PERCI

Penetrator for Environmental, Regolith, and Crater Investigation

Uranus Orbiter and Probe Secondary Payload

Final Report

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1 Executive Summary

In 2022, the 2023-2032 Decadal Survey named a mission to Uranus as the top priority for NASA's next decade of planetary exploration. The proposed Uranus Orbiter and Probe (UOP) mission would tour the Uranian system and return science on the formation of the Uranian system. The task assigned by our customer (JPL), was to develop a secondary payload for the UOP mission. This payload aims to enhance the science return of the UOP mission through a penetrator that would impact one of the main Uranaian moons. A penetrator allows for in-situ measurements that can strengthen the UOP's science return.

This paper discusses the engineering analysis of the penetrator mission, identifies the key systems engineering topics, delves into the science goals and Science Traceability Matrix, and gives a summary of the penetrator design and subsystem details. Moreover, the schedule and integrated management system (IMS) are mentioned towards the end of this document. The document encompasses the work completed over a 9 month period.

2 Introduction

The problem statement presented to our team from The University of Michigan and the Jet Propulsion Laboratory (JPL) is as follows: *Develop a smallsat mission concept that can best leverage the proposed Uranus flagship mission to provide a greater science return targeted at a moon in the Uranus system.* To begin, our team conducted a thorough review of the UOP mission to identify scientific goals that could be supplemented by an accompanying smallsat mission.
The UOP mission is a top priority mission in the 2023-2032 Decadal Survey submitted by the Planetary Society. Led by NASA JPL, the mission is currently baselined for a 2031 launch, followed by a 13.7-year cruise to the Uranian system. Upon arrival, the UOP will perform a detailed investigation of Uranus, including its atmosphere, magnetosphere, ring system, and moons. The science objectives can be broadly grouped into four key areas, (1) Understanding Solar Stem Origins, (2) Studying Formation Processes, (3) Exploring Habitability, and (4) Informing interconnections of the Uranian atmosphere.

The first point deals with the formation of Uranus itself, any potential migration it had in the history of the Solar System, and if a giant impact could explain its peculiar orientation. The second point deals with how heat and energy are exchanged between the gas giant and its major satellites, as well as any other interaction between these bodies, external forces influencing the system, and potential causes of Uranus' magnetic field. The third point deals with the major satellites of Uranus, and whether they have or had oceans in its past. The last area deals with drawing connections between Uranus and exoplanets beyond our solar system, as well as how the solar wind interacts with a complex and dynamic magnetosphere. PERCI aims to help the UOP mission answer questions related to habitability in Uranus' major satellites.

The UOP flagship proposal outlines a trajectory that includes multiple ballistic flybys of four of the five major moons: Ariel, Oberon, Titania, and Umbriel. However, these flybys only allow for brief remote observations of the moons' spectral composition and topography. In-situ measurements of surface chemistry and thermal properties remain out of reach using just the flagship. To address this gap, we propose PERCI (Penetrator for Environmental, Regolith, and

Crater Investigation), a smallsat mission designed to deliver a penetrator directly to the surface of a Uranian moon. This mission presents a unique opportunity to conduct in-situ measurements directly on a moon's surface and greatly enhance the scientific findings of the proposed mission. This mission concept was developed in close collaboration with mentors from the University of Michigan and NASA JPL, and is aligned with both the 2023-2032 Decadal Survey and the goals of NASA's Small Innovative Missions for Planetary Exploration (SIMPLEx) program.

2.1 Scope

The goal of this project is to develop a spacecraft concept that will fulfill the primary objective of our customer: Enhancing the science return for the Uranus Orbiter Probe missions. This will include identifying requirements, constraints, and developing a system architecture. Work on this project includes:

- Identification of the top-level requirements and constraints
- Selection of Mission Concepts that enhance the science return of the UOP
- Development of a concept of operations to satisfy customer objectives
- Work breakdown into subsystems
- Trade Studies for Instrumentation and payload
- Risk Analysis and Mitigation Strategies

3 Systems Engineering

The development of this mission concept follows a systems engineering framework, incorporating axiomatic design principles to ensure that design choices remain aligned with mission-level objectives.

3.1 System Requirements

This subsection defines the key requirements, constraints, and drivers that inform the system architecture. Derived from stakeholder input and mission objectives, these requirements establish the foundation for the PERCI design and analysis.

3.1.1 Customer Needs

NASA JPL, the primary customer, was consulted to identify key customer needs for the secondary payload. A complete requirements table can be found in Appendix A.1. The following is a summary of the customer needs directly associated with the UOP mission:

- The system shall fit within the payload fairing of the launch vehicle with [TBR] margin.
- The system shall enhance the scientific objectives of the UOP mission
- The system shall use COTS and TRL 6+ technology.
- The system shall adhere to guidelines listed in NASA's SIMPLEx program.
- The system shall not exceed a power draw of 2W total power from the UOP flagship during transit.
- The system shall not impede performance of the UOP mission.
- The system shall do no harm to the UOP flagship.

- The system shall communicate via Ultra-High Frequency (UHF) to the UOP flagship at a maximum data rate of 20.1 kbps.

3.1.2 Top Level Requirements

The top-level requirements define the broad objectives and key technical capabilities needed to meet the payloads's most critical challenges. These reflect PERCI's integration into a broader NASA JPL proposal within the UOP mission context. A complete requirements table can be found in Appendix A.2. A summary of these requirements is provided below:

- The system shall support the current NASA planetary science goals.
- The system shall adhere to the NASA planetary protection guidelines for forward contamination.
- The system shall not impede the UOP orbiter's trajectory during separation from the orbiter.
- The system shall not have greater than [TBR] [W or W/m^2] of heat flow to/from the flagship.
- The system shall maintain all components within their temperature limits with ±5°C margin.

3.1.3 Constraints

Most of the system constraints are derived from NASA's SIMPLEx 2017 guidelines, originally established to guide requirements for low-cost planetary SmallSat missions. Although these requirements are tailored towards Mars and Lunar centric missions, their application was deemed appropriate by the customer due to the analogous nature of this small, cost-constrained planetary mission. While some constraints are mandatory ("shall"), others are preliminary targets ("should") that may be exceeded if justified by system performance or integration needs. A complete requirements table can be found in Appendix A.3. A summary of the constraints is provided below:

- The system shall meet each of the critical development milestones as per NASA SIMPLEx guidelines.
- The system shall not exceed a dry mass of 180 kg.
- The system should not exceed a total volume of 0.46 m^3 without an ESPA-grande or propulsive ESPA ring, and should not exceed a total volume of 1.19 m^{3}with an ESPA-grande or propulsive ESPA ring.
- The system should not exceed a cost of $55 million for all mission phases.

3.1.4 Critical Requirements

The following requirements drive schedule, cost, and budgets, derived from the customer needs. A complete requirements table can be found in Appendix A.4. A summary of the critical requirements is provided below:

- The system shall meet each of the critical development milestones as per the NASA SIMPLEx guidelines.
- The system shall have sufficient power during all mission phases.

- The system shall survive the launch environment.
- The payload shall remain functional during relevant mission phases.
- The system shall survive the transit phase.
- The system shall survive all deployment phases.
- The system shall survive [4.5 (CBE)] hours post-deployment.

3.1.5 System Drivers

The following requirements drive schedule, cost, and budgets, but are not directly tied to a customer need. A complete requirements table can be found in Appendix A.5. A summary of the system drivers is provided below:

- The system shall have a pointing accuracy of 1°.
- The system shall have 450 m/s delta-V capability.
- The system shall be able to have [TBR] for data handling storage
- The system shall be able to target the Uranian moons of Umbriel and Titania.
- The system shall be radiation tolerant to survive in all mission radiation environments.

