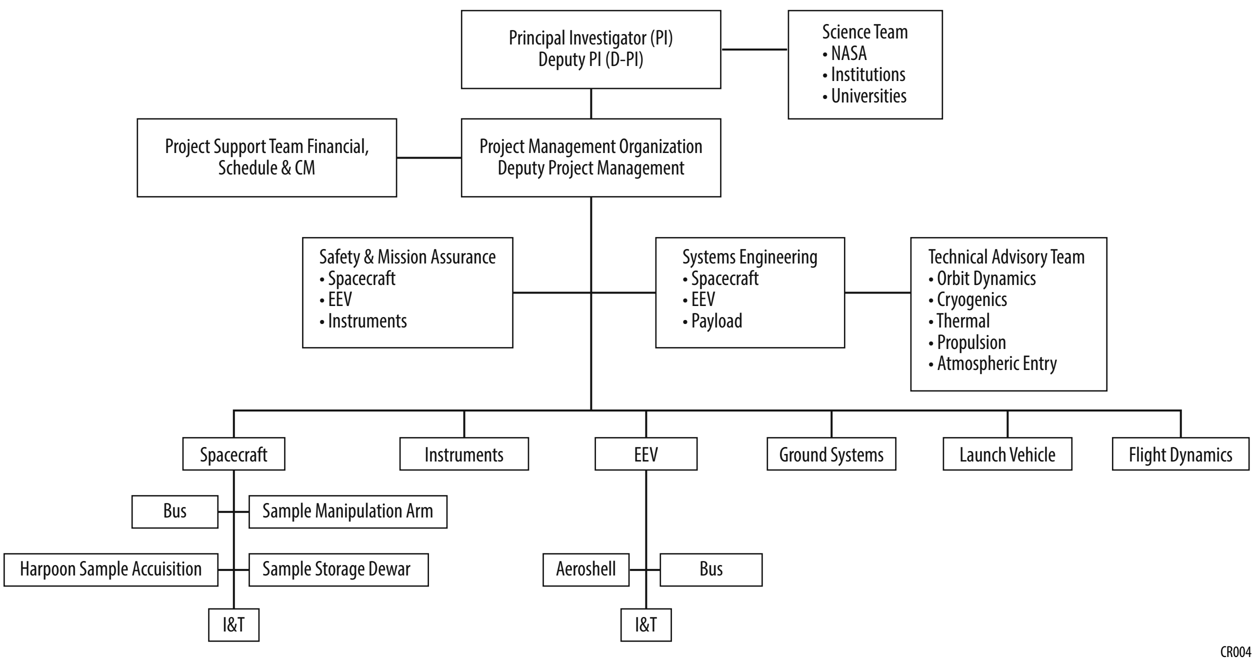

Figure 2-2: Notional CROCODILE Organizational Chart

Figure 2-2 illustrates a notional Organizational Chart assuming CROCODILE is implemented as a typical New Frontiers mission led by a PI with support from NASA centers, universities and industry. As typical with PI led missions the PI has overall authority and accountability for mission success and is supported by a deputy PI, a science team as well as project management and engineering staff. A cryogenic expert is recommended to be part the overall Technical Advisory team as strict and careful oversight of the cryogenic implementation is critical to mission success. Any organizational chart must show clear lines of responsibility and authority and this is especially crucial for a complex mission such as CROCODILE. The design concept relies on no foreign instruments or contributions.

Concept Maturity Level

The CROCODILE concept is consistent with a Concept Maturity Level (CML) 4 as defined in Table 2-1, with some aspects at CML 5. A point design is provided for the full end-to-end mission concept, which includes defined spacecraft subsystem hardware and mission concept of operations. A detailed Master Equipment List (MEL) has been created listing spacecraft subsystem mass and power by individual component. A mission cost has been generated based on the MEL or analogous hardware, where appropriate, and the top risks associated with the mission have been identified. The CML 5 level heritage for much of the spacecraft has been identified in the Design Report Appendix.

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

Technology Maturity

The CROCODILE mission concept relies on technology maturity to reduce mission risk and cost. Utilizing SEP in parallel with operational cryocoolers at solar distances up to 6 AU requires large solar arrays, which adds a layer of complexity to the design concept and is a significant spacecraft bus cost driver. Finding ways to reduce the mission electrical power needs would facilitate implementation. The extended mission duration also raises the need for reliable cryogenic hardware and the harpoon overall thermal design and performance needs to be understood with respect to cryogenic requirements. Table 2-2 summarizes the recommended areas of technology development. A specific TRL development for the cryogenic sample solution presented in this study is found in the TRL development section.

Table 2-2: Recommended Technology Maturity Areas

Technology	Mission Driver
Cryocooler Reliability	Reliable cryocooler operation over the full 15.5-year mission duration.
Small mass and volume sample measurement	Confirmation of a sample as small as 1 g has been retrieved.
Cryogenic Sample acquisition and Storage	• Cryogenic penetrator mechanisms, depth assurance, heat transfer sampling and retrieval systems design maturation including • Long term cryogenic storage assemblies and mechanisms design and test • Long life automated cryogenic seals • Mature rapid automated storage and transfer stages • Tether and tether mechanism dynamics at cold temperatures and rapid automated procedures
• Increased electrical efficiency • Solar Arrays • SEP Systems • Cryocoolers	Reduced power needs and less complicated and costly electrical power system

Key Trades

A variety of trades and decision points were made during the iteration of the design implementation and concept of operations in order to close on a viable mission architecture. The trade selections were guided by the required science objectives as well as associated risk and cost impact to the overall mission. The key trade studies performed are listed in Table 2-3 below.

Table 2-3: Key Trade Studies

Element	Key Trade Study/Result	Rationale
Target Comet	79P vs **67P**	Range from Earth and achieves required mass margin .
Sample Storage	Return entire cryogenic storage refrigerator to Earth vs **transfer samples to EEV**	Maintain O-REx and Stardust EEV heritage without significant increase to mass, cost & complexity; Argon Sample Dewars less complex/adequate
Cruise Propulsion System	Chemical vs **SEP**	Out-performs chemical-based propulsion
Sample Collection System	Drill, Scoop or **Harpoon**	Reduces complexity/risk of landing Can achieve sample depth with harpoon
# of Samples	1 vs **2**	Redundancy/Risk Reduction
Post landing EEV Cryo Storage	Active cooling vs **Passive**	Argon Sample Dewars passively maintain temperature requirements with significant margin on expected sample recovery time

3.0 TECHNICAL OVERVIEW

Science Instrument Payload Description

The science instrument payload is similar to the 2013 Comet Surface Sample Return study and primarily based on the instrument suite flown on the OSIRIS-REx mission. These heritage instruments will be reviewed to make required investments in the engineering modifications that will improve its capabilities for the mission. This science payload is comprised of a visible and infrared spectrometer modeled after OVIRS, thermal emission spectrometer modeled after OTES, a suite of cameras based on OCAM, Cryogenic Sample Dewars, Lidar and sample acquisition harpoons. Payload mass and power values are summarized in Table 3-1.

Table 3-1: CROCODILE Science Payload Mass and Power Table

	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Thermal Emission Spectrometer	6.3	30	8.2	10.8	30	14
Visible and Infrared Spectrometer	17.7	30	23.0	8.8	30	11.4
PolyCam	9.1	15	10.5	17.9	30	23.3
MapCam	3.8	15	4.4	18	30	23.4
SamCam	2.8	15	3.2	15	30	19.5
Cryogenic Sample Dewars (Total)	12.8	30	16.6	4	30	5.2
Lidar	20	30	26	70	30	91.0
Harpoon Sample Acquisition	13.5	30	17.6	36	30	46.8
Total Payload Mass	**86.0**	**27**	**109.5**			

The Thermal Emission Spectrometer (TES) is a compact, Fourier-transform interferometer that operates over the spectral range from 5.7 to 100 microns. It will conduct surveys to help determine the mineral composition and temperature distribution of the comet for global maps and local candidate sample-site areas.

The Visible and InfraRed Spectrometer instrument provides spectra over the wavelength range of 0.4 – 4.3 microns. It will measure visible and near infrared light from the comet and work in tandem with the Thermal Emission Spectrometer to produce full disk spectral data and global spectral images of the comet surface. These images will help guide in the sample site selection.

The visible camera suite is made up of three separate cameras: PolyCam (high resolution), MapCam (medium resolution) and SamCam (low resolution) cameras. The high-resolution images will be acquired during Detail Survey to identify gas or dust plumes, potential surface hazards in the region of interest and provide input to improve resolution of the comet shape model. The medium resolution camera will provide mapping of the comet nucleus surface to assess geomorphological variation and geologic context for the sampling site. Survey and mapping details are discussed further in the Mission Concept of Operations section. The SamCam is a close-range, wide-angle imager used to monitor and provide context imaging during reconnaissance passes and to document the SAG process.

The cryogenic sample dewars are two storage containers that provide passive cooling for the acquired comet sample during the Earth Entry and Recovery operations. This will help enable the mission to preserve bulk amorphous and porous granular water ice of the comet samples. In addition, temperature and pressure sensor are attached to dewars to document and provide historical data of the hermetic seal during the mission.

More information can be found about the harpoons and Cryogenic Sample Dewars in the Flight System section.

The instrument payload parameters are contained in Table 3-2. Figure 3-1 depicts the instrument suite on the payload deck.

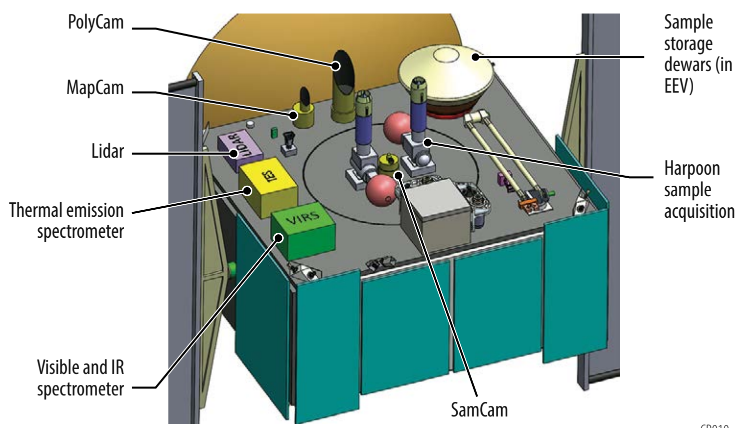

Figure 3-1: Instrument Payload Layout

Table 3-2: CROCODILE Science Instrument Payload Table

Item	Thermal Emission Spectrometer	Visible and Infrared Spectrometer	PolyCam	MapCam	SamCam	Cryogenic Sample Dewar	Lidar	Harpoon System
Type of Instrument	Spectrometer	Spectrometer	Narrow Angle Imager	Medium Range Imager	Wide Angle Imager	Pressurre, Temperature Monitor	Laser Range-finder	Sample Acquisition
Size/dimensions (m x m x m)	0.37 x 0.28 x 0.52	0.90 x 0.36 x 0.20	0.20 x 0.20 x 0.45	0.15 x 0.15 x 0.20	0.15 x 0.15 x 0.15	0.17 x 0.17 x 0.45	<.1 m^3	0.78 cm x 0.25 x 1.0 (ea)
Instrument mass without contingency (Kg, CBE*)	6.27	17.7	9.1	3.8	2.8	12.8	20	13.52
Instrument mass contingency (%)	30	30	15	15	15	30	30	30
Instrument mass with contingency (Kg, CBE+Reserve)	8.15	23	10.47	4.37	3.22	16.6	26	17.58
Instrument average payload power without contingency (W)	10.8	8.8	17.9	18	15	4	70	36
Instrument average payload power contingency (%)	30	30	30	30	30	30	30	30
Instrument average payload power with contingency (W)	14	11.4	23.27	23.4	19.5	5.2	91	46.8
Instrument average science data rate without contingency (kbps)	22	181	17.9	17.9	17.9	N/A	N/A	N/A
Instrument average science data rate contingency (%)	30	30	30	30	30	N/A	N/A	N/A
Instrument average science data rate with contingency (kbps)	29	235	23.3	23.3	23.3	N/A	N/A	N/A

Flight System

Flight System Overview

The Spacecraft consists of the instrument payload, the spacecraft's bus subsystems, and unique flight hardware for the collection, storage and return to Earth of the comet ice samples. Table 3-3 and Table 3-4 list the flight system's mass and power estimates, and flight elements parameters and an overview of the spacecraft concept is shown in Figure 3-2.

The spacecraft bus is of a typical design and function for planetary missions with specific accommodations made for the SEP system to include 50kW Roll-Out Solar Arrays (ROSA) necessary for mission operations. Reference the Design Report Appendix for additional S/C bus information. The unique sample acquisition and storage system includes the Cryogenic Storage Refrigerator, sample handling arm, harpoon sample system, cryogenic sample Dewar, and the Earth Entry Vehicle (EEV). Descriptions of the cryogenic sample chain that are the focus of this study follow.

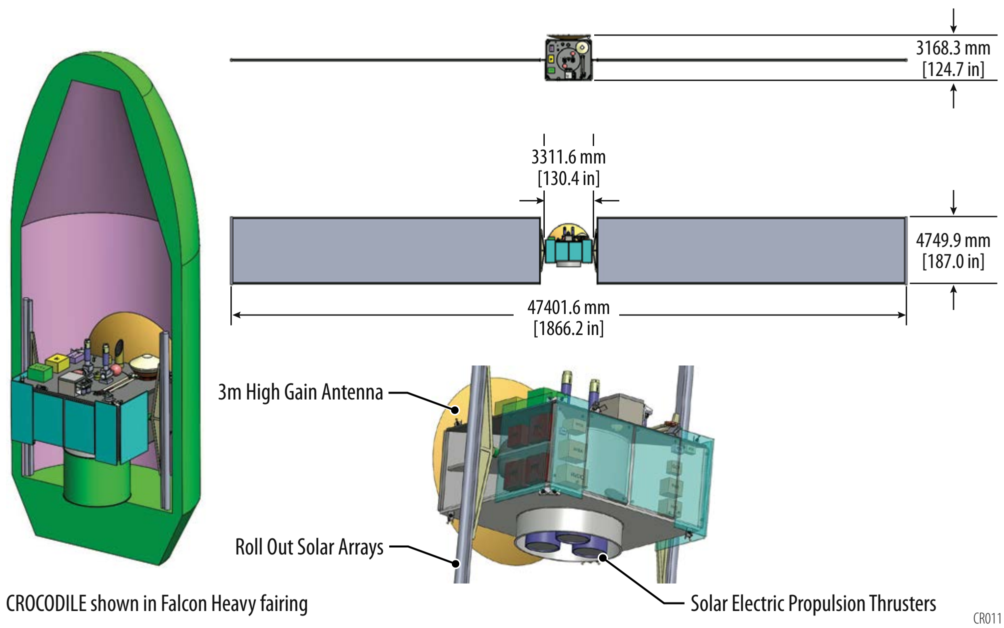

Figure 3-2: CROCODILE Spacecraft Concept

Table 3-3: CROCODILE Flight System Mass & Power

	Mass			Average Power		
	CBE (kg)	% Cont.	MEV (kg)	CBE (W)	% Cont.	MEV (W)
Structures & Mechanisms	449	30	610			
EEV	78.0	30	101.4	7.5	25	9.3
Sample Collection System (sans payload harpoons)	44.5	30	57.9			
Sample Handling Arm	44.0	30	57.2	77.0	30	100.1
Cryogenic Storage Refrigerator (includes solid Ar)	112.5	16	130.8	135.0	25	168.8
Electrical Subsystem	417.7	30	543.0	18.0	30	23.4
Attitude Control Subsystem	54.2	5	56.9	40.9	5	42.0
Laser Altimeter	0.10	30	0.13	2.0	30	2.6

Avionics Subsystem	143.3	26	180.2	386.4	30	502.3
Telecommunications	39.5	7	42.4	168.8	30	219(1)
Thermal Control Subsystem	138.5	30	180.2	142.3	30	185
S/C Harness	60.0	30	78.0			
Electric Propulsion (dry)	266.6	7	284.9	6982	10	7680(2)
Chemical Propulsion (dry)	100.7	10	110.7	10.5	15	12
Total Flight System Dry Mass	1948.5	25	2433.8	6992.5		7692

Notes:

1) Transmit power

2) Avg of 640 W (1 thruster lowest power setting) and a max of 14,720 W (2 thrusters at max power setting)

Table 3-4: Flight System Parameters

Flight System Element Parameters (as appropriate)	Value/ Summary, units	Value/ Summary, units
General	Spacecraft	Earth Return Vehicle (EEV)
Design Life	15.5+ years	100 hrs (release to gnd recovery operations)
Structure		
Structures material (aluminium, exotic, composite, etc.)	Primary - M55J Composite, Ti-6Al-4V ; Secondary : Al-6061-T6, Honeycomb Panels w/M55J Facesheets & Al Cores,	Al Honeycomb w/Composite Facesheets,; Al foam; TPS: 3MDCP Heat Shield & Phenolic-Impregnated Carbon Ablator (PICA)
Number of articulated structures	4x Joint/DoF Sample Handling Arm w/end effector	1x lid hinge/latch, 2x sample Dewar hold down latch
Number of deployed structures	Roll Out Solar Array (ROSA) wings (x2); Tethered Harpoon Sample Systems (x2)	Drogue/Main Parachute landing system
Aeroshell diameter		1.1 m
Thermal Control		
Type of thermal control used	MLI, coatings, radiators, heaters; Solid Argon cryogen tanks & annular dewars	MLI, Al foam, TPS Ablator
Propulsion		N/A
Estimated delta-V budget, m/s	190 m/s (Bi-Prop)	
Propulsion type(s) and associated propellant(s)/oxidizer(s)	SEP/ NASA Evolutionary Xenon Thruster (NEXT-C) & MMH/NTO Chem Bi-Prop (for prox ops)	
Number of thrusters and tanks	16 ACS thrusters for 3 axis translation control w/4x tanks; Active NEXT thrusters (x2) + redundant (x1) for Delta V w/7x tanks;	
Specific impulse of each propulsion mode, seconds	SEP Isp: 1500 - 4100 s; Bi-prop Isp: 305 s	
Attitude Control		
Control method (3-axis, spinner, gravity-gradient, etc.).	3-Axis Stabilized	Ballistic
Control reference (solar, inertial, Earth-nadir, Earth-limb, etc.)	Inertial	

Attitude control capability, degrees	Prox Ops: 1 arcmin/axis 3 sigma	
Attitude Knowledge limit, degrees	Prox Ops: 30 arcsec/axis. 3 sigma	
Positional control/knowledge	Sampling: 0.5 m/0.1m 3 sigma	
Articulation/#–axes (solar arrays, antennas, gimbals, etc.)	Single-axis SAs ; Fixed HGA/MGA/LGA ;	
Sensor and actuator information (precision/errors, torque, momentum storage capabilities, etc.)	RWA(x4), Coarse Sun Sensor Assembly (x3 sets); Novatel IMU-LN200 (x2); DTU Micro Advanced Stellar Compass (uAS/C); Laser Rangefinder Alt; Optical/IR Camera; LIDAR	
Command & Data Handling		
Flight Element housekeeping data rate, kbps	44	
Data storage capacity, mbits	128,000	1,000
Maximum storage record rate, kbps	50,000	0.016
Maximum storage playback rate, kbps	50,000	
Power		
Type of array structure (rigid, flexible, body mounted, deployed, articulated)	ROSA; Elastic Memory Composites (EMCs); one-piece composite slit-tube booms	
Array size, meters x meters	4.5 X 21.5 collection area, ea. wing	
Solar cell type (Si, GaAs, Multi-junction GaAs, concentrators)	Triple- Junction GaAs	
Expected power generation at Beginning of Life (BOL) and End of Life (EOL), kW	BOL: 50kW EOL: 44.7kW	
On-orbit average power consumption, watts	1060 (MEV)(1)	
Battery type (NiCd, NiH, Li-ion)	Li-Ion (Li/$SOCl_2$)	Li/$SOCl_2$
Battery storage capacity, amp-hours	155 (Harpoon 0.75)	23

Notes:

1) On Orbit avg power does not include periodic Comm Tx mode and SEP thrusting during cruise phase.

2) Avg of 640 W (1 thruster lowest power setting) and a max of 14,720 W (2 thrusters at max power setting)

Cryogenic Sample Chain

A cryogenic sample chain and associated concept of operations was developed to ensure the sample does not experience temperatures above 120 °K from the sampling event to final recovery on Earth. The cryogenic system consists of the five subsystems shown in Figure 3-3. The Harpoon sample acquisition system, a Cryogenic Storage Refrigerator for storing the samples during return cruise, a sample handling arm that moves the sample as needed, Cryogenic Sample Dewars to passively cool the samples in the EEV, and the EEV itself. An initial mass trade on the Cryogenic Sample Dewar architecture in the EEV led to a passive, stored solid cryogen solution. This trade drove the requirements for the Cryogenic Storage Refrigerator and ground servicing.

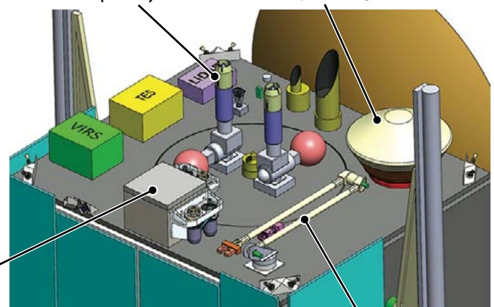

Figure 3-3: Cryogenic Sample Chain

Harpoon Sample Acquisition System

The sampling system (Figure 3-4) is a tethered projectile that penetrates the comet surface and returns a regolith sample to the spacecraft. It consists of:

1. A projectile capable of capturing a surface sample
2. A compressed gas expansion chamber that accelerates the projectile towards the comet.
3. A tether that keeps the projectile connected to the spacecraft
4. A tether management system

The projectile itself consists of an inner container, where the sample is captured, and an outer sheath which is added to help penetration. The inner container is a hollow shape with a mechanized lid to capture the sample. The momentum of the projectile forces the assembly into the regolith where the comet material is ingested. The lid closes once the projectile has come to a full stop, capturing the comet material in the process. Once the lid is closed, the outer sheath is decoupled and left at the comet. The sheath contains instrumentation for temperature and acceleration that is transmitted to the spacecraft.

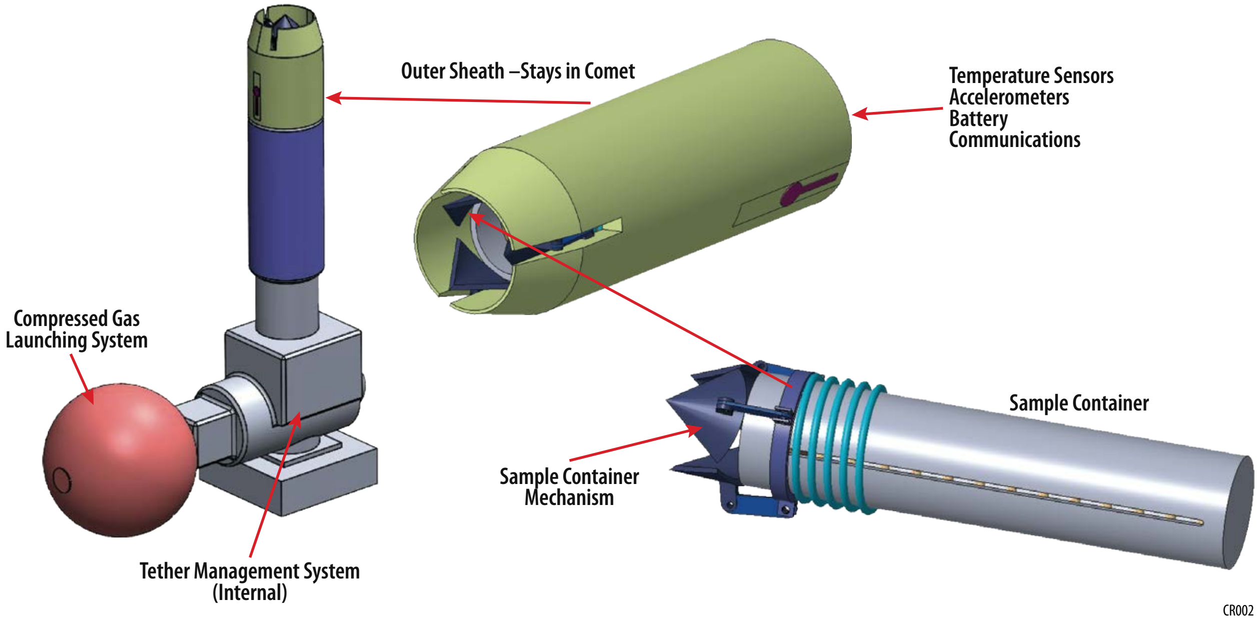

Figure 3-4: Harpoon Sample Acquisition System

The sample container is retracted by a tether that is controlled by the tether management system on the spacecraft. This retraction system is based on designs used on the ESA Rosetta mission. [1] The spacecraft will accelerate away from the comet during retraction to limit the risk of the projectile colliding with the spacecraft. Upon returning, the container is mechanically grounded to the spacecraft by pulling the tether tight and the tether is severed by a separate mechanism [2].

Harpoon Precooling

The Harpoon must be cooled to below the expected sample temperature at depth prior to the sampling event. Two stored cryogen tanks reside in the Cryogenic Storage Refrigerator and are initially filled with solid Argon. One stored cryogen tank will be used to cool each Harpoon. A heater will melt the Argon to pressurize one of the tanks. A valve will open releasing the liquid argon into a transfer line that leads to a heat exchanger on one of the Harpoons. Unused cryogen will be retained and slowly refrozen by the cryocooler to provide thermal ballast in the event of a cryocooler fault.

Robotic Arm

The Sample Handling Arm System is a 4 degree-of-freedom robotic arm with a gripper end effector. The arm takes the sample from the harpoon and places it in the cryogenic refrigerator. At return to earth, the arm moves the samples from the refrigerator to the EEV. The arm has an imager to confirm sample has been retrieved. The arm actuators draw heritage from the Mars 2020 Perseverance Rover and the robot flight software from OSAM-1 and Perseverance.

Cryogenic Sample Dewar

The Cryogenic Sample Dewars (Figure 3-5) contain several kilograms of solid argon to provide passive cooling during the entirety of Earth entry and recovery operations. The argon cryogen is charged into each of the Dewars prior to mission launch. The Cryogenic Sample Dewar can also provide some thermal capacitance in

[1] https://adsabs.harvard.edu/full/2003ESASP.524..239T

[2] https://www.ebad.com/tini-cable-cutter/

the event of a system fault such as temporary loss of cryocooler operation in the Cryogenic Storage Refrigerator. The solid cryogen in both Cryogenic Sample Dewars and secondary stored cryogen tanks provides an extended period of time to recover from such a fault without allowing the sample temperature to exceed 84 °K. The Cryogenic Sample Dewars will hermetically seal the comet samples to retain volatiles. Additionally, sensors will provide temperature and pressure history of the collected sample.

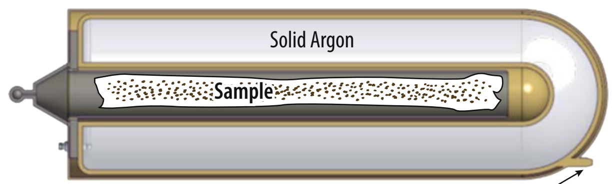

Figure 3-5: Cryogenic Sample Dewar

Cryogenic Storage Refrigerator

The Cryogenic Storage Refrigerator utilizes redundant cryocoolers to ensure that the stored cryogen and comet sample are maintained at temperature for the duration of the mission. The cryocooler is required to operate at a temperature below 84 °K, the triple-point of the stored argon cryogen, to keep the cryogen frozen solid. The Refrigerator contains two Cryogenic Sample Dewars and two stored cryogen tanks, all filled with solid argon. The stored cryogen tanks are used to precool the harpoons and robotic arm end effector prior to sample acquisition. Upon retrieval of the comet sample, the robotic arm will insert the Sample Container into one of the Cryogenic Sample Dewars stored in the Refrigerator. That action triggers a passive mechanism that creates a hermetic seal to contain the Comet Sample and its volatile species.

Earth Entry Vehicle

The EEV (Figure 3-6) design is based on heritage from the previously flown Stardust capsule and the design of the O-REx sample return capsule. The CROCODILE version is larger to include thermal insulation surrounding the sample storage dewars. The entry vehicle is a 60 degree sphere-cone, while the two stage parachutes consist of a 0.8 m diameter disk-gap-band drogue and a 7.3 m diameter triconical main parachute, similar to both Stardust and O-REx. The EEV has a larger diameter, it is heavier than Stardust and O-REx and has a maximum expected value ballistic coefficient of 126 Kg/m^2, which is double the ballistic coefficient of the other capsules. The higher ballistic coefficient and the entry conditions described below results in heating conditions that will necessitate a different heat shield (HS) thermal protection system (TPS) choice compared to Stardust and O-REx. The forebody HS TPS material will be 3D Mid-Density Carbon Phenolic (3MDCP), which is derived from the Heatshield for Extreme Entry Environments (HEEET) Dual Layer TPS, while the Backshell (BS) TPS will be Phenolic Impregnated Carbon (PICA). During the development of the HEEET TPS, the insulating layer was tested independently and shown to have excellent performance capability at ~2000 W/cm^2 and 2-3 atm of stagnation pressure which is much more capable than the expected heating conditions for the EEV and can be manufactured to the sizes for the EEV design. PICA has been used on Stardust's forebody, two previous Mars missions, Mars Science Laboratory (MSL) and Mars 2020, SpaceX Dragon capsules, and is the primary TPS for the Dragonfly mission to Titan.

An initial assessment of the heat up (post landing) at the Utah Test and Training Range (UTTR) was done, assuming the EEV was laying in full sun surrounded by 120 °F air and desert. The sample dewar was presumed to be surrounded with 3" of titanium foam Insulation, with a thermal conductance of 0.07 W/m-K. Without a stored cryogen, the sample temperature was estimated to exceed 120 °K in approximately 3 hours. The 4.5 kg of solid Argon in each sample dewar will absorb enough heating to maintain the sample temperature requirements for at least 6 hours on the ground.

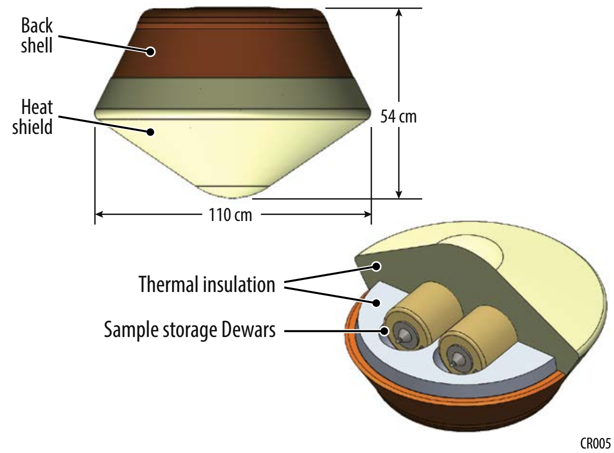

Figure 3-6: Earth Entry Vehicle with Sample Storage Dewars

Mission Design

Baseline Trajectory Design

Launch opportunities for the baseline CROCODILE trajectory that maintain a dry mass margin of at least 30% span May 6 – July 30 of 2036. The first 21-day span is considered the primary launch period, with a backup 21-day launch period opening as late as July 10, 2036. Trajectories across the entire three-month span are nearly identical, with Earth entry interface of the EEV occurring on the same day and time throughout.

Figure 3-7 shows the baseline trajectory (on the left) as it appears with launch on May 6, 2036 along with the final approach for EEV entry (on the right) with landing site at the Utah Test and Training Range (UTTR). The trajectory model of the EEV from entry interface to the UTTR landing site is discussed in ConOps section. The spacecraft launches from Kennedy Space Center (KSC) on a Falcon Heavy Recovery. The trajectory uses and Earth gravity assist (EGA) one year after launch to increase the orbital energy to eventually rendezvous with the comet in January 2042. Long Solar Electric Propulsion thrust arcs are used for most of the outbound trajectory—primarily to increase the orbital energy and ultimately velocity-match the comet for rendezvous.

Operations at the comet occur while the comet passes through its aphelion. During this time, the spacecraft does not have sufficient power to operate the SEP system, and will rely on chemical propulsion during proximity operations. Operations at the comet will continue for 4 years until departure in February 2046, when the comet reaches a solar distance that provides sufficient power to operate the SEP system. Similar to the outbound leg, the inbound cruise trajectory from 67P back to Earth (for EEV entry) uses an EGA and long thrust arcs to reduce the orbital energy for EEV entry on November 15, 2051 at about 9 AM MST.

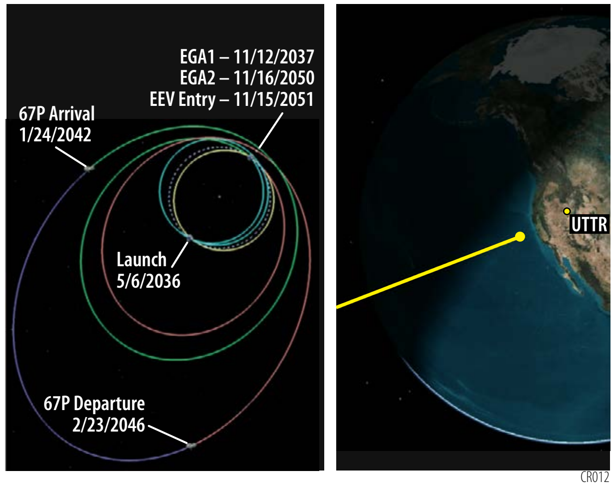

Figure 3-7: Left: Top-down view (from ecliptic north) of CROCODILE's SEP trajectory. The different legs of the trajectory are shown in various colors to distinguish them on the plot. Earth's orbit is shown for reference as a dashed line. Right: EEV approach up to atmospheric entry interface at a radius of approximately 6532 km. Target landing site at UTTR

Concept of Operations

The CROCODILE Concept of Operations (ConOps) is portioned into nine (9) mission phases with five (5) subphases for the Proximity Operations Phase as outlined in Table 3-5. The overall mission duration time of flight of ~15.5 years includes ~4.1 years at Comet 67P. Each phase/subphase will have further detailed operational activities, entry/exit criteria, contingencies and known conditions and constraints. The proximity operations enables global mapping of 67P as well as high resolution mapping of potential sample sites over a two-year period during the four-year stay at the comet. The Shoot and Go (SAG) sample acquisition approach uses a tethered harpoon allowing for no direct spacecraft contact with the comet's surface. The Deep Space Network (DSN) will be utilized during all mission phases for uplink and downlink communications and radiometric tracking.

Table 3-5: Mission Phases

Phase	Activity Description	Event Date	Duration
1.0 Launch	Launch from CCAFS, FL., on an Earth-escape trajectory	5/6/2036	85-day launch window
2.0 Commissioning	Spacecraft (S/C) checkout; Solar Electric Propulsion (SEP) calibrations. Payload checkout and instrument calibrations		1 mo
3.0 Cruise to Comet	Outbound cruise to comet 67P with Earth flyby gravity assist; Instrument survey rehearsal, Quiescent Ops during rest of the cruise		68.6 mo
4.0 Proximity Operations Phase (Sub-phases):			
4.1 Rendezvous	Initial detection ~1M km, initial comet rotation characterization	1/24/2042	1.4 mo
4.2 Approach & Preliminary Survey	Preliminary surveys to identify comet activity, rotation characterization, low-res topography map for landmark navigation		4.0 mo (1 mo gnd processing)
4.3 Detailed Survey (Nav/Science)	Survey ~9-12 candidate sample sites; down select to three sites. Mapping flybys for initial shape model; Three, ~2 km close passes/inspection of selected target sample sites		12 mo (2 mo gnd processing)
4.4 SAG Rehearsals	Shoot and Go (SAG) ops rehearsals to test navigation & control		3.0 mo
4.5 Sample Collection	SAG sample collections/storage at two (2) sites		6 mo

5.0 Quiescent Operations	Remain at 67P orbit; monitor samples and S/C health		23 mo
6.0 Departure/Cruise to Earth	Inbound cruise to Earth w/Earth flyby. Monitor samples and S/C health	2/23/2046	68.6 mo
7.0 Earth Return & Sample Transfer	Earth intercept. Sample transfer to EEV	11/12/2051	24-72 hr
8.0 EEV Release & Sample Recovery	EEV release and landing. Sample recovery and curation	11/15/2051	10.25 hr
9.0 S/C Divert & EOM	S/C divert maneuver; End of Mission (EOM) /disposal		~1 mo

Launch through SAG Rehearsals Phase:

Following launch, the SEP system will begin thrusting for the ~5.7 year cruise to 67P, with an Earth flyby and gravity assist occurring a year after launch. During the Earth flyby, a set of Earth observations will be performed for further instrument calibrations and to serve as a rehearsal for comet surveys.

Proximity Operations commences at the end of interplanetary cruise and comet Rendezvous at a range of ~1M km. The proximity operations concept follows recent small body rendezvous missions such as OSIRIS-REx and builds upon lessons learned from Rosetta's rendezvous with 67P. Moreover, data collected during Rosetta is utilized where possible to reduce the time needed for characterizing the comet and the overall proximity operations duration.

During rendezvous, the comet rotation will be characterized and stellar optical navigation used to update the comet's orbit solution and subsequent trajectory plans. Orbit Determination (OD) updates are performed by Delta-Differential One-way Ranging (DDOR) via Deep Space Network (DSN) contacts to prepare for close approach. At approximately 100,000 km, SEP thrusting ends and the spacecraft transitions to the approach subphases. The Approach and Preliminary Survey subphase will begin with far-field mapping at a range of 100,000 km through near-field mapping down to 200 km utilizing the science camera suite. The mapping will support landmark navigation for close proximity operations, sample site selection and updates to proximity ops based on observed comet activity and changes to the surface since it was last observed by Rosetta given that multiple perihelion passes of the comet will have occurred by the time of CROCODILE's arrival.

After ground processing of the preliminary survey data, a mapping campaign at 50, 25 and 10 km will generate gravity and science value maps, identify any hazards to flight, and generate a 20 cm topography map for sample site navigation. A series of hyperbolic flybys with close passes at 2 km from the comet surface are executed to capture high-resolution images of the candidate sample sites for final determination of the two sample sites.

Once the two primary sample sites are determined, the S/C mission operations and flight dynamics team will conduct a series of SAG sample collection maneuver rehearsals to verify S/C navigation and control, descent to comet surface and the back-away maneuver in preparation for the actual sample collections.

Sample Collection Phase:

The baseline objective is to collect two comet ice core samples over a six month period. Two primary sample sites (and at least one backup) will have been selected from the detailed global surveys and close up inspection flybys. Figure 3-8 shows the sampling site flight path from the 10 km home terminator orbit, where the spacecraft proceeds through the checkpoint, executes a maneuver to match the rotation rate of the comet at the match-point, and then, using the Lidar and Laser altimeter, begins a slow descent down to the ~10 m hover point for sample collection. The SAG collection approach avoids having the S/C contact or land on the comet surface.

Pre-sampling preps will have been accomplished such as handling arm checkout and prepositioning, and cryo refrigerator door opening. The harpoon system holder that contains the harpoon's inner/outer sample sheath will be pre-cooled along with the arm's end effector to minimize thermal exchange to the sample, both before and after sample collection. Once the S/C reaches the sampling altitude, it will hover a few seconds to stabilize.

The harpoon will be commanded to fire, and the S/C will immediately execute a back-away maneuver to assist dynamic stabilization of the tethered harpoon and extraction of the inner sheath containing the core sample from the outer sheath. The outer sheath contains pressure/accelerometer sensors that will begin transmitting data to the S/C upon firing to ascertain information of the *in situ* sample. The harpoon tether retrieval mechanism will then reel the sample back on board the S/C. A camera will record the sample site collection activity. Upon retrieval of the sample back into the harpoon holder, the handling arm will grip the sample, maintaining positive control, while the tether is commanded cut. It will then transfer the sample into the sample dewar and place it into the Cryogenic Storage Refrigerator. During the transfer, images will be taken of the inner sheath's sample core depth gauge for post-ground data assessment. The sample dewar will then be hermetically sealed and Cryogenic Storage Refrigerator door closed for long term storage. The sample dewar's pressure/temperature monitoring sensors will be downlinked along with cryo cooler telemetry throughout the mission. This process is repeated for the second sample site collection.