For the penetrator concept, post-impact survival is *optional* and may be descoped based on project constraints. A summary of the optional system drivers is provided below:

- The system should survive 30 minutes post-impact.
- The system should be able to reduce to a maximum impact velocity of [TBR].

3.1.6 Functional Requirements

The functional requirements for this mission were developed using a Science Traceability Matrix (STM) guided by science priorities outlined in the Decadal Survey. As shown in Appendix A.7, each functional requirement is derived by mapping high-level Decadal Survey questions to specific science goals and then identifying the necessary system capabilities to achieve those goals. A complete requirements table can be found in Appendix A.6. A summary of the functional requires are provided below:

Core "Shall" Requirements:

- The system shall support continuous operation to all instruments, as well as the CDH and TT&C subsystems during its descent.
- The system shall maintain continuous line-of-sight access with the orbiter throughout descent and impact.
- The system shall have a pointing stability of [TBR]°/sec during descent.
- The system shall shield the imagers from dust and other hazards during approach during descent.
- The system shall be able to reduce to a maximum impact velocity of [TBR].
- The system shall create a crater with radius >3 meters.
- The system shall capture, process, and transmit scientific imagery during descent.
- The system shall maintain a high signal-to-noise ratio across all onboard instruments.
- The system shall measure and transmit [TBR (# of data types x 1 matrix)] bits of data of scientific temperature measurement data for each data type.

Post-Impact "Should" requirements:

The following requirements represent desirable post-impact capabilities. These all would enhance the science return but are subject to descoping depending on mass, power, and survivability trade-offs.

- The system should survive the impact, with all necessary post-impact components operational.
- The system should record acceleration data to determine the deceleration profile at the time of impact.
- The system should remain operational in a low temperature moon surface embedded environment at 30K.
- The system should conduct high-frequency temperature measurements following impact.
- The system should reach a depth of >[TBR] meters below the surface.

3.2 Verification Matrix

Each top-level system requirement was paired with a verification method to ensure traceability and inform the performance of the system against mission objectives. The matrix defined in Appendix A.8 includes the planned verification approaches depending on the nature and criticality of the requirement. While all requirements are currently unverified, the matrix outlines verification campaigns for future payload development phases.

3.3 Work Breakdown Structure

The overall system is divided into four primary work tasks: System, Concept, Science, and Spacecraft. There are seven subsystems (Payload & Instrumentation, Power, Propulsion, CD&H, ADCS, Structures, and Thermal. The WBS flowdown in Figure 1 highlights the organization of responsibilities among the team providing an outline for scheduling, integration, and critical path to mature the concept.

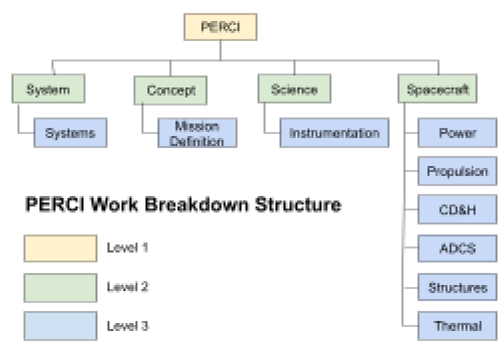

Figure 1. PERCI Work Breakdown Structure

3.4 Budgets

Budget Item	Program Limit	Current Estimate	Margin and Notes
Cost	< $55 M (all mission phases)	TBR	Instrument list & structures trades just closed; cost carries the highest risk (RPN = 80)
Mass	180 kg wet	159 kg	21 kg (12 %) margin. Mass drivers are structures (57 %) and propulsion (25 %)

Volume	0.42 m^3 (1.19 m^3 with ESPA ring)	0.072 m^3 (71,690 cm^3)	>80 % unused; easily nests inside the UOP carrier.
Average Cruise Power	2 W maximum draw from orbiter during cruise	0 – 1 W (survival heaters only)	Survival heaters need < 1 W thanks to passive insulation
Science-Mode Power	Internal batteries only	37 W avg / 140 W pulse	Peak heater load occurs post-separation; does not affect the 2 W cruise limit
RF Link Margin (S-band)	Secure data transmission	24 dB	Allows for -104dBm received power on orbiter, at 10° mismatch, -118dBm at 45°
Data	Sufficient onboard memory	50 GB	Need to be able to handle at least 20 GB. Allows for 2.14mbps data rate using maximum margin (~1dB). 10kbps at 15dB margin

Table 1: Budget Overview

Note on Power Budget: 37 W is the average system drawn during the post-separation science sequence. 140 W is the momentary heater peak during that same sequence. Because the orbiter will not supply power after separation, those peaks are covered by the onboard primary-cell battery stack; the cruise-phase limit of 2 W from the orbiter is respected because the survival heater requirement is < 1 W while docked.

3.5 Risk Analysis

Risk management for PERCI was approached with a top-down systems engineering mindset, emphasizing proactive identification, evaluation, and mitigation of high-impact threats to mission success. With a mission profile centered around a penetrator spacecraft designed for impact-based science collection on a Uranian moon, the inherent technical complexity and operational uncertainty needed a robust and dynamic risk management.

Risks were categorized by their nature (technical, cost, or schedule) and prioritized using a Failure Modes and Effects Analysis (FMEA) framework. Each risk was scored across three dimensions: severity, likelihood, and detectability, producing a Risk Priority Number (RPN) used to triage issues and allocate mitigation resources. Fifty-six total risks were identified with fourteen top risks being most likely to compromise mission success.

The highest-ranked risks focused on operational uncertainty, particularly cost overruns and scheduling delays, due to ambiguous mission requirements and external dependencies. The top three risks and their management strategies highlight the overarching approach:

CST-03: Cost Overruns (RPN: 80 → 45)

Ambiguities in early cost modeling posed a threat to the financial viability of the mission. Mitigation included detailed bottom-up cost estimation, refined historical cost modeling, and active tracking of high-variance items. A backup plan involving re-baselining and de-scoping was prepared in case mitigation efforts fall short.

SHD-01: Vendor Delivery Delays (RPN: 75 → 30)

Given the reliance on external vendors for key components, late deliveries could disrupt the mission timeline. The team implemented early and frequent vendor engagement, structured schedule reviews, and early-warning mechanisms to detect potential delays.

SCI-06: Instrument Activation Failure (RPN: 60 → 30)

This technical risk threatened the mission's scientific return. Mitigation involved early-stage integration testing using engineering models that replicate thermal, vibrational, and electrical environments to catch issues well before final assembly.

In addition to early detection, layered mitigation, and flexible contingency planning, the team carried out (thermal and structural) simulations and modeling whenever able, incorporated redundancy into key subsystems, and updated risk matrices as the design matured. To understand the acronyms and see the spread of the other risks refer to the original and updated risk matrices in section C.3 in the appendix.

4 Mission Concept Selection

4.1 Penetrator Overview

As mentioned earlier, the mission is to send a penetrator spacecraft as the secondary payload to the UOP mission. A penetrator is designed to survive impact and take surface measurements which would be transmitted to the main orbiter (UOP). To meet the goals of this mission, the penetrator would include the following instruments and components: batteries, reaction wheels, seismometer, camera, RTDs, accelerometer, SRBs, antenna, microcontroller, etc.

4.2 Alternative Concepts

Two alternative concepts were considered for this project: an orbiter and an impactor. The overviews for both, as well as their associated trade studies, can be found in Appendix C1.