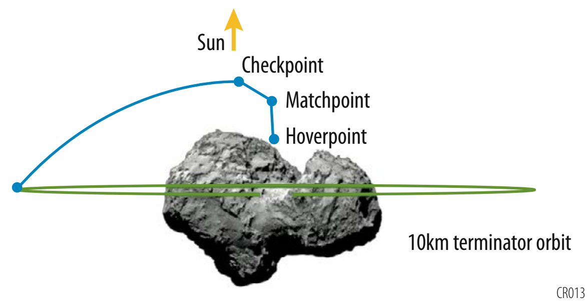

Figure 3-8: Flight Path for Sample Site Collection

After sample collection, the S/C will remain in heliocentric orbit alongside 67P for ~2 yrs in a Quiescent Operations phase, monitoring the comet samples and spacecraft health, while awaiting the Departure phase. The S/C then departs the comet for its ~ 5.7 year journey back to Earth. Earth intercept occurs one year after another Earth flyby, marking the end of the Earth Return Phase.

EEV Sample Transfer and Recovery Phase:

At approximately 100 hours prior to EEV release, the sample dewars will be robotically transferred from the Cryogenic Storage Refrigerator to the EEV. The handling arm gripper will be precooled and the EEV and Storage Refrigerator door opened. The arm will transfer each sample Dewar into the EEV's sample housing cradle. The EEV system will mechanically latch down the sample dewars for re-entry loading and secure them in the cradle in a clocked position to engage their pressure/temperature sensor connectors for monitoring and to properly position their pressure vents to allow solid argon to freely vent as they warm. The vented cold argon gas will assist in maintaining a cold thermal environment inside the EEV during re-entry and ground recovery. Once properly latched and connected into the EEV, the clamshell door will be closed and readied for release. This activity is expected to take approximately one hour. Within 2-3 days after sample transfer, the S/C will perform final attitude and trajectory adjustments for EEV release.

Entry Descent and Landing (EDL)

Figure 3-9 shows EEV entry, descent and landing. The EEV is released from the S/C 4 hours before atmospheric interface, which is defined as 125 km altitude. The EEV has a similar entry speed and velocity as O-REx and reaches a peak acceleration of 28.7 Earth g's. The EEV will be tracked both by radar and optical ground tracking stations. After the drogue parachute is deployed at Mach 1.35, which is similar to conditions experienced by Stardust, a larger main parachute is deployed at 3 km altitude to complete the descent sequence. Upon chute deployment, the EEV's directional finding locator beacon will be activated to assist ground team recovery. Approximately 11 mins after reaching atmospheric interface, the capsule touches down at UTTR at 7 m/s and the parachute is cut to reduce risk of dragging the sample on the ground. The landing ellipse footprint is expected to be similar to O-REx with major and minor axis of 65 x 15 km. The EEV is expected be recovered (Figure 3-10) well within 6 hours, maintaining sample cryo temperatures after landing and transfer to a field lab where the sample dewars will be placed in a portable refrigerator for transport to the curation facility. The S/C will execute its Earth divert maneuver ~20 minutes after release for its eventual end of mission disposal.

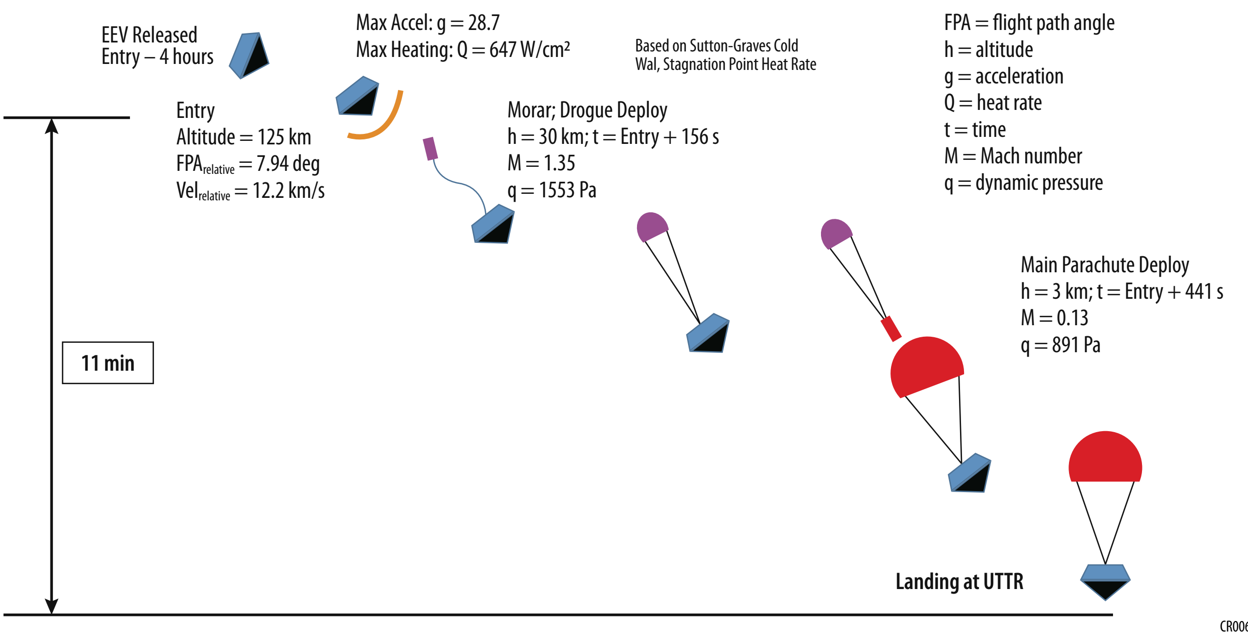

Figure 3-9: The concept of operations for the Earth Entry Vehicle.

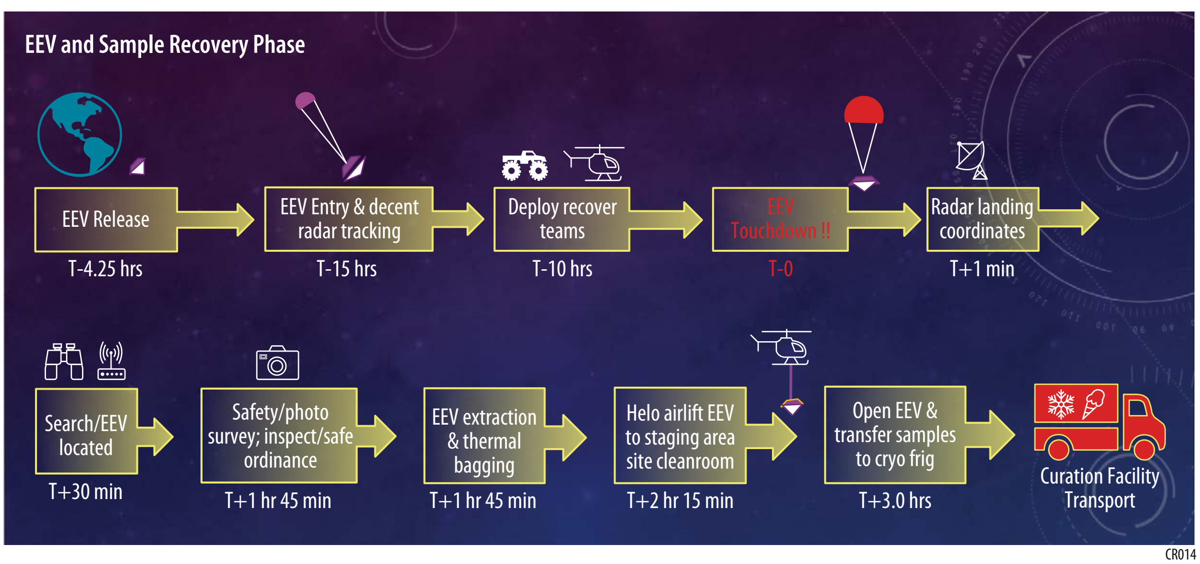

Figure 3-10: EEV/Sample Recovery Operations

Risk List

Maintaining sample temperature requirements for the duration of the mission is the top mission risk (Figure 3-11, Table 3-6). This encompasses everything from initial sample acquisition to ground recovery and poses multiple challenges. Pre-Phase A TRL development, extensive thermal modeling and analyses, and prototype building in Phase A and B are necessary to ensure sample thermal requirements are met. Other risks derive from uncertainty in 67P mechanical and atmospheric properties, complications associated with the large spacecraft electrical power system, and the critical functions associated with preparing the sample for release and return to Earth.

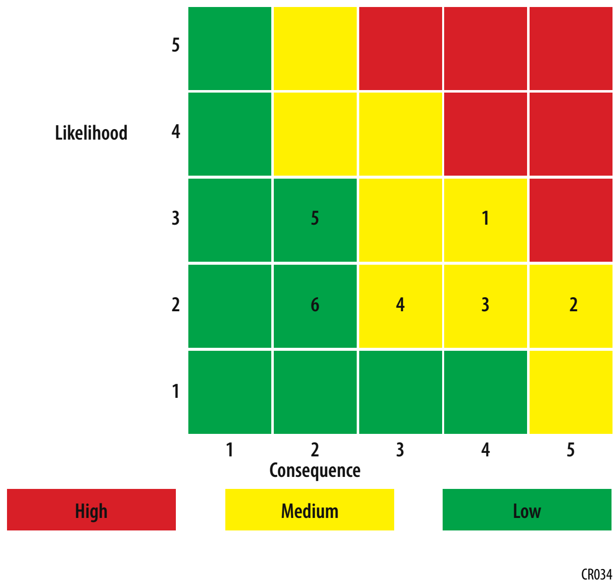

Figure 3-11: CROCODILE Risk Matrix

Table 3-6: CROCODILE Risks and Mitigation Strategies

Rank	Risk Title	Risk Description	Mitigation Strategy
1 (L3, C4)	Sample Temperature Control and Storage	**Given that** the Sample acquisition, storage, and return is a complex and multi year process **There is a possibility** sample temperature requirements may be violated **Resulting in** degradation of sample integrity and loss of science return	• Plan for and budget for extensive Phase A and Phase B thermal modeling • Allocate funding in Phase A and B to develop and test sample chain prototypes to characterize thermal performance • Invest in technology development to carefully characterize harpoon/surface thermal interaction
2 (L2, C5)	Sample Return Operations	**Given that** the trajectory allows one opportunity to release the EEV **There is a possibility** issues may compromise the approach and release process **Resulting in** loss of sample	• Implement redundancy on mechanisms associated with sample manipulation and EEV release. • Implement high heritage hardware where possible in sample handling chain • Include time and technical margin on processes associated with sample return to earth
3 (L2, C4)	Incomplete Cometary Surface Properties Model	**Given that** existing models may not accurately reflect comet mechanical properties **There is a possibility** sample acquisition depth requirements may not be met **Resulting in** diminished science return	• Conduct penetration testing in Phase A that will span the range of expected comet mechanical properties • Design sample retrieval system for expected worst case surface mechanism properties with margin.

4 (L2, C3)	TRL development	**Given that** multiple aspects of the sample acquisition , manipulation and storage system require TRL development **There is a possibility** complications may delay TRL advancement **Resulting in** cost increase and schedule delays	• Invest in technology development prior to Phase A • Implement robust TRL development plan • Include appropriate schedule and budget reserves on hardware requiring technology development
5 (L3, C2)	Comet Proximity Operations	**Given that** existing models may not accurately reflect comet activity **There is a possibility** comet activity could be higher and more disruptive than expected **Resulting in** impact to proximity operations	• Design trajectory for sample acquisition post perihelion • Implement Extensive mapping and observation phase to characterize comet surface and activity • Implement ability to position solar panels to minimize perturbance forces from the comet ejecta during sample acquisition process
6 (L2,C2)	Electrical Power System	**Given that** the mission power requirements require a larger solar array than typical for planetary missions **There is a possibility** unexpected complications will arise in solar array implementation **Resulting in** cost and schedule delays	• Invest in technology development to increase cryocooler, solar array efficiency • Assess in Phase A and B the systemic impacts of the large solar arrays on the spacecraft design

4.0 DEVELOPMENT SCHEDULE AND SCHEDULE CONSTRAINTS

High-Level Mission Schedule

CROCODILE is assumed to be implemented consistent with a 2 step New Frontiers Announcement of Opportunity starting with a 9 month Phase A followed by a 6 month bridge phase leading to Phase B. The schedule is driven by the launch date and commencement between 2023 and 2032 as per the study guidelines. The schedule is generally consistent with previous New Frontiers sample return missions with additional time allotted for the substantial prototype development and testing necessary to reach TRL 6. As per the technology development section, all hardware is at TRL 6 prior to Mission PDR in March 2032.

The spacecraft bus is on the critical path, allowing time to procure the large solar array and implement them into the design concept. The secondary critical path is through the harpoon system due to the time needed to build and test prototypes. Figure 4-1 shows the CROCODILE Phases A-E schedule and includes a summary of schedule milestones and Table 4-1 lists expected key phase durations. The launch window opening in May 2036 allows for secondary opportunities through July 2036 while still maintaining the required mass margin. The implementation assumes the harpoon system and cryogenic hardware has been matured to TRL 5 prior to mission implementation with maturation to 6 and system level testing completed during mission Phases A and B.

Launch readiness date is May 6, 2036. Launch is followed by a 6 year cruise, 4 years of Comet Proximity Operations and sample return to Earth in November 2051. This concept does not facilitate an extended mission.

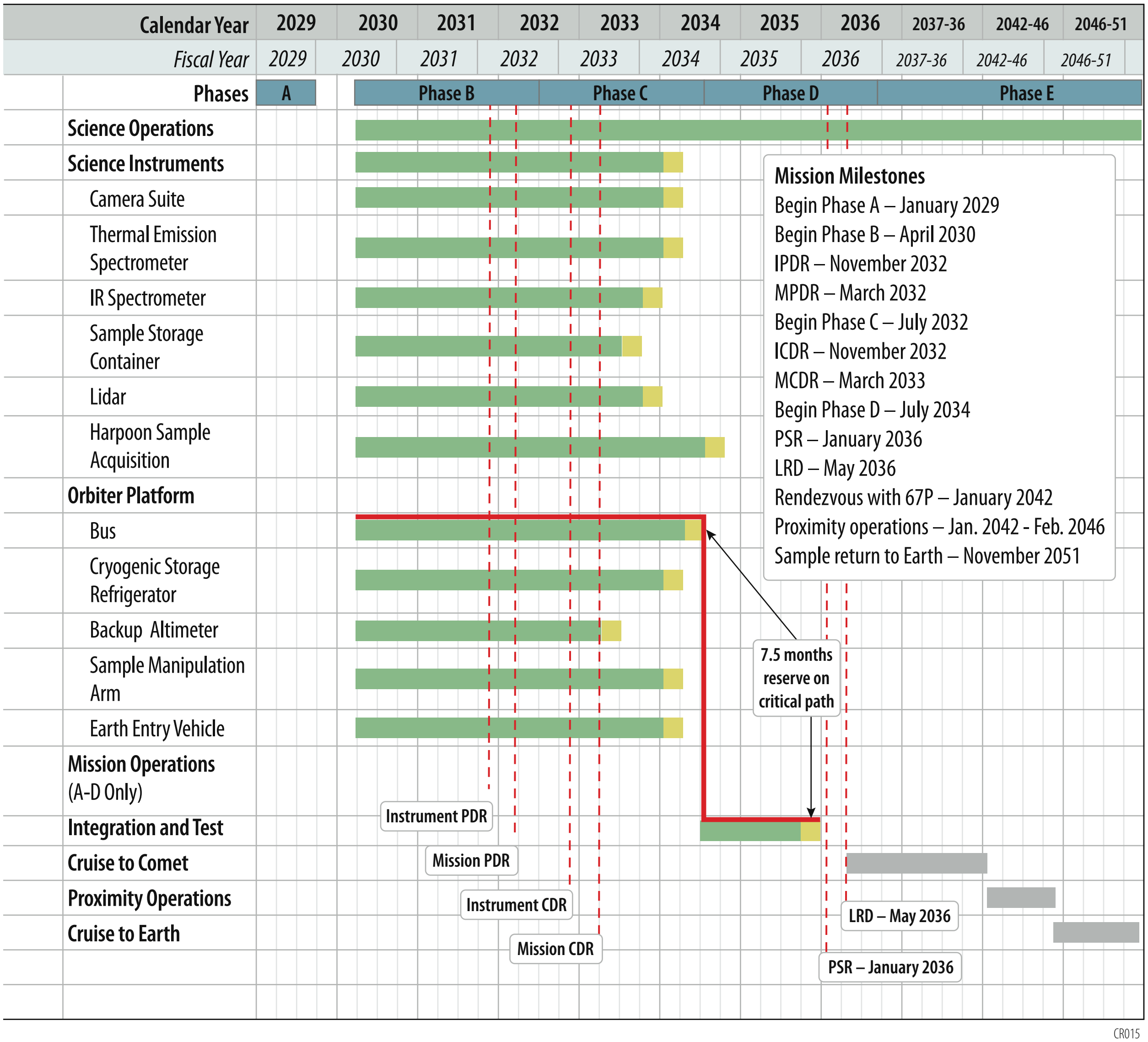

Figure 4-1: CROCODILE Implementation Schedule

Table 4-1: CROCODILE key Phase Duration

Project Phase	Duration (Months)
Phase A – Conceptual Design	9
Phase B – Preliminary Design	27
Phase C – Detailed Design	24
Phase D – Integration & Test	23
Phase E – Primary Mission Operations	186
Phase F – Extended Mission Operations	N/A
Start of Phase B to PDR	23
Start of Phase B to CDR	35
Start of Phase B to Delivery of Camera Suite	42
Start of Phase B to Delivery of Thermal Emission Spectrometer	42
Start of Phase B to Delivery of IR Spectrometer	40

Start of Phase B to Delivery of Sample Storage Dewars	37
Start of Phase B to Delivery of Lidar	42
Start of Phase B to Delivery of Harpoon Sample Acquisition System	48
Start of Phase B to Delivery of Bus	45
Start of Phase B to Delivery of Cryogenic Storage Refrigerator	42
Start of Phase B to Delivery of Backup Altimeter	33
Start of Phase B to Delivery of Sample Manipulation Arm	42
Start of Phase B to Delivery of Earth Entry Vehicle	42
System Level Integration & Test	17
Project Total Funded Schedule Reserve	7
Total Development Time Phase B - D	74

Technology Development Plan

Reducing mission power needs, increasing cryocooler efficiency, and maturing cryogenic sample acquisition and storage technology will enable future lower risk and lower cost implementations of a comet cryogenic sample return mission. The specific TRL development needed to implement the CROCODILE concept is listed in Table 4-2. Maturation to TRL 5 is assumed in this study prior to Phase A and maturation to 6 at the component and systems level is assumed via prototype development and Testing in Phases A and B. The time required for maturation to TRL 6 allows for completion prior to mission PDR in March 2032. Total cost for technology development is $28M, FY25 before reserves.