4.3 CONOPS

The mission is divided into 4 phases: deployment, approach, descent, and penetration. It begins with the deployment phase (T-4.6 hrs), during which the spacecraft performs a flyby of Umbriel at a controlled velocity, and the target location is identified. Next is the approach phase (T-4.4 hrs), where the spacecraft executes attitude control (ADCS) for stabilization, performs a delta-V burn for trajectory adjustment, and jettisons the solid rocket boosters (SRBs) while executing a 180-degree turn. In the descent phase (T-1 hr), ADCS continues to stabilize the probe, imagers and an accelerometer become active, and image data is compressed and transmitted. Finally, at penetration (T-0), the probe activates its thermal probe and seismometer for scientific data collection, and continues data compression and transmission back to the spacecraft or Earth.

5 Science

5.1 Overview

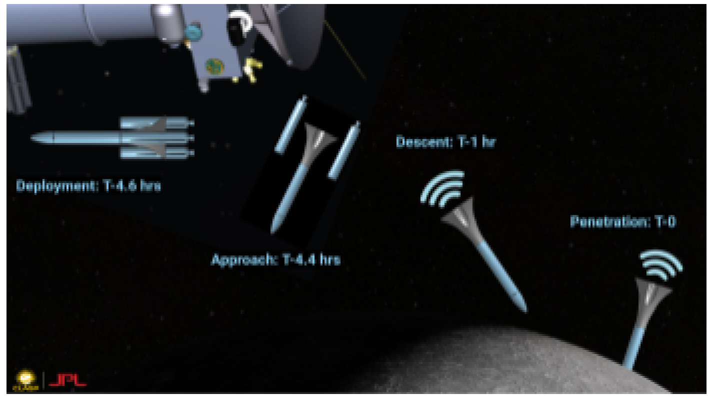

Figure 2: PERCI concept of operations

The science goals and objectives of the PERCI mission are the main drivers of the mission as a whole; its design, instrumentation, and concept was chosen based on the underlying objective of serving to enhance the science return of the UOP mission. While the UOP orbiter will be able to provide in-depth remote sensing measurements, its design and concept inherently prevents it from conducting direct surface/subsurface measurements. This gap in science return is exactly what PERCI will fill as a secondary payload to the UOP. As a penetrator to one of the Uranian moons, PERCI will allow for direct in-situ measurements on the moon that cannot be done by an orbiting spacecraft, and are defined below in the STM summary. These capabilities allow PERCI to contribute unique and complementary measurements that will help to significantly expand the scientific return of the UOP mission. As is also described in more detail below in the STM summary, PERCI takes a science-first approach in order to be a valuable secondary payload to NASA's flagship UOP mission that will enhance its science return in ways not possible by the UOP alone.

5.2 Science Traceability Matrix

With the main goal of PERCI being to enhance the science return of the UOP mission, Decadal Survey questions were identified that were not intended to be answered by the UOP mission but could be answered by a supplementary mission that can take on more risk. The following five questions related to Uranian satellites were identified:

- Q12.8a What Can Observations of Material Reveal About Formation of Circumplanetary Systems?
- Q4.4b What Materials Ejected from Impact Craters Are Deposited on Planetary Surfaces?
- Q5.1c How Does Porosity, Ices, Liquids, or Gases Affect the Physical Properties of the Crust?
- Q5.1a How Much Variability in Composition and Internal Structure Is There Within and Between Solid Bodies, and How Did Such Variability Arise and Evolve?
- Q8.2f How Are Heat and Material Transported Through—and Ultimately Out of—Satellite Interiors?

From these questions, a list of science goals and objectives were developed: collect images of the satellite's surface to determine the surface topography (Q12.8a); determine the chemical composition of the satellite's surface by identifying and quantifying the relative abundance of key chemical elements and minerals (Q4.4b); determine the physical properties of the satellite's surface at the point of impact by determining the surface hardness (Q5.1c); probe the interior of the satellite to determine the interior composition, structure, and ice sheet thickness (Q5.1a); and determine the thermal properties of the satellite's surface by determining the thermal conductivity and heat dissipation rates as well as their variation at different depths (Q8.2f).

To accomplish these goals and objectives, measurements and instrumentation were identified. First, images of the satellite's surface would be taken in nadir and azimuth via a downward-facing multi-spectral (R, G, B, NIR-1, NIR-2) imager and a side-view visible spectrum imager (Q12.8a). Next, a visible and near-infrared imaging spectrometer would be used for composition identification (for water, carbon, and silicates) of the plume generated by the impact of PERCI into the satellite's surface (Q4.4b). A deceleration profile will be generated during impact via an accelerometer (Q5.1c), and seismic noise (including coda of body waves and the harmonic frequency of Crary Waves) will be recorded via a seismometer (Q5.1c). Lastly, post-impact temperature decay will be recorded using a Resistive Temperature Detector (RTD) (Q8.2f).

Functional requirements were derived from the measurement principles and goals of the STM, as discussed in the Functional Requirements section above. Additionally, see Appendix A.5 for a diagram of the STM flowdown.

5.3 Instrument summary

In order to capture imaging data during PERCI's descent towards one of the Uranian moons, the OSIRIS REx SamCam/TAGCAMS was selected. This camera system was flown on NASA's OSIRIS-REx mission, demonstrating heritage. The system also has exceptional low-light sensitivity and a design that supports integration with attitude control systems. It will be utilized for descent phase imaging, helping PERCI accomplish its science objectives of collecting images of the moon's surface and determining the moon's surface topography. Detailed specifications are given

In order to determine the surface composition of one of the Uranian moons as well as to analyze impact eject, PERCI will utilize the Lucy L'Ralph Vis/NIR imaging spectrometer already aboard the flagship UOP Orbiter. The imaging spectrometer will allow for the detection of materials in both the regolith and the plume created by impact, allowing the mission to accomplish its scientific objective of identifying and quantifying relative abundances of chemical elements and minerals present on the moon's surface.

The third instrument chosen to be used in the PERCI mission is the PCB Piezotronics accelerometer (Model 350B41) in order to capture and transmit the deceleration profile during the mission's impact phase, helping to characterize the moon's surface hardness. This accelerometer was selected due to its high shock tolerance and precise sensitivity, ensuring that accurate measurements are recorded in order to meet the science objectives of determining the surface hardness and deceleration profile

As a means of investigating and probing the interior structure of the Uranian moon, the Artius Seismometer will be used aboard PERCI. The Artius Seismometer is a broadband seismometer able to detect a wide range of frequencies while maintaining a sufficient tilt tolerance. Equipping PERCI with the Artius Seismometer will help accomplish the science objectives of determining the interior composition and structure of the Uranian moon as well as detecting seismic noise post impact.

The last instrument that will be placed aboard PERCI is the Cernox Resistance Temperature Detector by Lake Shore Cryotronics. This thermal detector is operable in extreme cryogenic conditions white demonstrating precise and reproducible results as well as radiation tolerance. This instrument is critical in allowing PERCI to accomplish its science objectives of determining the thermal conductivity and heat dissipation rates on the surface as well as post-impact temperature decay.

Trade studies were conducted in order to select these instruments, and these trades can be found in Appendix C.2. Chosen instruments' specifications are given in Appendix B.1.