Table 4-2: CROCODILE Technology Development

Item	Lowest TRL	TRL Justification	Enabling Technology	Pre -Phase A	Mission Implementation	Duration	Est. Cost (M$FY25)
Harpoon Sample Acquisition System	3	• Concepts flown on Rosetta • General establishment of harpoon penetration via testing GSFC drop facility (Reference) • Sample Capture Mechanism not demonstrated at expected conditions	• Characterizing harpoon/ Comet thermal interaction • Establish performance of sample sheath acquisition mechanism • Consistently achieve depth requirement given expected range of target mechanical properties	• Demonstrate performance of sheath sample capture mechanism concept. • Characterize harpoon - sample thermal behavior Demonstrate controlled sample depth	High fidelity prototype to confirm performance in expected operating temperature and range of target specific mechanical properties	3 yrs	15
Sample Manipulation Arm	5	Heritage from multiple missions including Mars 2020, OSAM -1 and Perseverance	Characterize Operations under cryogenic conditions		Systems level testing at operating conditions	14 mths	3
Cryogenic Storage Refrigerator	3	Cooling concept demonstrated in a Lab Environment Long term hermetic cryogenic seal not demonstrated	• Predictable and controlled coolant flow • Establish Ar cooling efficiency in a vacuum • Long term cryogenic hermetic sealing	• Demonstrate controlled coolant flow • Develop and test prototype seal concepts	Systems level testing at operating conditions	24 mths	5

Cryogenic Sample Dewar	5	Prototype not tested in target specific operating conditions	Performance at expected cryogenic conditions		Build and test prototype in Mission Phase B	12 mths	5

5.0 MISSION LIFE-CYCLE COST

At ~1.9 B (FY25 with reserves) for Phase A-D, including all technology development the CROCODILE concept falls outside of the range of a typical New Frontiers Announcement of Opportunity cost cap. The total is reflective of a capable instrument suite required for site selection, a Lidar and backup altimeter for close approach safety, a technically challenging cryogenic sample acquisition/handling system, and all associated technology development. The large solar arrays are significant spacecraft bus cost drivers.

At ~$300M (FY25,including reserves) Phase E costs are significantly higher than typical planetary mission due to the extended operations at the comet, the long mission duration, and the cryogenic sample ground handling.

The total mission cost, including Phase E and launch vehicle services is ~ $2.54B. As per study guidelines 50% reserve was added to Phase A-D costs and 25% to Phase E costs. 50% was also added to TRL development cost estimates. For simplicity, all TRL development costs are separated from A-D costs. Mission lifecycle costs in Real Year and FY25 are shown in Table 5-1.

Costing Methodology and Basis of Estimate

For much of the payload, analogous instrument and hardware costs was used to generate the costs, adjusted for fiscal year. The spacecraft bus was costed by the GSFC Cost Estimating, Modeling, & Analysis (CEMA) office based on a detailed MEL generated by the engineering team. CEMA output is provided in the cost basis appendix. The Harpoon system cost was estimated by the GSFC Requirements Assessment Office (RAO) office from a database of analogous hardware. Percentage wraps consistent with sample return missions were applied to generate project management, systems engineering, S&MA and Science estimates. Grass roots TRL development costs were provided by subject matter experts. Phase E costs were generated from past sample return missions adjusted for the extended CROCODILE mission duration and increased complexity associated with handling a cryogenic sample.

Cost Estimate(s)

Table 5-1: Mission lifecycle cost estimate

Item	Technology Development (M$FY25)	FY29	FY30	FY31	FY32	FY33	FY34	FY35	FY36	Total (RY)	Total (FY25)
Phase A Concept Study		$9.7								**$9.7**	$8.7
		Phase A-D									
Project Management		$0.7	$6.1	$12.6	$13.9	$17.0	$18.2	$21.0	$14.4	**$103.9**	$83
Systems Engineering		$0.5	$4.3	$8.8	$9.7	$11.9	$12.8	$14.7	$10.0	**$72.8**	$58
Safety and Mission Assurance		$0.2	$1.8	$3.8	$4.2	$5.1	$5.5	$6.3	$4.3	**$31.2**	$25
Pre Launch Science		$0.7	$6.1	$12.6	$13.9	$17.0	$18.2	$21.0	$14.4	**$103.9**	$83
PolyCam		$0.2	$2.7	$5.5	$5.4	$4.8	$3.8	$0.4	$0.0	**$22.7**	$19
MapCam		$0.1	$2.2	$4.6	$4.5	$4.1	$3.2	$0.3	$0.3	**$19.4**	$16
Sample Camera (Close Range)		$0.2	$2.6	$5.4	$5.3	$4.8	$3.8	$0.4	$0.2	**$22.7**	$19
Thermal Emission Spectrometer		$0.2	$3.1	$6.3	$6.2	$5.6	$4.4	$0.4	$0.3	**$26.6**	$22

IR Spectrometer		$0.4	$5.6	$11.5	$11.3	$10.1	$8.0	$0.8	$0.3	**$48.1**	$40
Sample Storage Dewar	$5.0	$0.0	$0.2	$0.3	$0.3	$0.3	$0.2	$0.0	$0.6	**$2.0**	$1
Lidar		$0.4	$5.9	$12.0	$11.8	$10.6	$8.4	$0.9	$0.0	**$49.9**	$42
Harpoon Sample Acquisition	$15.0	$0.4	$6.7	$13.8	$13.6	$12.2	$9.6	$1.0	$0.0	**$57.4**	$48
Spacecraft Bus, Including Integration		$4.5	$70.9	$145.6	$143.4	$128.3	$101.4	$10.3	$0.0	**$604.4**	$506
Cryogenic Storage Refrigerator	$5.0	$0.4	$6.6	$13.6	$13.4	$12.0	$9.4	$1.0	$0.7	**$57.0**	$47
Backup Altimeter		$0.1	$1.1	$2.3	$2.3	$2.0	$1.6	$0.2	$0.0	**$9.6**	$8
Sample Manipulation Arm	$3.0	$0.3	$5.3	$10.9	$10.7	$9.6	$7.6	$0.8	$7.1	**$52.3**	$38
Earth Entry Vehicle		$0.2	$1.7	$6.8	$6.8	$6.5	$5.7	$2.7	$0.0	**$30.4**	$26
Mission Operations (A-D Only)		$0.1	$2.1	$4.3	$4.3	$3.8	$3.0	$0.3	$0.7	**$18.6**	$15
Ground systems		$0.0	$7.6	$7.8	$16.0	$24.7	$42.3	$52.1	$0.0	**$150.6**	$133
Payload Integration and Test		$0.0	$3.3	$3.4	$7.0	$10.8	$18.5	$22.8	$0.6	**$66.5**	$58
Total Development w/out Reserves	**$28**	**$10**	**$146**	**$292**	**$304**	**$301**	**$286**	**$157**	**$54**	**$1,550**	**$1,288**
Reserves	$14	$5	$73	$146	$152	$151	$143	$79	$27	$775	$644
Total A-D Development Cost		**$15**	**$219**	**$438**	**$456**	**$452**	**$428**	**$236**	**$81**	**$2,325**	**$1,932**
Technology Development (Including Reserves)											**$42**
Total A-D Including Technology Development (w/Reserves)											**$1,974**
Phase E Costs - Mission Operations											**$241**
Phase E Costs - Sample Curating											**$25**
Phase E Reserves											**$67**
Launch Services		$0.0	$0.0	$0.0	$0.0	$48.0	$96.0	$96.0	$0.0	$240.0	**$240**
Total Including Phase E and Launch Services											**$2,546**

APPENDICES

1.0 DESIGN STUDY REPORT

Mission Design and Concept of Operations

Baseline Trajectory Design

A summary of the baseline trajectory parameters are provided in Table 1-1 and a plot of the distances from the spacecraft to the Sun, Earth, and comet is provided in Figure 1-1. The assumptions used in the trajectory design are provided in Table 1-2 where the margins used for EP power, Xenon (Xe) mass, and EP thruster duty cycle, are selected based on the findings of Oh et al. The trajectory was designed using Goddard Space Flight Center's (GSFC's) Evolutionary Mission Trajectory Generator (EMTG) trajectory-design tool. For this study, all trajectories were modeled using a Keplerian force model with patched-conic EGAs. For the final Earth approach prior to entry interface, the Sun-centered Keplerian force model switches to an Earth-centered Keplerian force model when the spacecraft is within Earth's sphere of influence (SOI) at a distance of 1.47e6 km from Earth's center. Plots of the trajectory were generated by rendering binary SPK files in AGI's Systems Tool Kit (STK).

Table 1-1: Baseline trajectory characteristics (correspond to launch-period open)

Parameter	Value
Launch Date	May 6, 2036
Launch C3	7.8 km^2/s^2
Right Ascension and Declination of Launch Asymptote	-315.4 deg and 5.6 deg
EGA1	Nov 12, 2037, 300-km alt
67P Arrival	Jan 24, 2042
67P Departure	Feb 23, 2046
EGA2	Nov 16, 2050, 300-km alt
EEV Entry Interface	Nov 15, 2051
Mission Lifetime	15.5 years
Minimum Solar Distance	0.84 AU
Maximum Solar Distance	5.68 AU
EP Thruster	NEXT-C
Number of Thrusters	2 primary + 1 backup
Total Chemical Propellant with 10% Contingency	280 kg
Total Xe Mass with 10% Margin	1500 kg
Total Wet Mass	5441 kg
Launch Vehicle	Falcon Heavy Recovery
Launch Site	KSC
Launch Vehicle Lift Capability	5450 kg
Max Dry Mass including EEV (trajectory limited)	3660 kg
Observatory Dry Mass MEV (with 30% Contingency)	2541kg
Dry Mass Margin (%)	30%

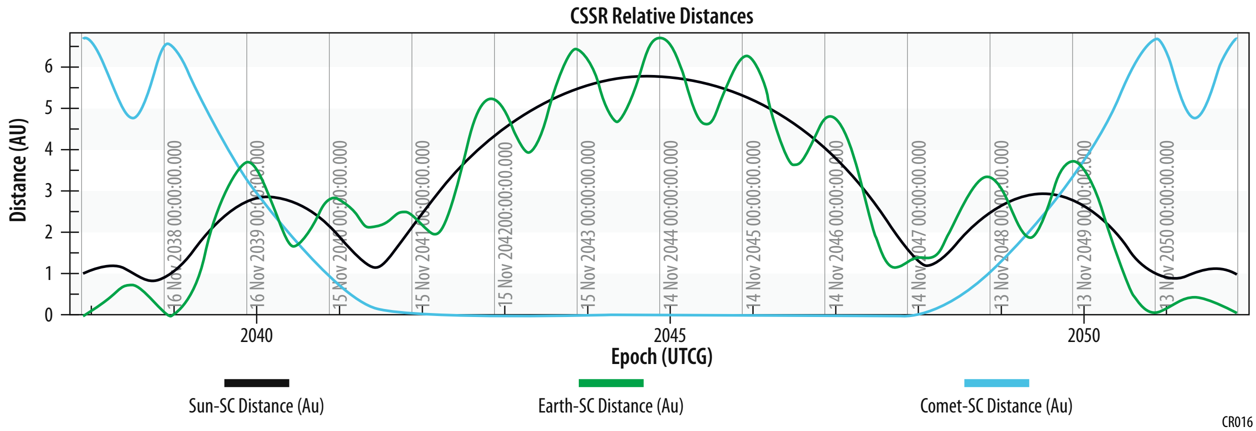

Figure 1-1: Distances from the spacecraft to the Sun, Earth, and comet throughout the baseline trajectory for the launch-period open.

Table 1-2: Trajectory design assumptions and constraints

Parameter	Value
Launch date range	Jan 1, 2029 – Dec 31, 2038
Declination of launch asymptote (DLA)	+/-28.5 deg
Solar array power at 1 AU BOL (P0)	50 kW
Solar array constant degradation	0.7%/year
Power reserved for observatory during EP thrusting	1.036 kW
Power margin for EP (solar array and observatory reserve)	10%
Power model with solar distance	$P0/r^2$, kw/AU^2
Xe mass margin	10%
Xe tank capacity	1500 kg
Mass reserved for chemical prop at comet with contingency	250 kg
Mass reserved for chemical prop for Earth divert maneuver with contingency	30 kg
Thruster duty cycle	90%
Maximum flight time	16 years
Minimum stay-time at comet	2 years
Forced coast duration after launch and prior to Earth encounters	45 days
Forced coast duration after EGAs	2 days
Entry interface inertial radius	6532.459 km
Entry interface inertial flight path angle	-9.137779 deg
Entry interface inertial azimuth (clockwise from true north)	60.59292
Target landing site	UTTR

Early Design Considerations and Trades

Several design trades were conducted in identifying the baseline trajectory design. The key design trades are listed in Table 1-3.

Table 1-3: Trajectory Design Trades

Parameter	Value
Launch date range	Jan 1, 2029 – Dec 31, 2042
Launch vehicle	Falcon Heavy Recovery and Expendable
Thruster	NEXT-C, XIPS-25
Array size at 1 AU	30, 45, 50, 55 kW
Max flight time	14, 15, 16 years
Xe tank size	1500 kg, unlimited
Flyby sequence (E = Earth, V = Venus, M = Mars, and c=comet encounter)	EEcEE, EcEE, EEcEEE, EEEcEE, EVEcEE, EEcMEE, EMEcEE, EEcEME, EEcMEE
Target comet	79P, 67P, 222P, 1884 O1, 1978 R1, 306P

The target comets listed in Table 1-3 were those that performed the best of the targets investigated, in terms of maximum dry mass achieved from a trajectory design. The full list of targets investigated is provided in Table 1-4 in order to best to worst performing. Note that once 67P was identified as one of the top performers, it was adopted as the baseline. Thus, a thorough trajectory search to the other targets was not conducted, and may yield similar or better dry-mass performance than 67P if investigated further. The targets listed in Table 1-4 were selected based on their orbital characteristics, to reduce the search to comets that were most likely to allow for a feasible mission design. Specifically, comets were selected with the following characteristics:

- Ecliptic inclination ≤ 10°
- 0.65 AU ≤ perihelion ≤ 1.3 AU
- Aphelion ≤ 6 AU

Table 1-4: Target Comets Investigated

Comet	Perihelion, AU	Inclination, deg	Aphelion, AU	Period (yrs)
79P	1.12	3.15	4.77	5.06
67P	1.24	7.04	5.68	6.45
222P	0.78	5.15	4.93	4.83
D/1884 O1	1.28	5.47	4.87	5.39
D/1978 R1	1.10	5.95	5.48	5.97
306P	1.25	8.36	4.96	5.47
384P	1.12	7.30	4.70	4.96
P/2007 T2	0.70	9.90	5.49	5.44
41P	1.05	9.23	5.12	5.42
15P	0.97	6.80	6.00	6.51
300P	0.82	5.70	4.56	4.42
P/1999 RO28	1.23	8.19	5.82	6.62
P/2009 WX51	0.80	9.59	5.36	5.41
289P	0.96	5.90	5.13	5.31
D/1770 L1	0.67	1.55	5.63	5.60
320P	0.99	4.89	5.22	5.47

Electric propulsion was selected early on for the baseline design as the most likely to provide a feasible solution. A limited search of chemical-propulsion solutions was conducted but was by no means thorough. Depending on the available launch vehicles, a heavy-lifting launch vehicle may be capable of enabling a chemical propulsion design, though no such design was identified in this study.

Trade-Study Results and Alternate Launch Opportunities

The results from one of the many trade studies conducted are presented in Figure 1-2. The results indicate that launch opportunities with over 20% dry-mass margin exist for any launch year from 2034 through 2038 depending on the selected spacecraft configuration and maximum flight time.

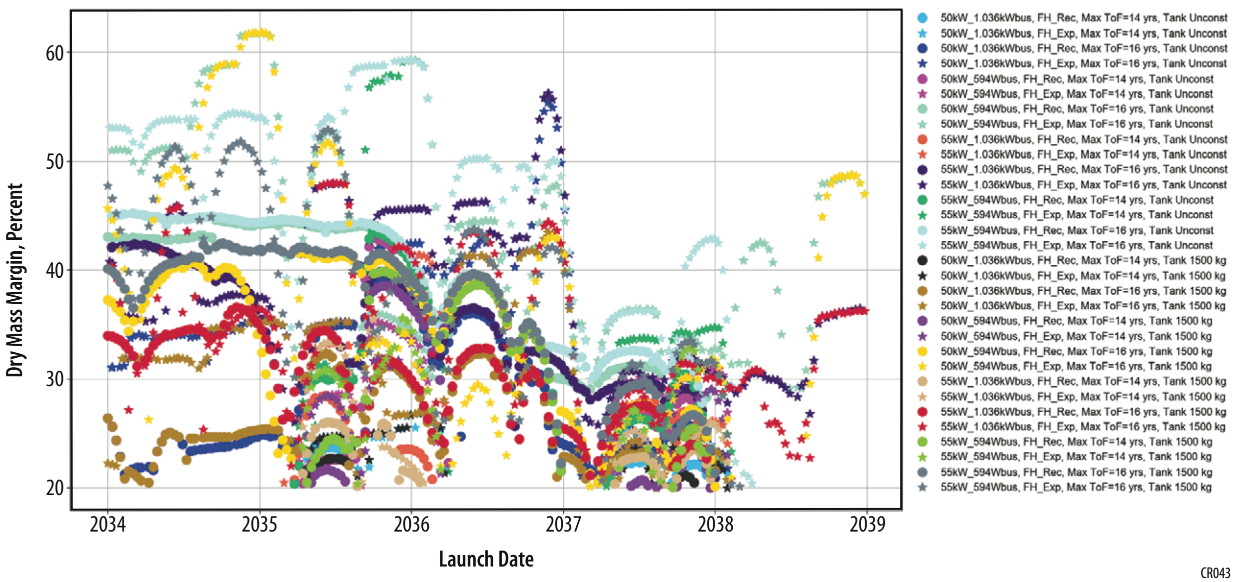

Figure 1-2: Trade results for target 67P for a variety of spacecraft-hardware configurations and maximum flight times.

The parameters considered in the set of results shown in the figure are:

- 50 and 55 kW solar array (BOL)
- 1.036 kW and 594 W reserved for the observatory (from the arrays) while thrusting
- Falcon Heavy Recovery and Expendable launch vehicle
- Max flight time of 14 and 16 years
- Xe tank of 1500 kg and unconstrained

The values considered for the observatory power needed while thrusting (with EP) is based on 1) a purely solar-powered spacecraft, needing 1.036 kW, and 2) the use of both a solar array and a Next Gen Mod 2 RTG capable of providing an average of 340 W to the observatory. In the case of the RTG, only 594 W is needed from the arrays for the observatory—leaving more for the SEP thrusters. The selected baseline spacecraft configuration and max flight time is shown in the figure as dark-gold circles, which has launch opportunities in 2035 and 2036 (the baseline) with 30% dry-mass margin, and in 2034 with 25% margin. Later launch years require more power for thrusting and/or the use of the Falcon Heavy Expendable.

Mission ConOps

Proximity Operations

The proximity operations subphase enables global mapping of 67P as well as high resolution mapping of potential sample sites over a two-year period during the four-year stay at the comet. Digital Terrain Maps (DTMs) built during proximity operations are dual purpose, utilized for terrain relative navigation as well as for science. Five preliminary sample sites are initially determined before final sample site selection based on science merit, engineering feasibility, and risk. The proximity operations concept follows recent small body rendezvous missions such as OSIRIS-REx and exploits lessons learned from Rosetta's rendezvous with 67P. Moreover, data collected during Rosetta is utilized where possible to reduce the time needed for characterizing the comet and the overall proximity operations duration. Changes to the surface of 67P since it was last observed by Rosetta are expected given that multiple perihelion passes of the comet will have occurred at the time of CROCODILE arrival. A breakdown of the subphases comprising the Proximity Operations mission phase is outlined in Table 1-5 and Mission ConOps timelines in Figures 1-6 and 1-7. The individual subphases are discussed briefly in the following sections.