6 Spacecraft System Design

6.1 Payload

6.1.1 Overview

As mentioned previously, the instrument suite inside the PERCI spacecraft bus, shown in figure 3, includes an accelerometer, resistive temperature detector (RTD), camera, and seismometer. The placement of these instruments directly correlates to the PERCI's ability to collect quality data during the descent and post-impact phases. Important factors to consider include:

- Level of sensitivity of the instruments to the outside medium
- Field of view
- Level of thermal isolation from the outside medium
- Post-impact survival

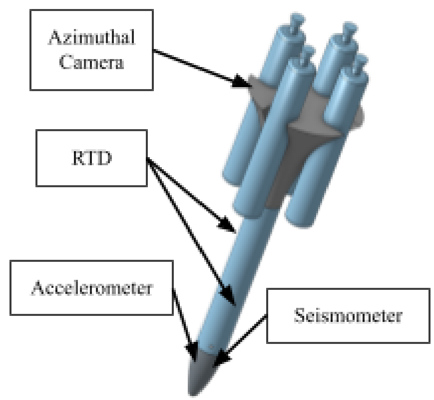

Figure 3: Instrument Suite Placement

Starting with the accelerometer, sensitivity to the surrounding environment shall be maximized physically, thermal isolation shall be maximized, and should survive post-impact. Field of view is not a contributing factor. Thus, the best placement for the accelerometer would be in the nose cone, closest to the point of impact.

Moving on to the RTD, the measurements shall be sensitive to outside temperature decays, be kept within its operating temperature range, and shall survive post-impact. Field of view is again not a contributing factor. Thus, similar to the accelerometer, the temperature probe should be kept within the forebody section, but in the main body section. In order to properly model the temperature variation with depth, multiple RTDs can be placed within the main body section. Additionally, multiple sensors adds redundancy in case one fails.

For the camera, sensitivity to the surrounding environment shall be minimized thermally, field of view shall be maximized, thermal isolation should be maximized, and the camera does not need to survive post-impact. In addition, since the science objective driving the camera selection requires azimuthal and nadir pointing imagery, two cameras will be required. To maximize field of view, the azimuthal camera shall be placed on the aft flare of the penetrator, with the nadir camera placed on the main body at the bulkhead connecting the main body of the nose cone.

Lastly, for the seismometer, sensitivity to the surrounding environment shall be maximized physically, thermal isolation shall be maximized, and the instrument should remain operational post-impact. Field of view is not a contributing factor. Thus, the seismometer shall be placed in the nose cone with the accelerometer. The organization of the seismometer with the accelerometer in the nose cone is pushed to later stages of development. Figure 3 is a visual representation of the instrument placement on PERCI.

6.1.2 PERCI CONOPS

The concept of operations for the instrument suite can be broken down into two distinct phases: the descent and post-impact phases. During descent towards the surface, the cameras will be actively taking images, and the accelerometer will be activated in preparation for the impact. Lastly, after the impact occurs and PERCI is embedded in the surface, the seismometer and RTDs will be activated, and will be operational until batteries fully discharge. Figure 2 in section 5.1 is a visual representation of the payload CONOPS.

6.2 TTC

Communications will be facilitated by a microstrip patch antenna integral to the penetrator structure, operating at 2GHz. Two orthogonally oriented feedlines offset by a quarter wavelength allow for both transmission and reception with circular polarization, providing increased communications resiliency and reducing dependence on relative orientation between transmitter and receiver. This frequency was chosen as it maximizes available space on the rear of the penetrator, while also avoiding most attenuation from the combination ice and rock surface of Uranian satellites should the probe be buried. Depending on the satellite in question and temperature, RF attenuation at 2GHz of the surface will be between 0.05 and 0.5dB/m. A link budget was performed on the system, assuming a medium gain antenna present on the orbiter craft at flyby range (~6500km), which provides a communications margin of 15dB when transmitting at 10kbps, and 1dB if the link usage is maximized at 2.14mbps. The TTC system will be transmitting imagery and ADCS information during its transit down to the surface while it has a higher data rate available in free space, and will dynamically adjust its data rate lower as link quality reduces. Post-impact, the remaining 10kbps will be sufficient to relay all temperature, seismometer, and latent accelerometer data.

6.3 CDH

The CDH system must withstand a total ionizing radiation dose of at least 25 krad. The Microchip ATMegaS128 is a space rated, radiation hardened 8-bit microcontroller capable of withstanding 30 krad. Due to their low cost, complexity, and size, three of these microcontrollers can run simultaneously in a triple-modular redundant configuration. These microcontrollers will have authority over all physical systems including ADCS, navigation, propulsion, and craft detachment, and will have all conops-related commands stored on board as a failsafe should communication be lost with the orbiter.

Furthermore, this system will collect and process all data from scientific instruments both on the descent to the target moon, as well as operational data obtained from the surface post-impact. The TT&C system will work in conjunction with the C&DH system to send all available data via wireless link to the orbiter while the orbiter is within fly-by distance.

6.5 ADCS

The ADCS system was sized to meet pointing requirements during the deployment de-tumble, SRB burn, the impact phase, and the 180° SRB slew (detailed in the Propulsion section). To ensure even thrust distribution from the four SRBs and prevent net torque, an initial spin will be induced using tiny solid rocket motors housed in the spacecraft's main body in order to leverage the gyroscopic effect. A de-spin maneuver will follow utilizing a Yo-Yo desaturation mechanism, where two masses spin in the opposite direction to reduce angular momentum. The ADCS is primarily sized for the largest maneuver, the critical 180° spin after SRB shutdown, which affects impact pointing accuracy. A trade study led to the selection of reaction wheels, specifically the Rocket Lab 60mNms model, which will provide sufficient margin to avoid saturation before impact. To maintain accurate pointing, the spacecraft will employ two star trackers, a 3-axis IMU, and an accelerometer to deliver real-time attitude updates throughout the mission. This configuration system of reaction wheels and sensors will allow the spacecraft to maintain the 1° pointing requirements to the orbiter and moon surface (when required) during all phases of the mission.

6.6 Structures

The penetrator consists of three sections: a nose cone, a cylindrical body, and a flared aft section. Figure 3 below shows the configuration. The nose cone is a blunt ogive shape, a shape which is intended to prevent deflection, and reduce deceleration forces imparted to the body of the penetrator during impact. The body of the penetrator is a hollow cylinder, and houses the crafts batteries and other components. Lastly, the flared base, onto which the antenna and solid rocket boosters are mounted, is intended to prevent over penetration of the surface to ensure a stable connection can be made between the penetrator's antenna and the spacecraft. The material selection has yet to be fully determined, but titanium is the primary candidate material for the nose cone due to its high strength per unit mass, while steel is the primary candidate for the body and flare sections for reasons of strength and cost.

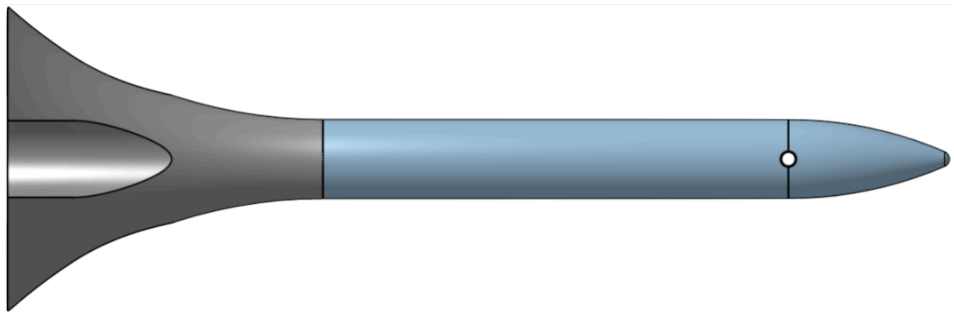

Figure 4: A CAD rendering of the penetrator structure.