Table 1-5: Proximity Operations Phase Overview

Subphase	Duration [weeks]	Range to Comet	Objective	Key Events	Navigation
Rendezvous maneuver-ing	6	1,000,000 km to 100,000 km	• Reduce speed relative to comet Initial detection • Refine approach traj. with comet OD updates • Initial rotation characterization	• SEP thrusting at reduced duty cycle • Chemical-based TCMs if needed for final targeting	• stellar optical navi-gation with NFOV to update comet ephemeris • DSN radiometric data for s/c
Far-field approach	7	100,000 km to 1,000 km	• Identify comet activity Rotation characterization • Start building low-res topography map to support landmark naviga-tion	• Trajectories through ~5 waypoints at variety of phase angles [60, 45, 30, 15, 0]	• NFOV camera for imaging
Near-field approach	6	1000 km to 200 km	• Initial low-resolution map delivery • Update mapping conops based on activity and any apparent changes to surface from Rosetta	• Trajectories through ~5 way-points at variety of latitudes	• NFOV camera for mapping images • Begin landmark-based relative nav
High-altitude mapping	8	50 km orbits	• High altitude mapping with NFOV camera • Generate initial shape model • Refine gravity mapping	• Multiple orbit ping pongs on sun side to ensure diversity of incidence, emission, azimuth angles from 50 km terminator home orbit	• NFOV camera to build nav map & WFOV for science mapping, NFOV for navigation images
Mid-altitude mapping	10	25 km orbits	• Mid altitude mapping with NFOV camera to generate preliminary hazard and science value maps	• Multiple orbit ping pongs on sun side to ensure diversity of incidence, emission, azimuth angles from 25 km terminator home orbit	• NFOV camera to build nav map & WFOV for science mapping, MFOV for navigation images
Low-altitude mapping	12	10 km orbits	• Low altitude mapping with NOFV camera to generate high-resolution map • Improve hazard and science value maps • Determine target sample sites for inspection	• Multiple orbit ping pongs on sun side to ensure diversity of incidence, emission, azimuth angles from 10 km terminator home orbit	• NFOV camera to build high-res map & WFOV for science mapping, MFOV for navigation images

Sample site survey	10	2 km close passes, 10 km home orbit	• Close inspection of target sample sites • Determine baseline sampling sites	• 3 site surveys (~3 weeks per site)	• WFOV camera images
Sample dry runs	12	~200 m, 100 m, 10 m waypoints	• Rehearsals to test navigation & control through final checkpoint maneuver prior to descent	• 3 rehearsals incrementally stepping to next waypoint	• Lidar and WFOV camera for TRN
Sampling	16	~200 m, 100 m, 10 m	• Safely descend to hover point and collect sample before get-away maneuver	• Traverse from terminator home orbit to checkpoint, match rotation rate at matchpoint, descend to hover point and collect sample	• Lidar and WFOV camera for TRN
Contin-gency	26	variable	• Time allocated for off-nominal surveying and additional sampling opportunities		

CROCODILE carries three instrument cameras that both map the comet and provide navigation measurements. These are PolyCam with a narrow field of view NFOV, MapCam with a medium field of view (MFOV), and SamCam with a wide field of view (WFOV). Camera specifications are outlined in Table 1-6. The NFOV is employed heavily for mapping at larger ranges while the MFOV is used mostly for navigation. The WFOV has large a footprint at larger ranges and can be used for low-resolution, large-area images and for navigation purposes at short range to the comet. In addition to the cameras, a lidar and laser altimeter are used for navigation range measurements when near the comet.

Table 1-6: Camera Parameters

	NFOV Camera	MFOV Camera	WFOV Camera
Field of View (deg)	0.8 x 0.8	4 x 4	44 x 32
Detector Size (pixels)	1024 x 1024	1024 x 1024	2592 x 1944
Focal Length (mm)	630	125	7.6
Pixel Size (um)	8.5 x 8.5	8.5 x x8.5	2.2 x 2.2

Rendezvous

The end of interplanetary cruise and start of proximity operations begins as 67P becomes observable with the narrow field of view (NFOV) camera. During the rendezvous subphase, beginning at roughly one million kilometers from the comet, the thrusting cadence is reduced to allow for regular observations of the 67P, and stellar optical navigation is used to update the comet's orbit solution and subsequent trajectory plans. At approximately 100,000 km, SEP thrusting ends and the spacecraft transitions to the approach subphases.

Approach

Far-field approach begins with the phase angle (Sun-67P-spacecraft angle) less than 90 degrees and a series of five chemical-based maneuvers to reduce the phase angle to zero as illustrated in Figure 1-3. An open trade is to maneuver with SEP instead of chemical propulsion for Far-field Approach. During far-field approach a campaign is conducted to identify any comet activity and update the rotation characterization since Rosetta. The NFOV camera gathers images to build an initial low-resolution map.

At roughly 1000 km to the comet the Near-field Approach subphase begins at a relatively low phase angle (<30 degrees). Five maneuvers take the spacecraft on sequence of hyperbolic trajectories through a series of waypoints, enabling imaging of a variety of comet latitudes at different phase angles. NFOV images during this subphase can be used for initial testing of landmark navigation with the low-resolution map from Far-field Rendezvous. That initial map based on Far-field approach imaging is improved upon with higher resolution images collected through the Near-field Rendezvous Subphase. Additionally, any necessary refinement of the proximity operations plan given changes to the comet since Rosetta are made as the spacecraft transitions to a near-circular mapping orbit, starting at approximately 200 km range.

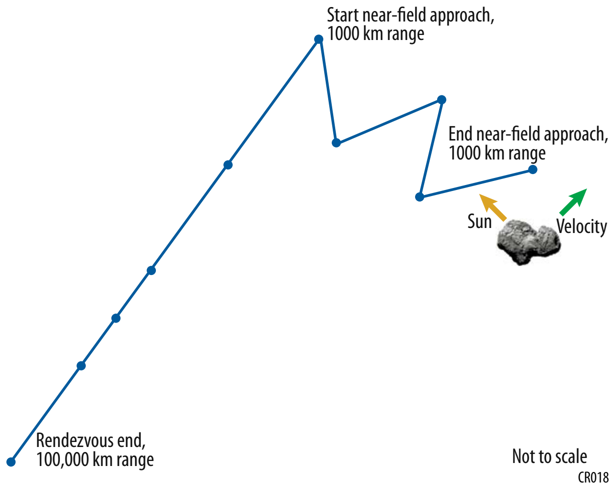

Figure 1-3: Approach subphase trajectory concept

Mapping

Three mapping subphases enable the construction of increasing resolution DTMs, which allow for close-range operations and determination of target sample sites. Following Far-field approach, the spacecraft transfer into a 50 km High-altitude Mapping orbit. The initial orbit is near-circular and roughly in the terminator plane of 67P for long-term stability given solar radiation pressure. Excursions, called ping-pongs, are made from this home terminator orbit on the sun-side of 67P. These ping-pongs occur over different orbital nodes to image the comet from a variety of incidence, emission, sun azimuth, and spacecraft azimuth angles as depicted in Figure 1-4. This diversity of viewing and lighting geometries enables high resolution terrain mapping through stereophotoclinometry (SPC).

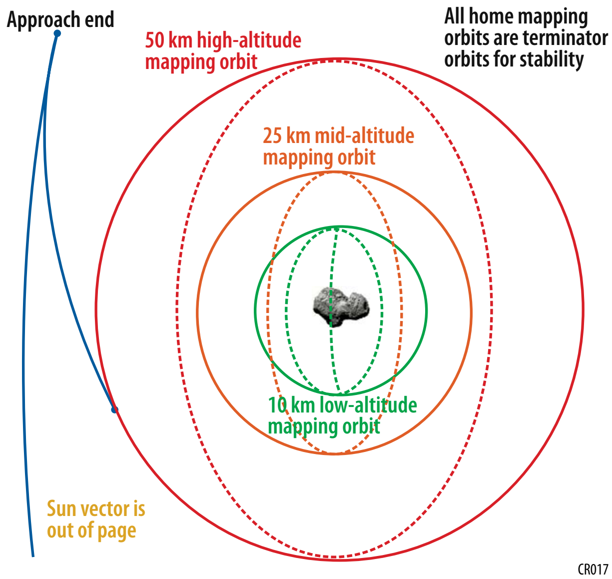

Figure 1-4: Mapping subphase orbit concepts

The DTM built during High-altitude Mapping is applied for landmark navigation during Mid-altitude Mapping in 25-km orbits. As in High-altitude Mapping, a terminator orbit serves as a stable base for Mid-altitude mapping ping-pong trajectories at different orbital nodes. The roughly 50-cm resolution map constructed during Mid-altitude Mapping is applied to build global safety and science value maps, which are used to develop initial target sample locations.

After 10 weeks in the Mid-altitude Mapping subphase, the spacecraft lowers to a 10-km Low-altitude Mapping orbit, with an imaging focus on potential sample locations. Final determination of candidate target sample sites are made during Low-altitude Mapping using the roughly 20 cm DTM and associated update of safety and science value maps given images from 10 km range. The maps built during Low-altitude Mapping serve as the basis for navigation during sample site surveying and final sampling operations. Twelve weeks are allocated to mapping and candidate site selection in the Mid-altitude Mapping subphase.

Sample Site Survey and Sample Dry Runs

From the 10-km terminator orbit a series of hyperbolic flybys with close passes at 2 km from the comet surface are executed to capture high-resolution images of the candidate sample sites. Each of these flybys, illustrated in Figure 1-5, are passively safe if maneuvering capability is lost following a spacecraft fault. After flybys of each candidate sample site, the final selection of two sample sites is made.

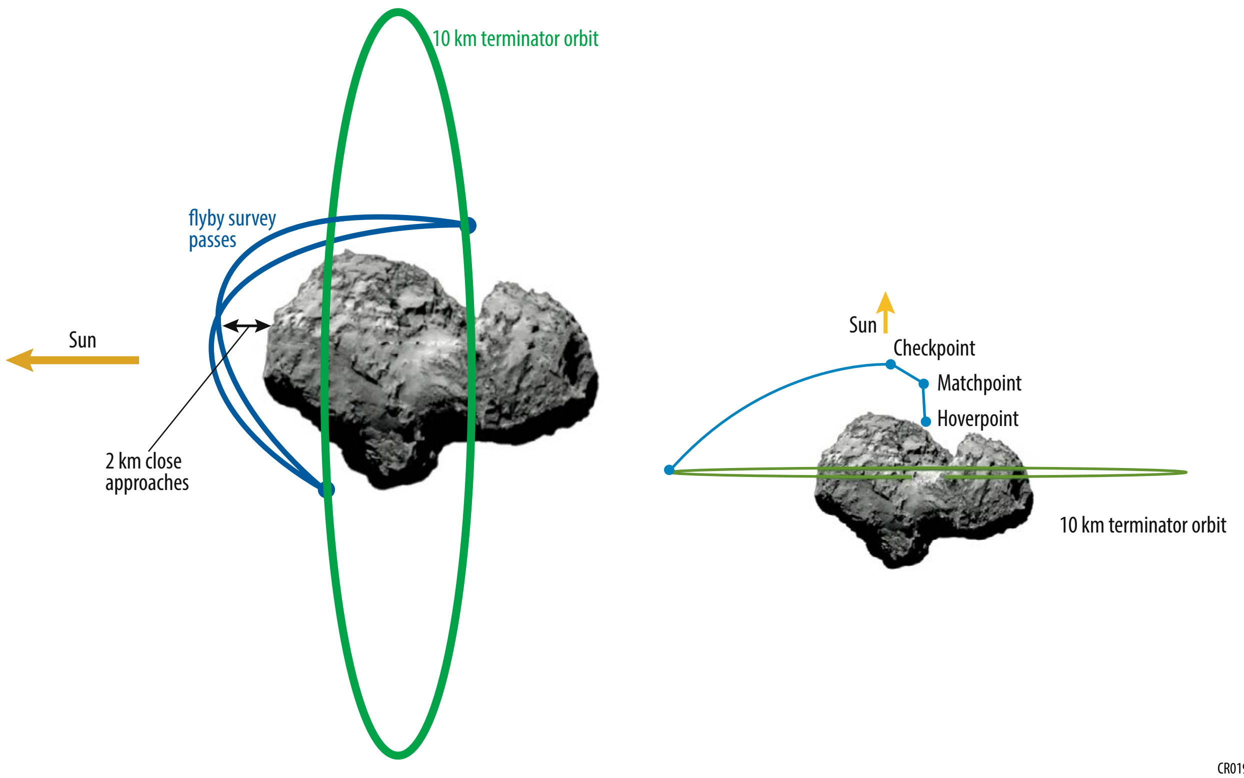

Figure 1-5: Example sample site survey and Sample Dry Run trajectory concepts

With the sample sites determined, a series of dry runs are conducted to each sample site. During the dry runs the spacecraft departs from the 10 km home terminator to a checkpoint waypoint. This hyperbolic trajectory is passive safe without maneuvering at the checkpoint. On the first dry run, no checkpoint maneuver is executed, and the spacecraft continues on a passively-safe, hyperbolic orbit past the comet. After returning to the home terminator orbit, a second dry is commenced to transfer to the checkpoint, and maneuver down to the matchpoint, where the spacecraft is directly over the sample site. Instead of matching rates with comet rotation, an abort maneuver is executed to send the spacecraft safely away from the comet on an escape trajectory. The spacecraft then returns again to the 10 km home terminator orbit. In the final dry run, the spacecraft proceeds through the checkpoint, executes a maneuver to match the rotation rate of the comet at the matchpoint, and then descends down to the roughly 10 m hover point before aborting on safe, hyperbolic escape. After returning back to the 10 km home terminator orbit, the spacecraft executes the full descent sequence to the hoverpoint and collects the sample. This sequence is repeated for the second sample site.

Mission ConOps/Timelines

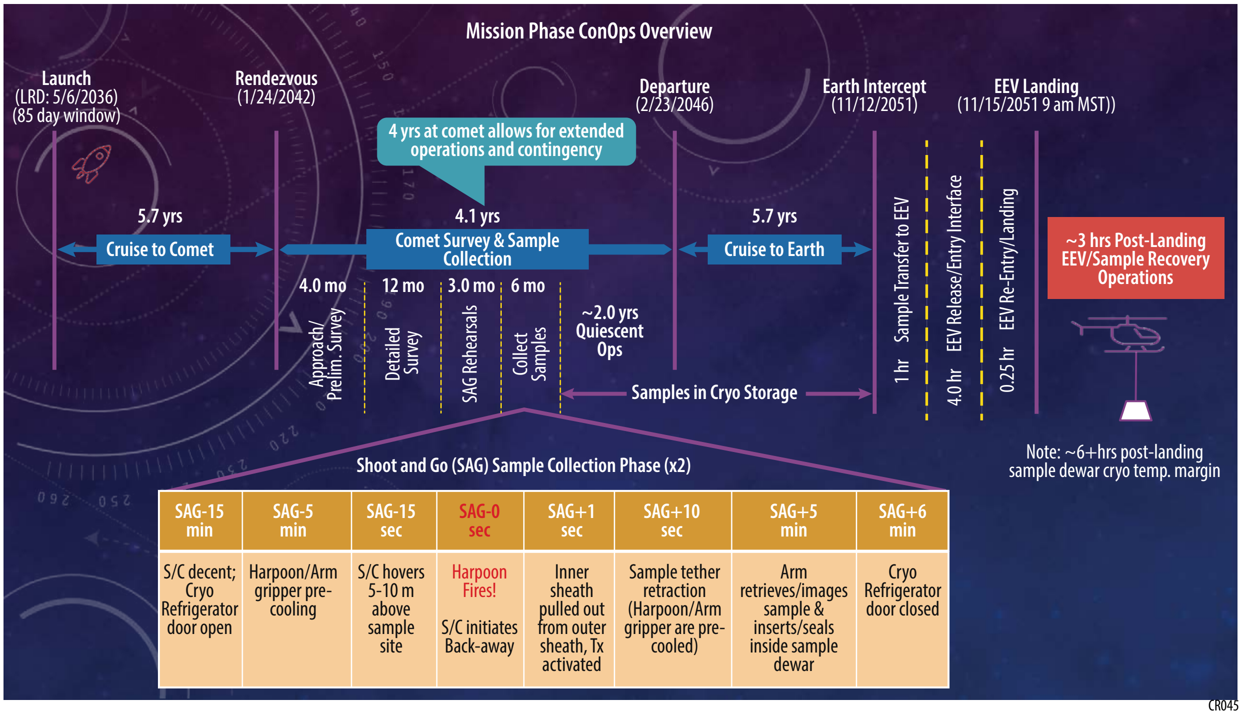

Figure 1-6: SAG Sample Collection Phase Timeline

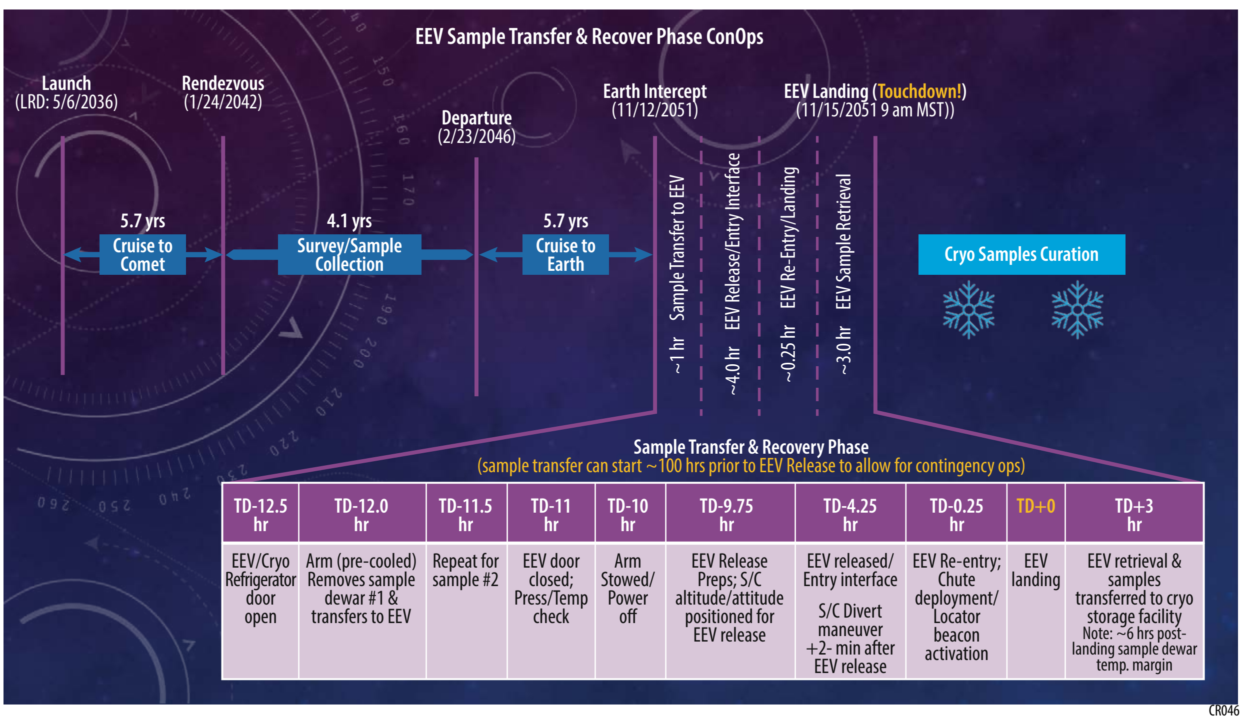

Figure 1-7: Sample Transfer and Recovery Phase

The Ground Ops Recovery plan is based on the Stardust ground recovery experience; JPL's Sample Return Primer & Handbook. A morning landing is planned in an ~80 km long and 25 km wide (50 x 15.5 mi) ellipse at the Utah Test and Training Range (UTTR) Hill AFB radars track the descent of the EEV and landing position accurate to within ten meters. The EEV itself carries a UHF radio locator beacon for locating the EEV in adverse weather/terrian conditions. Helicopters are the primary means for EEV location, extraction and transport operations with ground vehicles as back-up, staged at nearby Michael AAF. The worst-case distance/timeline for air ops is ~3 hrs from landing to sample transfer into cyro storage at staging site. The sample dewar container has~ 6+ hrs of margin for maintaining cyro temperatures inside the EEV upon landing and transfer into cyro storage. There are several options to further minimize sample recovery time and risk to warming of the cryo samples..e.g. pre-positioning GSE assets, mobile cleanroom and portable cyrogenic refrigerator within the landing zone.

2.0 COMMUNICATION SYSTEMS

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. For this mission a dual frequency system was determined to meet the needs and constraints. This system can be separated into specific craft/vehicle sections. The main spacecraft communications system consists of a General Dynamics (GD) Small Deep Space Transponder (SDST), Traveling Wave Tube (TWT) with electronic power conditioner (EPC), high power isolators, a parabolic high gain antenna (HGA), a medium gain antenna (MGA), and two low gain antennas (LGAs), and an S-Band Slink-PHY transceiver with a patch antenna. The EEV communications system has a UHF beacon transponder with 3 UHF loop antennas. The harpoons each have an S-Band Slink-PHY transceiver with a patch antenna. These components have redundancy where necessary and are connected with RF cabling, waveguides, switches, and diplexers to allow for options in the communications path. These systems are designed to use Ka-, X-, S-Band, and UHF for different use cases as they provide the best option for high data rates and performance within the mission constraints.