A primary design constraint for PERCI's structure is to ensure the operational capability of all necessary components during all phases of the mission. This becomes particularly important during the penetration phase. Multiple analyses were conducted to estimate the impact force on the nose cone during penetration. Details on this analysis can be found in Appendix B.5.3. The solution highlights show the nose cone will experience an impact force between 6.5 and 16 MN, and penetrate between 2.7 and 3.5 meters into the surface.

6.7 Thermal

The thermal subsystem was designed to meet the requirements listed in Table 25 in Appendix B.6.1. Due to concerns of surviving and operating in cold thermal environments, the thermal subsystem was identified as a high risk area for the project early. To address concerns that a thermal control solution would not be possible within the budget, thermal analysis was conducted to determine if a solution was possible to meet the project's thermal subsystem requirements. Additionally, the analyzes would be helpful for determining necessary power allocations. 1-node and 70-node Thermal Desktop models representing a general small spacecraft and the penetrator were created to demonstrate a possible thermal management system. Details on all analyzes completed and thermal management system recommendations can be found in Appendices B.6.2 and B.6.3. A viable solution was found that met all requirements without needing expensive and complex thermal control solutions. The solution's highlights are as follows: uses less than 1W of survival heater power for transit and safe modes, uses 140W of heater power for operating in Uranian moon ice, and transfers less than 1W of heat to/from the UOP orbiter.

6.8 Power

Electrical power for the penetrator will be provided via 48 batteries of four, impact resistant, space-qualified lithium thionyl chloride (LTC) primary cells, shown in Figure 5 below. These cells will build off of the impact resistant design employed by JPL for the Deep Space 2 mission. As each cell ranges from 3.5V at maximum charge to 2.75 at minimum, this configuration will result in a 12V system. A power budget was calculated to determine the power draw during both transit as well as post-impact, including total energy needed for the entire system. The most significant power draws come from the science instruments, the CDH subsystem, the TTC subsystem, and the heating coils. The heating coils, which combat the low temperature of the impacted moon, by far consume the most power post impact at 140W. The 48 parallel configuration was included to spread the high current load of the heater coils across enough batteries to avoid overcurrent events in any one set of cells.

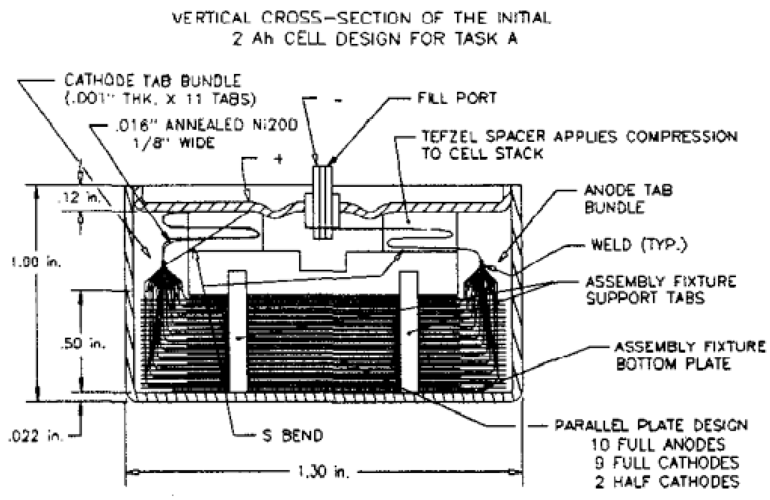

Figure 5: JPL/Yardney high impact LTC cell design

6.9 Trajectory

In order to approximate the flagships trajectory around the moon a physics based simulation was created accounting for the gravitational forces from Uranus and the moon as well as the fly-by altitude and velocity of the flagship. A table of fly-bys are provided in the UOP concept study displayed in figure 14 in Appendix B.8. An optimal fly-by was selected for each moon to minimize impact velocity. This is done by choosing the lowest fly-by velocity. The trajectory simulation takes deployment location in the form of time before closest approach and delta V in the form of velocity at deployment as inputs. It then provides flight duration and impact velocity and angle.

Trajectories for each moon are displayed in table 2. Each trajectory is optimized to minimize impact velocity and allow for an impact angle close to 90 degrees. Ultimately Umbriel was chosen for this mission as it has the lowest impact velocity. Additionally, it will be in line of sight of the flagship after impact as the trajectory flight duration is less than the time before closest approach. The DeltaV determined from this simulation is a driving requirement for the propulsion subsystem.

Moon	Flyby	Time Before Closest Approach [hrs]	ΔV [m/s]	Impact Velocity [m/s]	Impact Angle [deg]	Flight Duration
Ariel	7/29/47	3.0	479	1170	89	2.8
Oberon	7/28/47	4.7	414	1397	84	6.1
Titania	8/11/45	2.5	820	2258	85	3.3
Umbriel	6/22/47	4.4	415	901	86	4.2

Table 2: Simulation results for each moon

6.10 Propulsion

To induce the impact trajectory described from the previous section, a DeltaV burn of approximately 450 m/s is required. To perform this, a trade study was conducted analyzing the different modes of propulsion such as cold gas, hydrazine, liquid chemical, electric, and solid fuel. Based on the results, it was concluded that solid rocket fuel is the best option as it is low cost/mass/volume, easy to start up, empty weight can be shed from spacecraft, and it satisfies the do no harm requirements to the main orbiter. Once the fuel type was selected, we were able to source a specific motor from Orbital ATK/Northrop Grumman which met all of the size, mass, cost requirements. The Star 5D is a solid fuel rocket motor which flew on the Mars Pathfinder mission through NASA in 1997 and is the selected motor for this mission given the heritage and reliability. With a configuration of four motors centered around the aft flare base, the spacecraft's total DeltaV burn achieves 476m/s, satisfying the trajectory requirements for an impact trajectory. Furthermore, to ensure there is no unnecessary damage to the main body or tip of the spacecraft, the SRBs will be mounted with the nozzles facing aft. To then accommodate the 180 slew maneuver for a tip leading trajectory, the ejection of the SRBs will be used and staggered in a way to automatically induce the flip with minimal ADCS input.

7 Schedule

Table 3 below shows the Gantt chart developed and used throughout the 9 months used to develop and mature the PERCI concept. As it can be seen, the only thing left to do is write the scientific paper detailing all the analysis done.

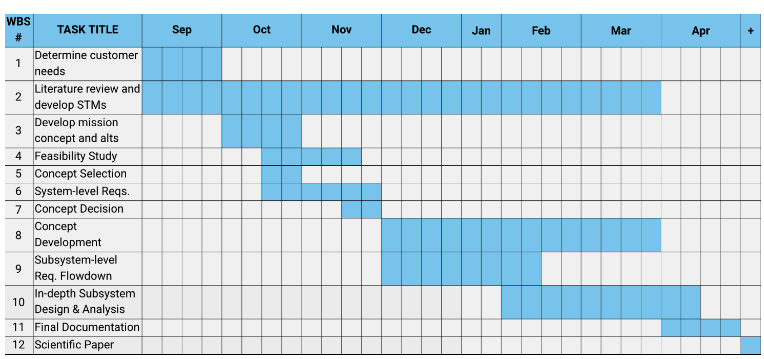

Table 3: PERCI's Gantt Chart

7.1 IMS

Table 50 in Appendix C.4.1 below depicts the Integrated Management System (IMS) of the PERCI mission. It is important to note that this IMS contains only the tasks deemed in-scope for this project specifically. Other tasks important to the mission design but deemed out-of-scope for this course are designated for future work beyond this course, and can be found in Appendix C.4.2.