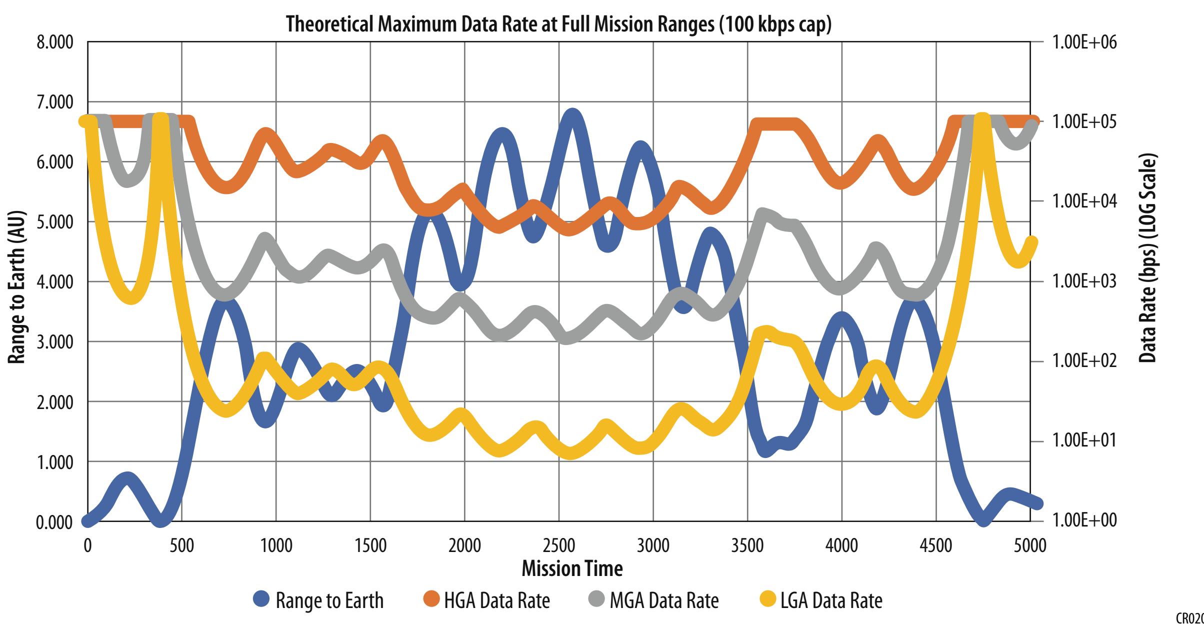

Figure 2-1: Max data rate for each antenna system

Figure 2-1 shows the max data rate for each antenna system as well as the range to Earth as a function of mission elapsed time. Data rate is measured on the right axis and is scaled logarithmically. This chart assumes a fixed power input to the communications systems as well as perfect pointing, 34 m DSN receiving system, no obstructions, nominal weather conditions, no solar interference, and the data rate is capped at 100 kbps for purposes of scale (max data rate may be less than or greater than 100 kbps as determined by mission requirements).

The systems will always be in receive mode but to save power the transmitting components can be turned off for long durations when/if they are unnecessary. During the cruise phase of the mission the transmit will need to be turned on periodically for the purposes of ranging with the DSN and for telemetry updates. As the spacecraft approaches the comet there will be a long duration mapping operation with the need for high data rate transmission. Once a target location is chosen, the actual sampling mission will need to be highly automated as the two way lightspeed delay will make real-time command and control impossible. The harpoon is equipped with a small S-Band transmitter which will relay critical temperature and other data to the main spacecraft to help in determining the adequacy of the sample captured and its physical state. The higher quality data and images taken during the sampling operation can be transmitted later as the available data rate may not be high enough for it to be transmitted in a quickly. During the return mission to Earth the same power saving measures may be taken if there is no more data to be transmitted. Upon close approach to the Earth the EEV will be released and radar tracking will commence soon after. An accelerometer-based switch will trigger the UHF beacon transmitter to turn on upon sensing the spacecraft is low in the atmosphere. The beacon will continue to transmit for 16 hours so recovery operations can ascertain the location of the EEV even in inclement weather conditions.

3.0 SPACECRAFT AVIONICS

Figure 3-1 shows the CROCODILE spacecraft block diagram. The CROCODILE avionics has a High Voltage Electronics Assembly (HVEA) which has three units: A High Voltage Relay Assembly (HVRA), a High Voltage Down-Converter (HVDC), and a High Voltage Control Electronics (HVCE). These units take in the solar array and battery power, and supply the high voltage power required by the Electric Propulsion System and the 28V primary power required by the rest of the spacecraft. They are based on JPL heritage units from the Dawn Mission (Launch 2007), but are up-scaled in terms of power handling capability. These units are internally redundant.

The avionics also has a 28V Power Distribution Unit (PDU), a Command and Data Handling unit (C&DH), and a Propulsion / Mechansism Electronics unit (PME). The PDU, C&DH, and PME are implemented with GSFC's Mustang Avionics. The Mustang Avionics will fly on the PACE mission in 2023. Figure 3-2 shows pictures of the PDU and C&DH. The PDU, 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 PDU consists of 4 types of cards. Its PSE Monitor Card provides PDU control and telemetry acquisition functions. Four High Current Power Output Modules provide four 15 Amp outputs, eight 7.5 Amp outputs, and sixteen 3 Amp outputs. One 15 Amp output is required for the Arm Electronics Unit and another 15 Amp output is required by the Harpoon Electronics Unit. Four 7.5 Amp outputs are required for the reaction wheels, one 7.5 Amp output is required for the X-Band TWTA, and one 7.5 Amp output is required for the Ka-Band TWTA. Three Low Current Power Output Modules have sixteen 3 Amp outputs each. Those outputs along with sixteen 3 Amp outputs on the High Current Power Output Modules provide a total of sixty four 3 Amp outputs. Fifty one 3 Amp outputs are required, leaving a margin of 25%. Lastly, a Low Voltage Power Converter provides the secondary voltage power required by the PSE cards.

The C&DH consists of 4 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 (12 are required), 8 pressure sensor inputs (4 are required), and 138 temperature sensor inputs. A Communication Card provides the RF Communication system interfaces. Lastly, a Low Voltage Power Converter provides the secondary voltage power required by the C&DH cards.

The PME consists of 3 types of cards. Three ACS Valve Drive Cards provide a total of twenty four thruster outputs (16 are required), and 12 latch valve outputs (4 are required). A Deployment Module provides 8 actuation outputs, 4 are required for the solar array deployments and 4 are required for the Earth Entry Vehicle release.

The avionics also consists of mechanism control electronics including a Solar Array Gimbal Control Electronics unit (SA GCE), an Electric Propulsion Gimbal Control Electronics unit (EP GCE), Refrigerator Door Electronics unit (RDE), and an EEV Accessibility Door Electronics unit (ADE). The SA GCE and EP GCE are implemented with Moog Gimbal Control Electronics (GCE). The Moog Gimbal Control Electronics flew on the NICER Mission. Figure 3-2 shows a picture of the Moog GCE. All of the mechanism control electronics are internally redundant. Their backup cards fly as cold backups. The mechanism control electronics units are functionally 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.

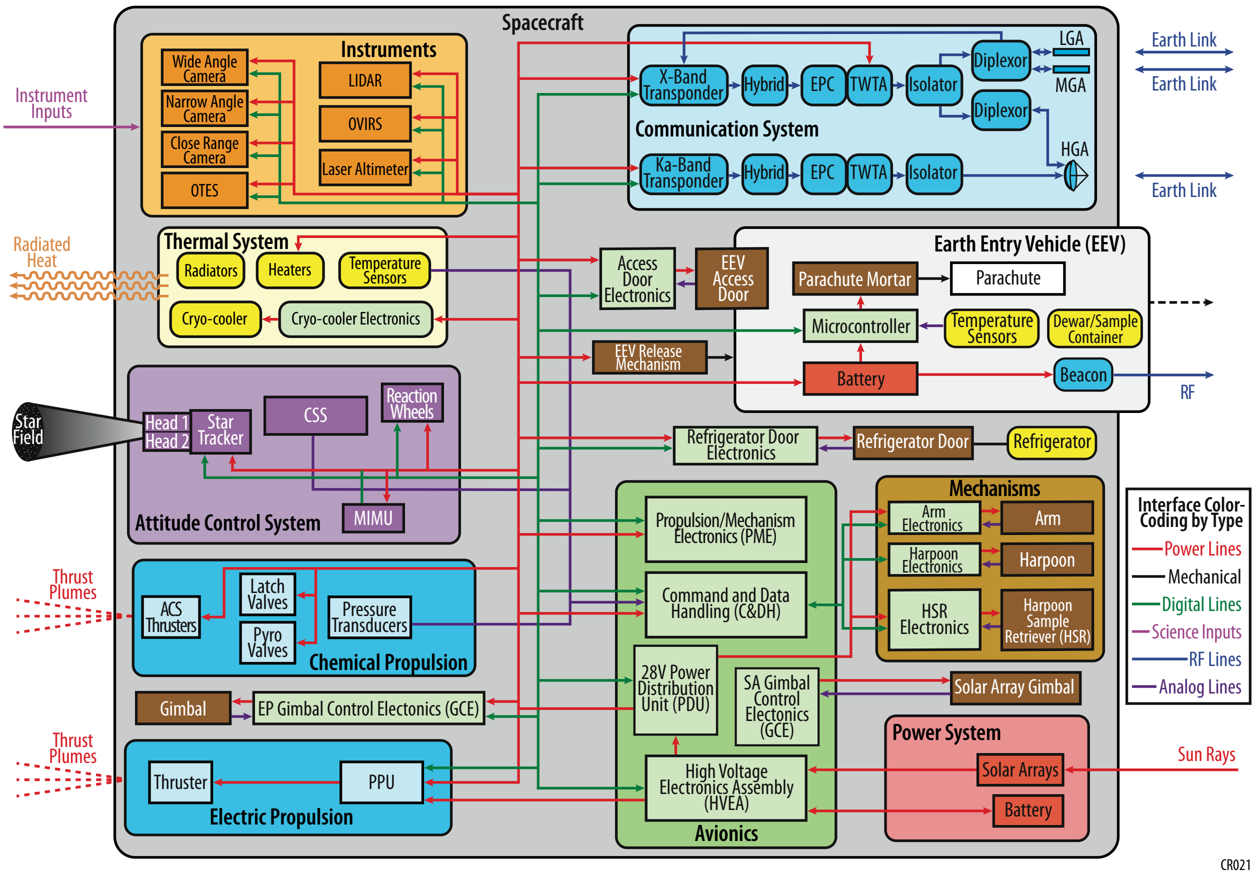

Figure 3-1: Spacecraft Block Diagram

Figure 3-2: (left) 28V Power Distribution Unit (PDU), (center) Command and Data Handling unit (C&DH), (right) Gimbal Control Electronics (GCE)

4.0 THERMAL SUBSYSTEM

The CROCODILE thermal environment varies from near Earth, to an outbound aphelion near 5.7 AU, before returning to meet the comet at 1.3 AU after 5.5 years. After 1 year of approach/proximity ops/landing, then collecting samples while riding the comet out to 5.7AU, the return Cruise Phase begins, with aphelion at 5.7 AU before returning to Earth 5.7 years after departure. During this 15+ year mission, the external environment (solar intensity) will range from 1420 W/m^2 to 59 W/m^2, a reduction of 95%.

During the mission, the operations will vary from only S/C bus operations during the cruise phases, while the full payload suite will be operated only during the proximity ops period. The critical cryocoolers, will operate from launch throughout the mission until the samples are transferred into the EEV just prior to separation for Earth re-entry. Solar Electric Propulsion will operate for most of the Cruise phases, with large, constant heat loads from the PPE electronics.

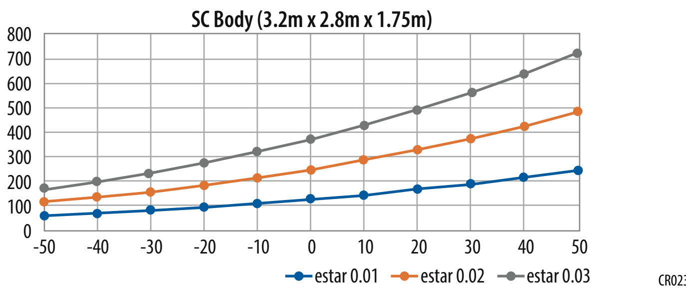

Figure 4-1: Heat Loss Thru MLI

At such distances from the sun with only solar based power resources, conserving energy is paramount so distributing the heat from operating electronics and motors throughout the S/C bus will minimize electrical heater power needed - heat spreading must be a primary systems design goal. Conceptually, this large bus (~40m^2) will lose up to ~550W even if fully blanketed and could gain nearly 140W during the near-Earth flyby. When the combined solar heating and dissipative heat load exceeds the MLI heat loss, radiator area is required to keep temperatures from exceeding the maximum and will include louvers to "close" the radiators in colder environments, greatly reduces heater power needed.

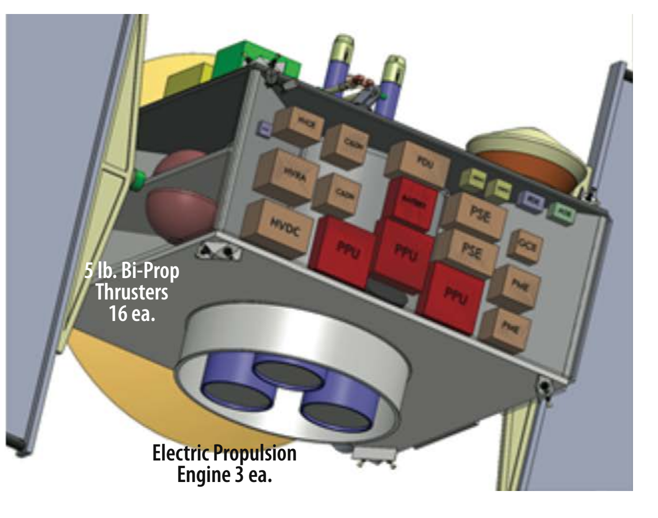

Figure 4-2: CROCODILE Mechanical Configuration Layout

The baseline thermal control approach is to package as much of the temperature sensitive equipment as possible within the S/C bus, and distribute the dissipative heat generated (Figure 4-2), to passively control the bus and minimize the need for electrical heater power.

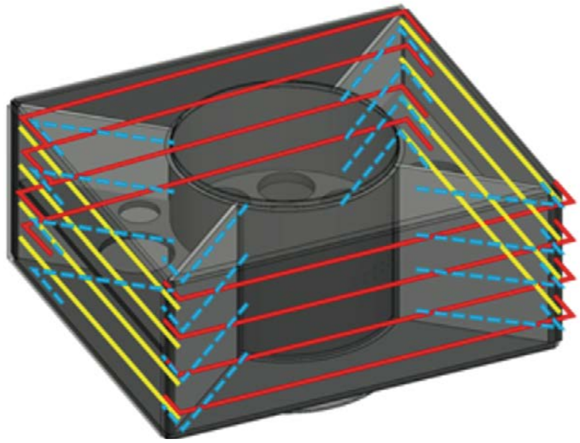

Figure 4-3: Thermal Concept for Heat Pipe Distribution Network

Distributing the heat from the operating electronics to all four sides and top/bottom decks requires conductive heat paths (aluminum structural panels). Embedded heat pipes in the 4 sides can be linked to heat pipes along the radial bulkheads to distribute heat to the propulsion tanks (Figure 4-3), as needed. Optimizing the component packaging to evenly distribute the heat distribution on all 6 panels to maintain an "operational average" on each throughout the mission will minimize the number of heat pipes needed. If needed, "vertical" heat pipe headers may be used to link the spreader pipes and would work in reflux during 1g testing. Similar links to the upper and lower deck heat pipes will complete the WEM concept, maintaining 0°C to 30° within the bus.

The propulsion subsystem, having the least tolerance to cold temperatures, benefits from this WEM approach with the tanks located in the center of the bus and requiring direct heating only for the lines and valves near the thrusters.

S/C bus: an energy balance on the large S/C bus estimates radiator area and heater power during the mission, with varying power levels and external environments.

- *Hot Environment:* Radiators were sized for the hot case Earth flyby and use louvers to reduce heat loss in the colder environments of the mission. Presuming the bus dissipates 2012W (MEV) during Cruise/Prop in the hottest near-Earth environment (flyby at 0.84 AU), 6.1 m^2 of louvered radiator area will maintain the bus at 30°C. This meets the coldest expected maximum temperature requirement (batteries).
- *Cold Comet Environment:* as the solar intensity diminishes, temperature will gradually reduce until the louvers begin to close. In the cold prox ops environment, with less power loads on the bus, the louvers will close to require no heater power while maintaining the WEM >0°C, which meets the warmest expected cold temperature limit (chemical propulsion sub-system).
- *Coldest Return Cruise Environment:* in the coldest extreme environment during the Return Cruise aphelion, with minimal bus power of <400W, ~205W of heater power will be needed even with the louvers fully closed.

Propulsion: with chemical propulsion and SEP tanks located within the WEM, the only heater power is estimated to be for the external thruster valve/line segments, based on typical satellites.

Payloads: with so many complex geometries and operating modes for this hardware, only coarse estimates were made using a "heater power is 50% of the operating power" metric.

- *Cryogenic:* the cryogenic aspects of this mission are discussed in Section xx
- *Cryocoolers:* the unique cryogenic aspects of this mission require two large cryocoolers that are operated for the entire mission to maintain the stored cryogen needed for the Earth return phase, provide cold storage for the harpoons to minimize heat transfer during sampling operations, and provide cold storage for the return cruise. The two cryocoolers are based on TIRS-2, along with their 1.12m^2 radiators. Louvers are added to reduce heater power when not operating at maximum capacity.
- *Sampling Operations:* during sampling & retrieval must ensure the sample retrieval operations do not expose the samples to excessive parasitic or local environmental heating as they are transferred to cold storage for the return phase. Operational timeline, in shadow, materials selection (conductive heat transfer) for the Robot Arm end effector that contacts the sample container, must be considered during the design phase.
- *Sample return:* an initial assessment of the heat up (post landing) in UTTR was done, assuming the EEV was laying in full sun surrounded by 120 °F air and desert. The SRC was presumed to be surrounded with 3" of titanium foam Insulation, with a thermal conductance of 0.07 W/m-K. Without a stored cryogen, the sample temperature was estimated to exceed 120 °K in approximately 3 hours. The 4.5 kg of solid Argon in each SRC will absorb enough heating to maintain the samples <120 °K for ~ 8 hours.

5.0 ELECTRICAL POWER SYSTEM

The CROCODILE EPS consists of solar arrays, batteries and electronics boxes to support three mission elements: the spacecraft, Earth Entry Vehicle (EEV), and harpoon. All components are TRL 6+.

The spacecraft power system consists of solar arrays, a secondary battery, and supporting power electronics. The CROCODILE solar array size is driven by the electric propulsion requirements of 50 kW at 1AU BOL and 44.7 kW EOL. TJGaAs solar cells with bare cell efficiency of 29.5%, a solar constant of 1352 w/m^2 at 1AU, operating temp at 140 °C, and Space Environmental Effects and Education System (SPENVIS) solar array degradation factors yield a panel output of 260W w/m^2 and a total array area of 192 m^2. Two tracking Roll Out Solar Array (ROSA) wings of 4.5 m by 21 m each provide the necessary power. A high energy density 155AH Li Ion battery is used to support loads at encounter. The Power System Electronics (PSE) will be a heritage 28VDC battery dominated bus included as cards in the avionics package. The PSE will control battery charging and power distribution. Solar Arrays are gimbaled to minimize perturbance forces during Comet proximity operations.