7.2 Execution Path

For the PERCI project, a structured execution path was followed to guide our mission concept from initial research through final deliverables, as shown in Figure 6. The breakdown of the tasks done is also shown in the WBS mentioned prior. For the execution path, we had certain higher-level tasks to track our progress and then individual milestones for certain larger deliverables. For Milestone 1, we began by performing Decadal Survey research and looking at the UOP mission and identifying questions that could potentially be answered by another mission concept to enhance the mission, this was followed by Instrument identification and scientific analysing the Decadal Survey and identifying scientific priorities that applied for our mission, followed by scientific trade studies and instrument identification to see our options and select optimal ones. We then traced and decomposed functional requirements and developed an initial STM, assessed impact survivability, and explored multiple mission concepts through feasibility and trade studies. This would allow us the achieve the first Milestone, which was deciding on a mission concept.. In Milestone 2, we further refined the technical details of the mission, which includes mounting and release trade studies, thermal analysis, developing a ConOps and an EVA and WBS. We then compiled all work done into our midterm presentation to complete Milestone 2, and marked the transition from conceptual planning to concrete system definition. In Milestone 3, we matured the mission concept by finalising the hardware, materials, controls, trajectory, EOL plan, along with developing risk analysis and management and budget strategies. As Figure 6 below illustrates, most tasks have been completed by this stage, only needing to complete the final report to achieve the 3rd Milestone. This milestone-based approach helped us stay organised and allowed us to flow down requirements and work, ensuring alignment with science goals, technical feasibility, and risk management.

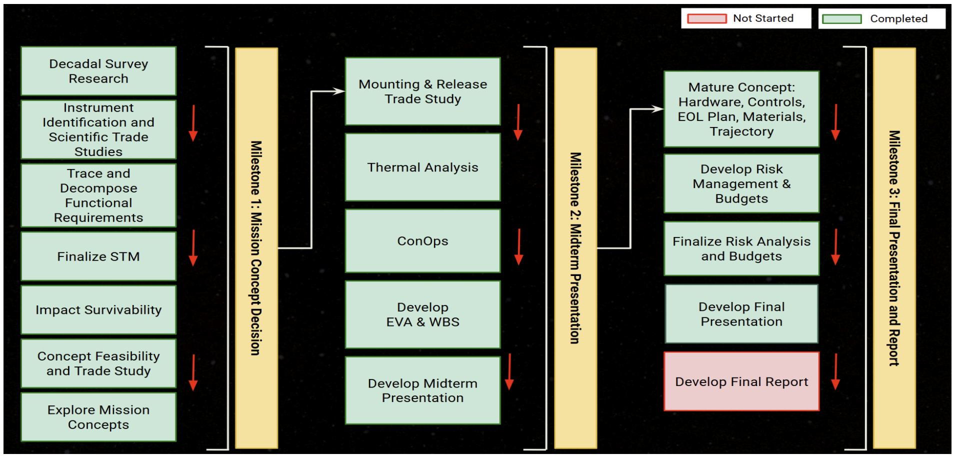

Figure 6: Execution Path

7.4 Deliverables

This project produced a set of key deliverables that lay the foundation for a viable secondary mission to support NASA's Uranus Orbiter and Probe (UOP). These deliverables reflect both the technical and scientific planning required for mission development:

- [x] **Science Traceability Matrix** linking mission goals to specific measurements
- [x] **Instrument recommendations and trade studies** evaluating scientific payload options
- [x] **System and subsystem requirements** detailing functional and technical needs
- [x] **Concept of Operations (ConOps)** outlining mission phases and spacecraft behavior
- [x] **Spacecraft design decisions** supported by trade studies across key subsystems
- [x] **Risk and budget analysis** identifying major challenges and cost considerations
- [x] **Final technical report** documenting the full scope of the project
- [ ] **Conference-ready academic paper** summarizing the mission concept for publication

These outputs form a comprehensive starting point for future development and collaboration with the broader space science community. All deliverables are completed except for the conference-ready academic paper which will be developed in the next few months.

8 Conclusion

This document serves as a summary of the work done for the PERCI mission over the course of this semester. PERCI serves as the secondary payload to the Uranus Orbiter and Probe Mission to enhance the science return of the primary mission. While great strides have been made in developing this penetrator concept, more work needs to be done, including but not limited to:

- Stress calculations to determine: flare, main body, and nose cone interfaces
- Component place optimization
- Material selection for the flare and main body
- Perform shock analysis
- Develop a general penetrator architecture using the UOP design as a case study
- Finalize cost budgets
- Write and complete conference paper

Appendix:

Appendix A: System Level Requirements

A.1 Customer Needs

System ID	Description	Parent Source	Child Requirements	Verification
SYS_CN_1	The system shall fit within the payload fairing of the launch vehicle with [TBR] margin when integrated to the UOP orbiter	Meeting with customer	SYS_CR_5, PAY_1, TTC_2, STRUCT_3	Inspection
SYS_CN_2	The system shall enhance the scientific objectives of the UOP mission	Meeting with customer	SYS_TL_1, SYS_CR_3, SYS_CR_7, SYS_CR_8, SYS_CR_9, SYS_CR_10	Analysis / Test
SYS_CN_3	The system shall use COTS and TRL 6+ technology.	Meeting with customer	SYS_C_1, SYS_C_2, ADCS_2, TTC_3, CDH_3, STRUCT_5 THERM_6, PWR_4, PROP_2	Inspection / Analysis
SYS_CN_4	The system shall adhere to the guidelines listed in NASA's SIMPLEx program	Meeting with customer	SYS_TL_2, SYS_CR_1	Inspection
SYS_CN_5	The system shall not exceed a power draw of [2W] total power from the UOP flagship during transit	Meeting with customer	SYS_TL_8	Analysis / Test
SYS_CN_6	The system shall not impede performance of the UOP mission	Meeting with customer	SYS_TL_3, SYS_TL_4, SYS_TL_5, PAY_1,	Analysis / Test
SYS_CN_7	The system shall do no harm to the UOP orbiter	Meeting with customer	STRUCT_6, THERM_1, PWR_5, PROP_5	Analysis/Test
SYS_CN_8	The system shall communicate via UHF frequency to the UOP orbiter at a maximum data rate of 20.1 kbps	Meeting with customer	SYS_C_5, TTC_1, TTC_2	Analysis/Test

Table 4: Customer Needs Requirement Table

A.2 Top Level Requirements

System ID	**Description**	**Parent Source**	**Child Requirements**	**Verification Method**
SYS_TL_1	The system shall support the current NASA planetary science goals.	SYS_CN_2	FR_1-13	Analysis
SYS_TL_2	The system shall adhere to the NASA planetary protection guidelines for forward contamination	SYS_CN_4	SYS_PP_1-21	Inspection/Tes t
SYS_TL_3	The system shall not impede the UOP orbiter's trajectory during separation from the orbiter	SYS_CN_6, SYS_CN_7	STRUCT_7, STRUCT_8, PROP_6	Analysis/Test
SYS_TL_4	The system shall not have greater than [TBR] [W or W/m²] of heat flow to/from the flagship.	SYS_CN_6	THERM_7	Analysis/Test
SYS_TL_5	The system shall maintain all components within their temperature limits with ±5°C margin	SYS_CN_5	THERM_2, THERM_3, PWR_2, PWR_3	Analysis/Test

Table 5: Top Level Requirements Requirement Table

A.3 Constraints

System ID	Description	Parent Source	Child Requirements	Verification
SYS_C_1	The system shall meet each of the critical development milestones as per the NASA SIMPLEx guidelines	SYS_CR_1	Schedule	Inspection
SYS_C_2	The system shall not exceed a dry mass of 180 kg	SYS_CR_1	Mass Budget	Inspection
SYS_C_3	The system should not exceed a total volume of 0.46 m^3 - if ESPA-grande or propulsive ESPA ring, should not exceed a total volume of 1.19 m^3	SYS_CR_1	Volume Budget	Inspection
SYS_C_4	The system should not exceed a cost of $55 million for all mission phases	SYS_CR_1	Cost Budget	Analysis