The CROCODILE EEV power system consists of a primary (non-rechargeable) battery and supporting power electronics. The power system configuration is driven by a requirement to support a load of 7W for 16 hours after release from the Orbiter. LS26500 7700 mAh 3.6V LithiumThionyl Chloride cells are used in a 4 series, 3 parallel (4s3p) to provide 23AH of energy at 14V. Heritage electronics will be included in the EEV avionics package.

The CROCODILE harpoon power system consists of a primary (non-rechargeable) battery and supporting power electronics. The power system configuration is driven by a requirement to support a load of 25W for 0.25 hour. ER14250M 750 mAh 3.6V Lithium Thionyl Chloride cells are used in a 4 series, 1 parallel (4s1p) configuration to provide 0.75AH of energy at 14V. Heritage electronics will be included in the harpoon avionics package.

CR026

Figure 5-1: CROCODILE Electrical Power System

6.0 PROPULSION SYSTEM

CROCODILE has two propulsion elements, a Solar Electric Propulsion stage, which is used for long term cruise to and from comet and a chemical propulsion system for proximity operations around the comet. The EP has 3 Nex-C Gridded ion engines (Figure 6-1), in a 2+1 configuration and 7 propellant tanks (PN 80458-1, 16.5” dia x 44.6” long, 20.4 kg, 7,300 in^3) holding 1500 kg of Xe propellant. A simple flow schematic of the SEP system is shown in Figure 6-2. The engines have been qualified and delivered to the DART mission. All SEP system components are at least TRL 7.

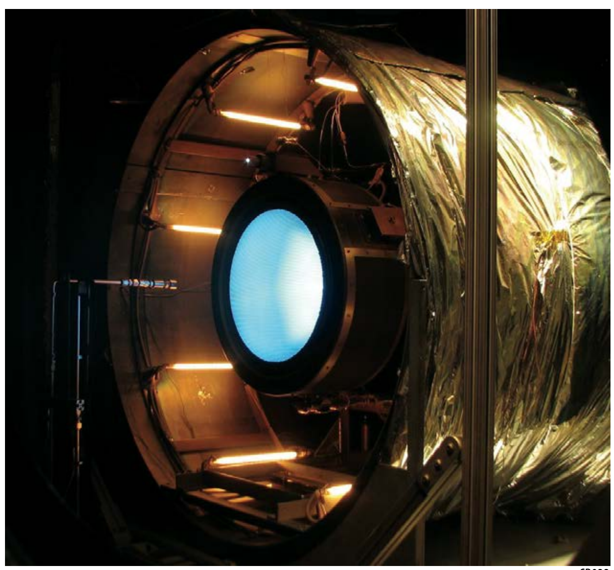

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Figure 6-1: NEXT Thruster

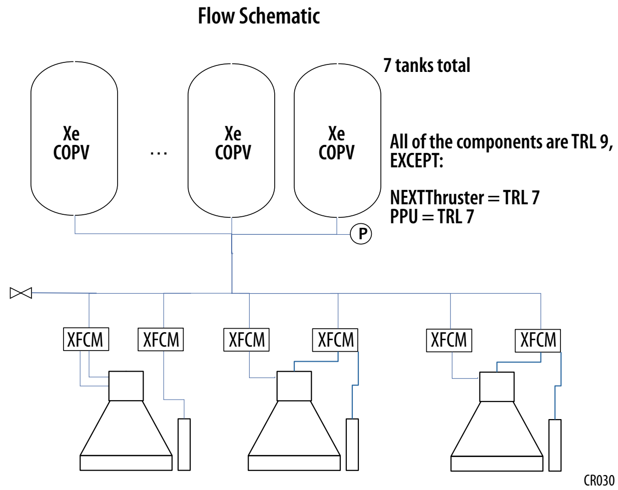


Figure 6-2: SEP System Flow Schematic

The Chemical propulsion system is a high heritage monomethyl hydrazine/MON-3 bi-prop system with 16 DST-3 thrusters (Figure 6-3). The biprop system flow schematic is shown in Figure 6-4.

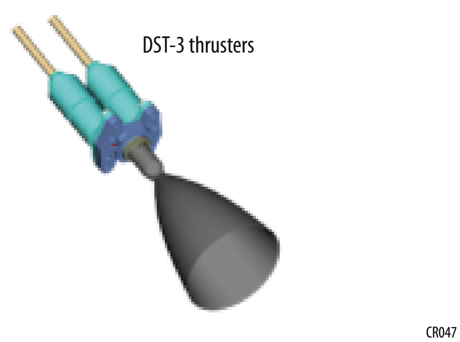


Figure 6-3: DST-3 thrusters

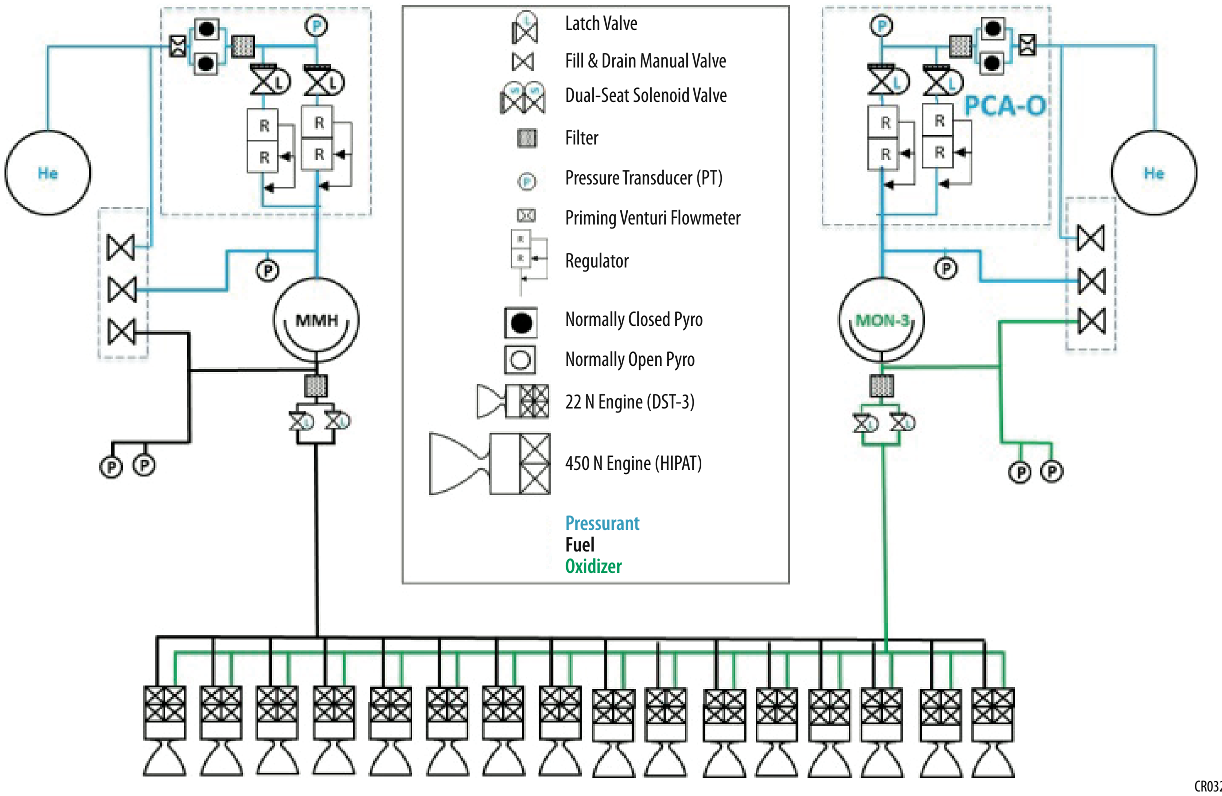


Figure 6-4: Biprop chemical propulsion system flow schematic

7.0 ATTITUDE CONTROL SYSTEM (ACS)

CROCODILE's ACS is based on similar sampling missions that require specific knowledge and control requirements for proximity and sampling operations. The driving requirements, which are listed in Table 7-1 enable the rendezvous, approach, mapping and sampling operations. Several subsystem trades were conducted to design resulting in the ACS sensors and actuators suite for navigation and spacecraft control/stability as described below and illustrated in the functional block diagram (Figure 7-1). The actuator suite consists of two redundant sets of eight 1 lb thrusters for 3-axis control, which can be commanded at the same time for additional authority. The reaction wheels will be used for fine pointing while not thrusting and momentum dumps. The sensor suite was selected to provide nominal attitude and Terrain Relative Navigation (TRN) position measurements, which can produce high precession imaging of < 1km. During Mapping, and sampling, the TRN components, which consist of a Lidar instrument, along with the Laser Altimeter and optical/IR camera (Malin ECAM) for hazard avoidance and positional fixes, will provide the needed attitude and position knowledge and control. The Attitude portion of the sensor suite consist of the standard ACS components (star trackers, gyros and coarse sun sensors). During cruise and approach, the ACS will utilize the standard ACS components, which will provide the required approach knowledge and control.

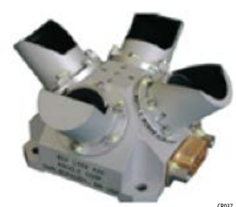
Table 7-1: ACS Driving Requirements

Phase	Control	Knowledge
- Approach/ Attitude	1 arcmin/axis 3 sigma	30 arcsec/axis. 3 sigma
- Mapping/ Attitude	5 arcmin/axis 3 sigma	1 arcmin/axis. 3 sigma
- Hover & sample Acquisition (Position height of 10m)	0.5 m 3 sigma	0.1 m 3 sigma
Mechanical	Center of Mass (CM) Control < 3 cm from sphere of CBE CM; Actuators are evenly spaced about CM	
Propulsion	Thruster alignment < 0.5 on all thrusters; Balanced Draining	

The ACS modes, which consists of Launch Rate Null, Sun Acquisition, Delta H, Delta V, and Mission are based on high heritage ACS control modes. These modes will utilize the ACS/TRN components via high heritage data ingest and data out gest algorithms for the various sensors and actuators. Figure 7-2 provides the baseline sensors and actuators for the ACS/TRN. During the Sample and hover phase, the ACS will use the thrusters as the baseline actuators but switch to a combination of wheel and thrusters to safe the spacecraft in the presence of a fault. It should also be noted that the fault detection and correction algorithms are also based of heritage FDC algorithms.

Baseline Hardware Description:

- Coarse Sun Sensor Assembly (x3 sets)
  - » Each assembly has > 2pi steradian angle
  - » Mount on body instead of solar array tip


CR037

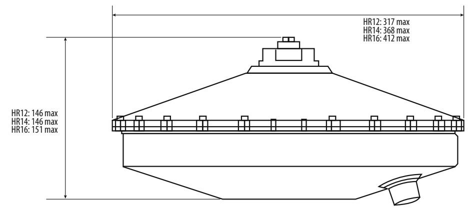
- Novatel IMU-LN_200 (x2)
  - » Northrop Grumman LN200s Fiber Optic Gyro (FOG) + Accelerometer
  - » 5 V +15V -15V, 12 W, RS-422
  - » 9 cm diameter, 9 cm height, 748 gram;


CR038

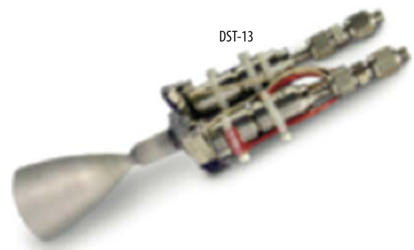
- DTU Micro Advanced Stellar Compass (uASC)
  - » Digital Processing Unit: mount anywhere on bus (x1)
  - » Camera Head Unit (x2)
  - » Each include a customized baffle and MEMS gyro.
  - » Accuracy: boresight=30 as, transverse=3 as
  - » All Camera pointed near cold direction to avoid FOV issues


CR039

- Reaction Wheels (1 Nms)
  - » Honeywell HR 16-75Nms, 0.2Nm
  - » Momentum buildup should be small (needs to be verified)
  - » Thruster firing for Momentum dump needs to be less than 2.5% duty cycle (assuming 10Hz Controller)
  - » 75Nms*2.5%=1.875 Nms
  - » 88N (4 5lb thr)* 0.1sec (ctrl cycle)* 0.2m (CM offset/residual)=1.76Nms



- ACS Thrusters
  - » 8 1 lb (2x) ➔ 16 ACS thrusters for 3 axis translation control
  - » Electric thrusters for Delta V



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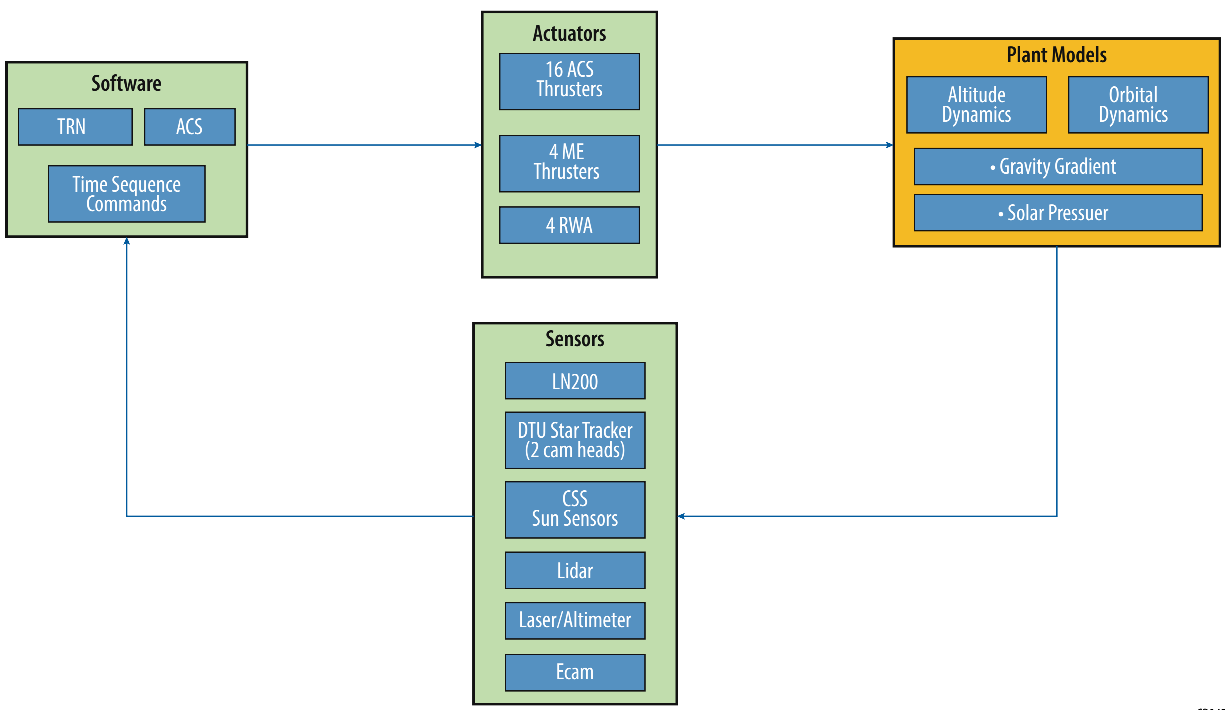

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Figure 7-1: GNC Block Diagram

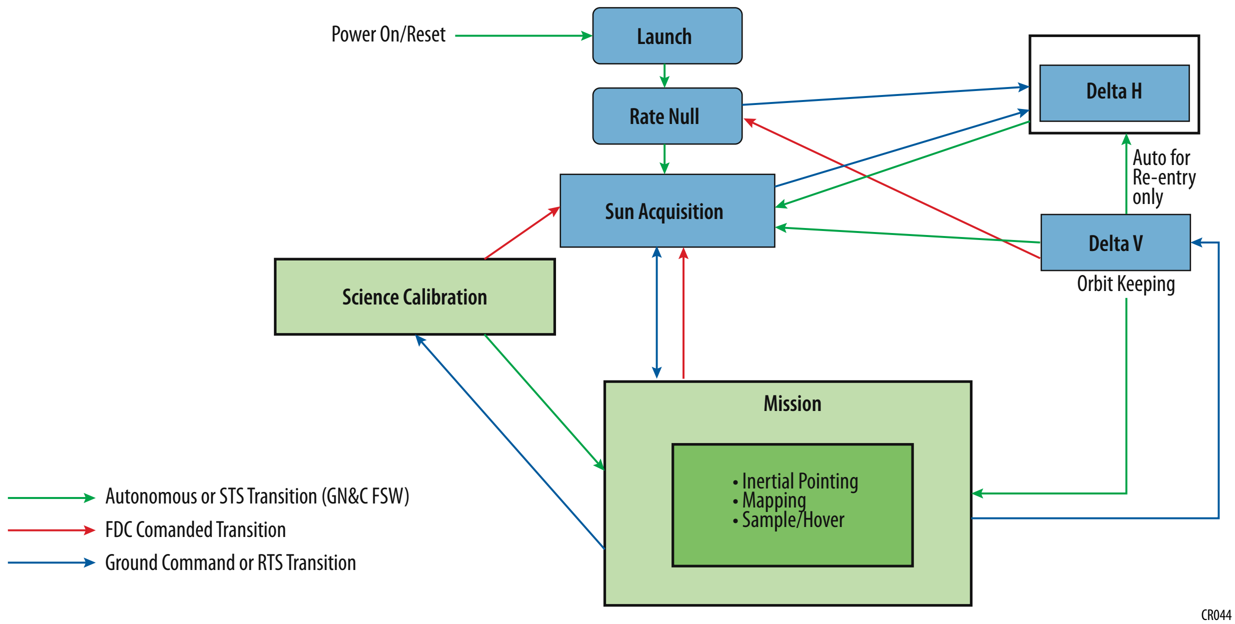


Figure 7-2: GNC Mode Diagram

8.0 SPACECRAFT STRUCTURES

The CROCODILE spacecraft structure is made from Aluminum honeycomb with Aluminum face sheets, drawing heritage from multiple past planetary missions. Avionics and other heat dissipating are coupled to the structure allowing for conductive dissipation at the radiations, and eliminating the need for additional heat pipes. The overall spacecraft need fits inside a Falcon Heavy fairing, utilizing a mounting interface for solar panel clearance (Figures 8-1 – 8-4).

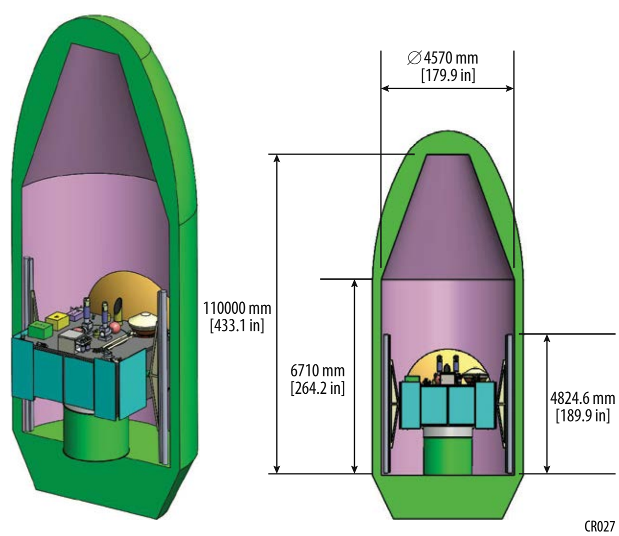
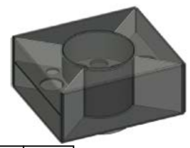


Figure 8-1: Spacecraft in Falcon Heavy Fairing



Item No.	Part Name	Material	Mass Kg ea.	Qty ea.
1	Control _Cylinder_Array		32.28	1
	Control_Cylinder	Aluminum 6061-T6	27.58	1
	Central_Cylinder_Angle	Aluminum 6061-T6	0.59	8
2	SC_Bus_Side_Panel	2 in. alum core .04 alum f/s	38.49	2
3	SC_Bus_Radial_Panel_Assy		12.09	4
	SC_Bus_Radial_Corner_Post	Aluminum 6061-T6	1.15	1
	SC_Bus_Radial_Cylinder_ Interface_Bracket	Aluminum 6061-T6	1.22	1
	SC_Bus_Radial_Panel	1.5 in. alum core .06 in. alum f/s	9.72	1
4	SC_Bus_Side_Panel_Short	2 in. alum core .04 in. alum f/s	33.52	2
5	Deck_Upper	2 in. alum core .04 in. alum f/s	58.19	1
6	Deck_Lower	2 in. alum core .04 in. alum f/s	58.19	1
7	PAF	Aluminum 6061-T6	72.48	1
8	CC-TankSupport_Panel	2 in. alum core .04 in. alum f/s	12.72	1
9	Center Deck	2 in. alum core .04 in. alum f/s	20.48	1
10	Chem_Prop_Tank_Support_Plate	2 in. alum core .04 in. alum f/s	11.88	2

Figure 8-2: Spacecraft Bus Structure Details

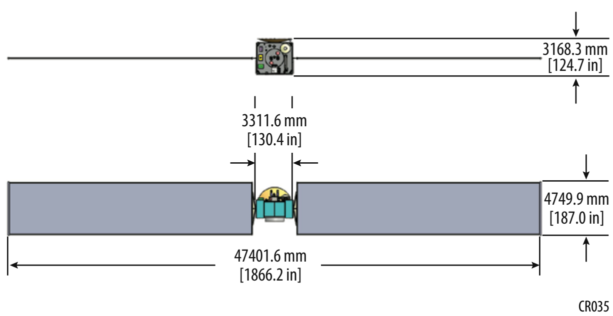


Figure 8-3: CROCODILE Bus Dimensioned

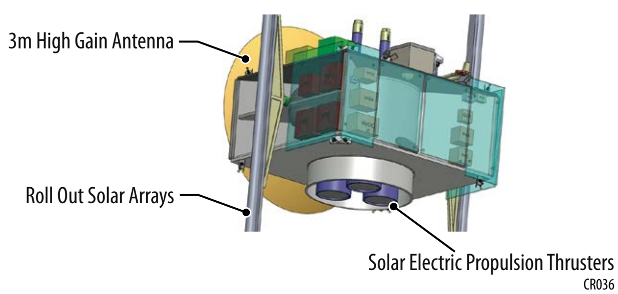


Figure 8-4: CROCODILE Bus

9.0 EARTH ENTRY VEHICLE

As illustrated in Table 9-1, the CROCODILE EEV is in family with the Stardust and OSIRIS-REx sample return capsules though slightly larger and heavier due to the need for additional thermal insulation around the sample Dewar. The maximum expected value ballistic coefficient of 126 Kg/m^2, double the ballistic coefficient of the other capsules, drives the needs for alternate heatshield and back shell materials.

Table 9-1: CROCODILE EEV as compared to past Sample Return Capsules

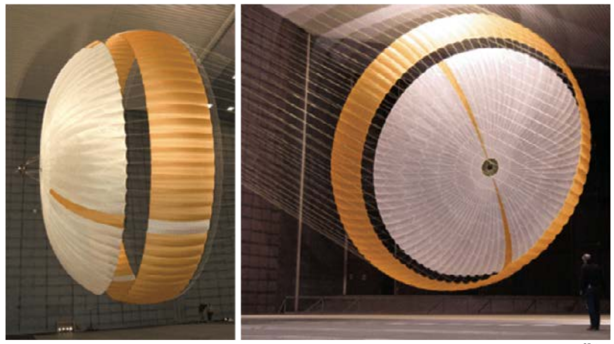
	CCSR (2010)[1]	OSIRIS REx[2,3]	Stardust[4,5,6]	CROCODILE
Shape, deg sphere-cone	60	60	60	60
Mass, kg	96 (MEV)	50.64	46	118 (CBE + 30% margin)*
Diameter, m	1.1	0.812	0.8	1.1
Ballistic Coefficient, kg/m^2	70	68	62.5	86.3
Parachute Deploy Conditions		Drogue at Mach 1.4, 30 km Main at 3 km	Drogue at Mach 1.4, 30 km Main at 3 km	Drogue at Mach 1.35, 30 km Main at 3 km
Probe Release, hr		EI - 4 hrs	EI - 4 hrs	EI – 4 hrs
Entry Planet Relative Velocity, km/s		12.2	12.8	12.2
Entry Planet Relative FPA, deg		-8.2	-8.2	-7.94
Entry Spin Rate, rpm		10.8	13.5	11
Duration (from EI), s	731.8	886	900	655
Landing Velocity, m/s	6.3	4.6	4.4	6.9
Max acceleration (during EDL)	31	31	34	28.7
TPS - Heatshield		PICA	PICA	3MDCP
TPS - Backshell		SLA-561V	SLA-561V	PICA

[1] Veverka, Johnson, Reynolds; Mission Concepts Study: Comet Surface Sample Return (CCSR) Mission
[2] Ajluni et al., "OSIRIS-REx, Returning the Asteroid Sample" 2015 IEEE Conference
[3] Lauretta, D.S., Balram-Knutson, S.S., Beshore, E. et al. OSIRIS-REx: Sample Return from Asteroid (101955) Bennu. Space Sci Rev 212, 925–984 (2017). https://doi.org/10.1007/s11214-017-0405-1
[4] Desai, Qualls, "Stardust Entry Reconstruction", JSR 2010
[5] Desai et al, "Entry, Descent, and Landing Operations Analysis for the Stardust Entry Capsule", JSR 2008
[6] Wilcockson, "Stardust Sample Return Capsule Design Experience", JSR 1999

The EEV has a two stage parachutes designed to traverse transonic and subsonic regime in stable configuration and then provide a large drag deceleration to land at appoximately 7 m/s. It has a 0.8 m diameter disk-gap-band drogue and a 7.3 m diameter triconical main parachute (Figure 9-1 and 9-2) in family with both Stardust and O-REx. A comparison of parachute designs is shown in Table 9-2.

Table 9-2: CROCODILE Parachute System as compared to heritage designs

	OSIRIS REx	Stardust	CROCODILE
Drogue Diameter, m (Disk Gap Band)	0.83	0.58	0.8
Main Para Diameter, m (Triconical)	7.3	7.3	7.3
Deploy Conditions	Drogue at Mach 1.4, 30 km Main at 3 km	Drogue at Mach 1.4, 30 km Main at 3 km	Drogue at Mach 1.35, 30 km Main at 3 km



CR048

Figure 9-1: Disk Gap Band Parachute

CR049

Figure 9-2: Triconical Parachute

10.0 CROCODILE COST BASIS OF ESTIMATE

Where payload is similar to previous instruments GSFC has built and flown analogous costs were adapted and adjusted for fiscal year. Percentage wraps based on GSFC's long history of planetary missions were applied to estimate management, systems engineering and safety and mission assurance costs. The GSFC CEMA office generated an estimate for the spacecraft bus (10.1), Sample Storage Dewar (10.2) and Cryogenic Storage Refrigerator (10.3) based on detailed Master Equipment Lists generated by the engineering team. The GSFC RAO Office estimated the Harpoon Sample Acquisition system. Mission Operations, Ground System, Integration, and Test estimate percentage wraps scaled from sample return missions and adjusted for the extended mission duration and additional complexity associated with handling a cryogenic sample. Table 10-1 summarizes the CROCODILE cost basis of estimate.

Table 10-1: CROCODILE Cost Basis of Estimate

Percentage Wraps	**Estimate Generated by GSFC CEMA Office**
Project Management	Sample Storage Dewar
Systems Engineering	Spacecraft Bus, including EEV
Safety and Mission Assurance	Cryogenic Storage Refrigerator
Science	
Analogous Instrument/Hardware Builds	**Wraps consistent with sample return missions and adjusted for additional complexity associated with a cryogenic sample**
PolyCAm	Mission Operations
MapCam	Ground systems
SamCam	Integration and Test
Thermal Emission Spectrometer	
IR Spectrometer	**Estimate Generated by GSFC RAO**
Lidar	Harpoon Sample Acquisition System
Backup Altimeter	
Sample Manipulation Arm	

CEMA estimates at the 70% confidence level are assumed to correspond to a point estimate with 50% reserve added.

10.1 CEMA Spacecraft Bus Cost Estimate Including EEV

CROCODILE Spacecraft Bus Ground Rules & Assumptions

- Estimate in FY25$
- Mission Risk Class B
- Contractor build
- Estimate covers cost from Phase A start through development and delivery to higher level I&T
  - » Phase A costs are rough estimates of maturing the following components from TRL 5 to TRL 6:
    - X-band LGA
    - Solar Array (ROSA)
- ETU, EDU, Flight Spares specified in MEL - parametrically estimated
- FPGA development cost - 3 FPGA assumed 100% New Development, 1 FPGA @ >30% New Dev, 4 FPGA @ >20% New Dev, and 1 FPGA @ >10% New Dev
- Environmental Testing includes testing prior to mission level environmental testing
- Using prices provided for X-/Ka-band HGA, Ka-band TWTA & EPC, and X-band TWTA & EPC
- Estimated the number of boards included in the following due to low fidelity information at this time:
  - » High Voltage Electronics Assembly (HVEA)
  - » Electric Propulsion Subsystem Power Processing Unit (PPU)
- Treated as Furnished Equipment (includes the I&T cost of the item to next higher assembly and associated Management, Systems Engineering and Safety & Mission Assurance costs):

» Laser Altimeter Subsystem

» S-band SLink-PHY Transceiver

» Patch Antenna

» Robotic Arm Subsystem

» Heatshield Thermal Protection System

» Parachute, Canister & Gas Generator Subsystem

» Backshell Thermal Protection System

» SEP Thrusters (aka Engines) and PPUs

- Hardware not included in estimate, per Customer:
  - » Lidar Subsystem
  - » Comet Sample Subsystem
  - » Camera Payloads (Long Range Imager, Sample Imager, Medium Range Imager, DVR)

CROCODILE Spacecraft Bus CEMA Parametric Cost Estimate (Development and Production Costs)	50% CL FY25 $M		70% CL FY25 $M	
Cost Model Summary 08-Jun-21				
Spacecraft Bus	**$492.59**		**$498.32**	
Mechanical Subsystem		*$66.02*		*$77.13*
Electrical Power Subsystem		*$102.40*		*$119.63*
Attitude Control Subsystem		*$20.97*		*$24.49*
Laser Altimeter Subsytem (See Note 1)		*$0.14*		*$0.17*
Avionics Subsystem		*$59.82*		*$69.88*
Communications Subsystem (See Note 2)		*$52.26*		*$61.05*
Electric Propulsion Subsystem (See Note 5)		*$44.68*		*$52.20*
Chemical Propulsion Subsystem		*$25.74*		*$30.07*
Thermal Control Subsystem		*$16.47*		*$19.24*
SC Harness		*$5.98*		*$6.98*
Robotic Arm Subsystem (See Note 3)		*$3.57*		*$4.17*
Earth Entry Vehicle (EEV) (See Note 4)		*$18.27*		*$21.35*
Management, Systems Engineering, Assembly, Integration & Test		*$76.26*		*$89.09*
Additional Costs	**$218.50**		**$260.39**	
Ground Support Equipment (GSE)		*$24.50*		*$28.62*
Environmental Test		*$24.50*		*$28.62*
FPGA Development (# of FPGAs specified in MEL)		*$9.14*		*$10.68*
Flight Software (FSW) Development		*$39.19*		*$45.79*
FSW Test Bed & Sim SW		*$1.96*		*$2.29*
Phase A (see GR&A)		*$119.22*		*$144.40*
CROCODILE Spacecraft Bus TOTAL	**$711.09**		**$758.71**	

Note 1: This includes the I&T cost for the Laser Altimeter Subsystem to the next higher assembly
and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T.
The hardware cost of the Laser Altimeter Subsystem is not included. It is treated as Furnished Equipment.

Note 2: This subsystem includes the I&T costs for the S-Band SLink-PHY Transceiver and Patch Antenna to the next higher assembly
and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T.
The hardware cost for theS-Band SLink-PHY Transceiver and Patch Antenna are not included. They are treated as Furnished Equipment.

Note 3: This includes the I&T cost for the Robotic Arm Subsystem to the next higher assembly
and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T.
The hardware cost of the Robotic Arm Subsystem is not included. It is treated as Furnished Equipment.

Note 4: This subsystem includes the I&T costs for the Heatshied TPS, Backshell TPS, Parachute and Parachute Canister & Gas System to the next higher assembly
and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T.
The hardware cost for the Heatshied TPS, Backshell TPS, Parachute and Parachute Canister & Gas System are not included. They are treated as Furnished Equipment.

Note 5: This subsystem includes the I&T costs for the SEP Thrusters (aka Engines) and PPUs to the next higher assembly
and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T.
The hardware cost for the SEP Thrusters (aka Engines) and PPUs are not included. They are treated as Furnished Equipment.

10.2 CEMA Sample Storage Dewar Cost Estimate

CROCODILE Cryogenic Sample Dewar Assembly CEMA Ground Rules & Assumptions

- Estimate in FY25$
- Mission Risk Class B
- Contractor Build

- Estimate covers cost from Phase A start through development and delivery to higher level I&T
  - » Phase A costs are rough estimates of maturing the following components from TRL 5 to TRL 6:
    - Cryogenic Sample Dewar
- ETU, EDU, Flight Spares specified in MEL - parametrically estimated
- No FSW or FPGA identified and no cost is provided
- Environmental Testing includes testing prior to mission level environmental testing
- Treated as Furnished Equipment (includes the I&T cost of the item to next higher assembly and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T activity):
  - » Sealing Mechanism

CROCODILE Cryogenic Sample Dewar Assembly CEMA Parametric Cost Estimate (Development and Production Costs)	50% CL FY25 $M		70% CL FY25 $M	
Cost Model Summary **04-Jun-21**				
Cryogenic Sample Dewar Assembly	**$0.90**		**$1.05**	
Cryogenic Sample Dewar		*$0.51*		*$0.60*
Pressure Sensor		*$0.06*		*$0.08*
Temperature Sensor		*$0.04*		*$0.05*
Sealing Mechanism (See note 1)		*$0.10*		*$0.11*
Management, Systems Engineering, Assembly, Integration & Test		*$0.18*		*$0.21*
Additional Costs	**$0.28**		**$0.32**	
Ground Support Equipment (GSE)		*$0.04*		*$0.05*
Environmental Test		*$0.04*		*$0.05*
Phase A (see GR&A)		*$0.19*		*$0.22*
CROCODILE Cryogenic Sample Dewar Assembly TOTAL	**$1.18**		**$1.37**	

Note 1: Includes the I&T cost for the Sealing Mechanism to the next higher assembly and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T activity.
The cost of the Sealing Mechanism Hardware is not included. It is treated as Furnished Equipment.

10.3 CEMA Cryogenic Storage Refrigerator Cost Estimate

CROCODILE Cryogenic Storage Refrigerator Ground Rules & Assumptions

- Estimate in FY25$
- Mission Risk Class B
- Contractor Build
- Estimate covers cost from Phase A start through development and delivery to higher level I&T
  - » Phase A costs are rough estimates of maturing the following components from TRL 5 to TRL 6:
    - Refrigerator Structure
    - Heat Exchanger and Lines
    - Solid Cryogen Tank
    - Cryo Latch Valves
- ETU, EDU, Flight Spares specified in MEL - parametrically estimated
- No FSW or FPGA identified and no cost is provided
- Environmental Testing includes testing prior to mission level environmental testing.
- Treated as Furnished Equipment (includes the I&T cost of the item to next higher assembly and associated

Management, Systems Engineering and Safety & Mission Assurance costs):
  - TIRS 2 TMU (Thermal Mechanical Unit)
  - TIRS 2 EBox
  - (Price of Argon Cryogen also not included)

- Estimated the number of boards included in the following due to low fidelity information at this time:
  - Redundancy Switch Electronics (RSE)

CROCODILE Cryogenic Storage Refrigerator CEMA Parametric Cost Estimate (Development and Production Costs)	50% CL		70% CL	
	FY25 $M		FY25 $M	
Cost Model Summary **04-Jun-21**				
Cryogenic Storage Refrigerator	**$39.13**		**$44.58**	
Mechanical (See Note 1)		*$27.53*		*$31.37*
Mechanisms		*$2.60*		*$2.97*
Electrical (See Note 2)		*$5.44*		*$6.20*
Management, Systems Engineering, Assembly, Integration & Test		*$3.55*		*$4.04*
Additional Costs	**$5.91**		**$6.89**	
Ground Support Equipment (GSE)		*$1.95*		*$2.22*
Environmental Test		*$1.95*		*$2.22*
Phase A (see GR&A)		*$2.02*		*$2.45*
CROCODILE Cryogenic Storage Refrigerator TOTAL	**$45.04**		**$51.47**	

Note 1: This subsystem includes the I&T cost for the TIRS 2 Thermal Mechanical Unit (TMU) to the next higher assembly and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T. The hardware cost for the TIRS 2 TMU is not included. It is treated as Furnished Equipment.

Note 2: This subsystem includes the I&T cost for the TIRS 2 Electronics Box to the next higher assembly and associated Management, Systems Engineering and Safety & Mission Assurance costs related to the I&T. The hardware cost for the TIRS 2 Electronics box is not included. It is treated as Furnished Equipment.

11.0 ACRONYMS

3MDCP	3D Mid-Density Carbon Phenolic
AAF	Army Airfield
ACS	Attitude Control System
AFB	Air Force Base
BS	Backshell
BOL	Beginning of Life
C&DH	Communication and Data Handling
CBE	Current Best Estimate
CM	Center of Mass
CROCODILE	Cryogenic Return Of Cometary Organics, Dust, and Ice for Laboratory Exploration
DRM	Design reference Mission
EEV	Earth Entry Vehicle
ESA	European Space Agency
NF	New Frontiers
CCNSR	Cryogenic Comet Nucleus Sample Return
CSSR	Comet Surface Sample Return
CML	Concept Maturity Level
DDOR	Delta-Differential One-way Ranging
DSN	Deep Space Network
DTM	Digital Terrain Map
EDL	Entry, Descent and Landing
EGA	Earth Gravity Assist
EMC	Elastic Memory Composites
EMTG	Evolutionary Mission Trajectory Generator
EOL	End of Life
EOM	End of Mission
EPC	Electronic Power Conditioner
ESA	European Space Agency
FDC	Fault Detection and Correction
FPA	Flight Path Angle
GD	General Dyamics
Gb	gigabit
GNC	Guidance, Navigation and Control
HEEET	Heatshield for Extreme Entry Environments
HGA	High Gain Antenna
HVDC	High Voltage Down-Converter
HVEA	High Voltage Electronics Assembly
HVRA	High Voltage Relay Assembly
IL	Insulation Layer
IPDR	Instrument Preliminary Design Review
ICDR	Instrument Critical Design Review
JFC	Jupiter Family Comet

kbps	kilobits per second
KSC	Kennedy Space Center
LGA	Low Gain Antenna
LRD	Launch Readiness Date
MCDR	Mission Critical Design Review
MGS	Medium Gain Antenna
MEL	Master Equipment List
MEV	Maximum Expected Value
MLI	Multi Layer Insulation
MPDR	Mission Preliminary Design Review
MSL	Mars Science Lab
MST	Mountain Standard Time
NFOV	Narrow Field of View
NIMBUS	Numerical Icy Minor Body evolUtion Simulator
OCAM	OSIRIS-REx Camera Suite
OD	Orbit Determination
OTES	OSIRIS-REx Thermal Emission Spectrometer
OVIRS	OSIRIS-REx Visible and InfraRed Spectrometer
PDU	Power Distribution Unit
PI	Principal Investigator
PICA	Phenolic Impregnated Carbon
PME	Propulsion Mechanism Electronics
PSR	Pre Ship Review
PSE	Power Supply Electronics
ROI	Regions of Interest
ROSA	Roll Out Solar Array
SAG	Shoot and Go
SDST	Small Deep Space Transponder
SEP	Solar Electric Propulsion
SOI	Sphere of Influence
SPC	Stereophotoclinometry
SPENVIS	Space Environment, Effects, and Education System
STK	Systems Tool Kit
TD	Touchdown
TRL	Technology Readiness Level
TRN	Terrain Relative Navigation
TPS	Thermal Protection System
TWT	Traveling Wave Tube
UHF	Ultrahigh Frequency
UTTR	Utah Test and Training Range
V&V	Visions and Voyages
WEM	Warm Electronics Module
WFOV	Wide Field of View

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