Table 6: Constraints Requirement Table

A.4 Critical Requirements

System ID	**Description**	**Parent Source**	**Child Requirements**	**Verification**
SYS_CR_1	The system shall meet each of the critical development milestones as per the NASA SIMPLEx guidelines	SYS_CN_4	SYS_C_1-4	Inspection
SYS_CR_2	The system shall have sufficient power during all mission phases	SYS_CN_2, SYS_CN_5, SYS_CN_6	PWR_1, PWR_2, PWR_3	Analysis/Test
SYS_CR_3	The secondary system shall survive the launch environment	SYS_CN_1	PAY_6, STRUCT_1, THERM_5, PWR_5, PROP_6	Analysis/Test
SYS_CR_4	The payload shall remain functional during relevant mission phases	SYS_CN_2	PAY_3	Analysis/Test
SYS_CR_5	The system shall survive the transit phase	SYS_CN_2, SYS_CN_5	SYS_SD_7, TTC_7, CDH_7, STRUCT_1, PWR_5	Analysis/Test
SYS_CR_6	The system shall survive all deployment phases	SYS_CN_2	PAY_3, TTC_7, CDH_7, THERM_5, STRUCT_1, PWR_5	Analysis/Test
SYS_CR_7	The system shall survive [4.5 (CBE)] hours post-deployment	SYS_CN_2	SYS_SD_7, STRUCT_1, PWR_5	Analysis/Test

Table 7: Critical Requirements Requirement Table

A.5 System Drivers

System ID	Description	Source	Child Requirements	Verification Method
SYS_SD_1	The system should survive 30 minutes post impact.		THERM_9, PWR_7	Analysis / Test
SYS_SD_2	The system shall have a pointing accuracy of 1°		ADCS_3	Analysis / Test
SYS_SD_3	The system shall have 450 m/s delta-V capability.		PROP_1	Analysis / Test
SYS_SD_4	The system should be able to reduce to a maximum impact velocity of [TBR].		PROP_3	Analysis/Test
SYS_SD_5	The system shall be able to have [TBR] for data handling storage		CDH_1	Analysis/Test
SYS_SD_6	The system shall be able to target the Uranian moons of Umbriel and Titania.		PROP_4, TRJ_1	Analysis
SYS_SD_7	The system shall be radiation tolerant to survive in all mission radiation environments	SYS_CR_7, SYS_CR_9	ADCS_5, TTC_4, CDH_4, THERM_8, PWR_6, PROP_7	Analysis/Test

Table 8: System Drivers Requirement Table

A.6 Functional Requirements

System ID	Description	Parent Source	Child Requirements	Verification Method
SYS_FR_1	The system shall support continuous operation to all instruments, as well as the CDH and TT&C subsystems during its descent	STM, SYS_TL_1	PAY_8, TTC_5, CDH_5, THERM_2, THERM_3, WR_1	Test/Analysis
SYS_FR_2	The system shall maintain a continuous line of sight access to the orbiter throughout descent and impact	STM, SYS_TL_1	TRJ_2	Analysis
SYS_FR_3	The system shall have a pointing stability of [TBR]°/sec during descent	STM, SYS_TL_1	ACDS_4	Analysis/Test
SYS_FR_4	The system shall shield the imagers from dust and other hazards during approach during descent	STM, SYS_TL_1	PAY_9, STRUCT_4	Analysis/Test
SYS_FR_5	The system should survive the impact, with all necessary post-impact components operational	STM, SYS_TL_1	PAY_7, TTC_6, CDH_6, STRUCT_2, THERM_5, PWR_8	Analysis/Test
SYS_FR_6	The system should record acceleration data to determine the deceleration profile at the time of impact	STM, SYS_TL_1	PAY_10	Analysis/Test
SYS_FR_7	The system shall be able to reduce to a maximum impact velocity of [TBR]	STM, SYS_TL_1	PROP_3	Analysis/Test
SYS_FR_8	The system shall create a crater with radius >3 meters	STM, SYS_TL_1	PROP_3	Analysis/Test
SYS_FR_9	The system shall maintain a high signal-to-noise ratio across all onboard instruments	STM, SYS_TL_1	PAY_11	Analysis/Test
SYS_FR_10	The system should remain operational in low temperatures moon surface embedded environment at 30K	STM, SYS_TL_1	THERM_9	Analysis/Test
SYS_FR_11	The system should conduct high-frequency temperature measurements following impact	STM, SYS_TL_1	PAY_12	Test
SYS_FR_12	The system shall measure and transmit [TBR (# of data types x 1 matrix)] bits of data of scientific temperature measurement data for each data type	STM, SYS_TL_1	PAY_13, TTC_1, CDH_1, CDH_2	Analysis/Test
SYS_FR_13	The system should reach a depth of >[TBR] meters below the surface	STM, SYS_TL_1	PROP_3	Analysis/Test

Table 9: Functional Requirements Requirement Table

A.7 Science Traceability Matrix Flowdown

STM Flowdown is simplified for clarity and is read down the column not across.					
Decadal Survey Questions	Q12.8a What Can Observations of Material Reveal About Formation of Circumplanetary Systems?	Q4.4b What Materials Ejected from Impact Craters Are Deposited on Planetary Surfaces?	Q5.1c How Does Porosity, Ices, Liquids, or Gases Affect the Physical Properties of the Crust?	Q5.1a How Much Variability in Composition and Internal Structure Is There Within and Between Solid Bodies, and How Did Such Variability Arise and Evolve?	Q8.2f How Are Heat and Material Transported Through—and Ultimately Out of—Satellite Interiors?
Science Goals	Collect images of the surface	Determine the chemical composition of the moon's surface	Determine the physical properties of the large moon's surface at the point of impact	Probe the interior of the Uranian satellites	Determine the thermal properties of the moon's surface
Science Objectives/ Measurements	Determine the spectral composition and topography of the surface Images of surface in nadir and azimuth	Identify and quantify the relative abundances of key chemical elements and minerals present on the moon's surface from the impact plume Vis/NIR Spectroscopy Species: H_2O, CO_2, CH_4, N_2	Determine the surface hardness of the Uranian moons Conduct in-situ ice failure studies on the surface Deceleration profile	Determine the interior composition and structure of the Uranian moons Determine the thickness of the ice sheet Seismic noise: Coda of body waves, harmonic frequency of Crary Waves	Determine the thermal conductivity and heat dissipation rates as well as their variation across the surface Post-impact temperature decay
Nominal Instrument	Downward- facing multi-spectral imager (R, G, B, NIR-1, NIR-2; 10 meter resolution) Side-view imager (Visible; 10 meter resolution)	Vis/NIR Imaging Spectrometer	Accelerometer	Seismometer	Resistive Temperature Detector
Functional Requirement	Image capture, processing, transmission during descent Stable descent trajectory.	Maintain line of sight with the orbiter	Record data at [TBR] readings/sec	High shock tolerance High signal to noise ratio	Low temperature operation High-frequency temperature readings post-impact Probe at least [TBR] meters into the surface

Table 10: Science Traceability Matrix Flow Down Chart

A.8 Verification Matrix

Req. Number	Req. Title	Verification Description	Analysis	Inspection	Demonstration	Test	Verification Description	Verified?
SYS_TL_1	Size Requirement	Verify that full orbiter and probe with the secondary payload attached shall fit within the payload bay of the Falcon Heavy Expendable launch vehicle	*X*	*X*			Inspection through CAD and integration with primary spacecraft.	FALSE
SYS_TL_2	Science Goals	Verify that the system supports the current NASA science goals	*X*				A science traceability matrix will be developed to demonstrate how the system's objectives and instrumentation align with NASA's current planetary science goals.	FALSE
SYS_TL_3	Planetary Protection	Verify that the system adheres to NASA's planetary protection guidelines for forward contamination	*X*	*X*			Analysis and inspections pass criteria provided in NASA guideline	FALSE

SYS_TL_4	Do No Harm to Primary Mission	Verify that the system does not create a single point failure to the primary UOP mission	*X*			*X*	Failure Mode and Effects Analysis (FMEA) along with functional and separation testing will be conducted to mitigate all failure scenarios, and demonstrate no adverse impact on UOP mission operations.	FALSE
SYS_TL_5	Separation from Orbiter	Verify that the separation from Orbiter will not impact the Orbiter's main trajectory	*X*			*X*	Analysis of orbital dynamics, and testing of relevant subsystems.	FALSE

Table 11: Verification Matrix

A.9 Planetary Protection Requirements

System ID	Description	Parent Source	Verification Method
SYS_PP_1	The system shall identify potential for a biological inoculation event caused by a spacecraft resulting from a nominal or off-nominal operation.	NASA-STD-8719.27 4.2.2.4	Test
SYS_PP_2	The system shall capture relevant biological and organic contamination knowledge in an inventory and archive relevant organic and biological reference materials from hardware suppliers and processing environments in order to understand future discoveries, as outlined in Table 4-2.	NASA-STD-8719.27 4.3.1	Test
SYS_PP_3	The system shall document a formal assessment of organic and biological materials present, as defined in paragraphs 4.3.5 and 4.3.6, in the PP Implementation Plan. Category II & IIa missions to Earth's Moon are excluded from this requirement.	NASA-STD-8719.27 4.3.2	Test
SYS_PP_4	The system shall document the PP Organic Inventory in the Pre-Launch PP Report.	NASA-STD-8719.27 4.3.4	Test
SYS_PP_5	The system shall have a PP Organic Inventory that includes all organic materials present on the spacecraft in amounts greater than 1.0 kg.	NASA-STD-8719.27 4.3.5	Test
SYS_PP_6	The system shall document the PP Biological Inventory in the Pre-Launch PP Report that includes estimates of all biological materials (intact organisms, remnants of organisms, and biological molecules) intentionally included as part of the hardware elements reaching the target destination (e.g., payload, spacecraft, launch vehicle, descent stage).	NASA-STD-8719.27 4.3.6	Test
SYS_PP_7	The system shall archive 50 g of organic materials that are present on the spacecraft in amounts greater than 25 kg.	NASA-STD-8719.27 4.3.7	Test
SYS_PP_8	The system shall document any archived organic materials, including intended method, duration, and location of archival preservation, in the Pre-Launch PP Report.	NASA-STD-8719.27 4.3.8	Test
SYS_PP_9	The system shall document any change to the PP Organic Inventory and the PP Biological Inventory after the delivery of the Pre-Launch PP Report in the Post-Launch PP Report and the End of Mission PP Report.	NASA-STD-8719.27 4.3.9	Test
SYS_PP_10	The system shall demonstrate that the management of bioburden during assembly, test, transport, and launch operations is sufficient to reach NASA	NASA-STD-8719.27 4.4.1	Test

	cleanliness requirements, as outlined in Table 4-3.		
SYS_PP_11	The system shall be built in a minimum of an ISO Class 8 cleanroom in accordance with ISO 14644-1:2015(E).	NASA-STD-8719.27 4.4.3	Test
SYS_PP_12	The system shall develop an approach to demonstrate management of contamination in the PP Implementation Plan. Missions may achieve compliance using any single or a combination of methodologies provided in Appendix C. Acceptable methodologies include, but are not limited to, analysis of: a. A Bioburden Control Approach that may include: (1) Bioburden at launch by addressing the following: (a) Managing facility cleanliness. (b) Managing bioburden. (c) Accounting bioburden using: (i) An agreed upon culture-based assay (Refer to Chapter 6). (ii) Specification values for hardware biological cleanliness. (iii)Alternative or augmented approaches (e.g., metagenomics analysis). (d) Preventing recontamination. (2) Breakup and burnup analysis of hardware. b. An Analytical Approach using the following parameters: (1) Relevant elements of the Bioburden Control Approach, and (2) Probability of encountering and impacting the target, including spacecraft reliability. (3) Post-launch bioburden reduction based on environmental conditions experienced in deep space and on the surface of the target body. (4) Mechanisms and timescales of transport to habitable environments at the target body, including surface impact location, lateral transfer, and the subsurface. Note 1: For 4.4.4.a and 4.4.4.b above the alternative and analytical approaches are considered by the OPP on a case-by-case basis and reviewed by a non-advocate review board who submits findings and recommendations to the OPP for concurrence. NASA-STD-8719.27 26 of 62 Note 2: Missions can choose to propose genomic or metagenomics analysis as an alternative to culture-based bioburden accounting.	NASA-STD-8719.27 4.4.4	Test
SYS_PP_13	The system shall document processes for notification of discoveries and anomaly events relevant to planetary protection during mission operations in the PP Implementation Plan.	NASA-STD-8719.27 4.4.5	Test
SYS_PP_14	The system shall document data and analyses used to demonstrate compliance with contamination avoidance requirements in the Pre-Launch PP Report.	NASA-STD-8719.27 4.4.6	Test

SYS_PP_15	The system shall include a record of bioburden accounting data in the Pre-Launch PP Report.	NASA-STD-8719.27 4.4.7	Test
SYS_PP_16	The system shall communicate, in timely fashion, to feed into the decision-making process, considerations that may impact planetary protection compliance (provided in Appendix D) occurring between Safety and Mission Success Review (SMSR) and Launch Readiness Review (LRR)/Flight Readiness Review (FRR) t LRR/FRR.	NASA-STD-8719.27 4.4.8	Test
SYS_PP_17	The system shall document in the Post-Launch PP Report updates to data and analyses provided per paragraphs 4.4.6 and 4.4.7 and any issues that arise after the delivery of the Pre-Launch PP Report, including considerations per paragraph 4.4.8	NASA-STD-8719.27 4.4.9	Test
SYS_PP_18	The system shall prevent occurrence of a biological inoculation event into a potentially habitable environment during landing and surface operations, including both stationary and mobile activities. Mission operation requirements may be achieved through either a bioburden control approach or an analytical approach demonstrated through a probability of occurrence analysis, as specified below.	NASA-STD-8719.27 4.6.1	Test
SYS_PP_19	The system shall demonstrate contamination avoidance at a probability of occurrence less than 1.0 x 10-4 for a biological inoculation event into a potentially habitable aqueous environment (e.g., liquid water body, brine) for 1,000 years.	NASA-STD-8719.27 4.6.3.1	Test
SYS_PP_20	The system shall determine the minimum cleanliness levels needed for the relevant hardware elements driven by the nature and sensitivity of the particular life detection experiments together with applicable strategies for preventing recontamination.	NASA-STD-8719.27 4.6.3.2	Test
SYS_PP_21	The system shall document the final disposition of hardware elements, including any updates to organic, biological, and combustion products inventories, in the End of Mission PP Report.	NASA-STD-8719.27 4.7.1	Test

Table 12: Planetary ProtectionRequirements Requirement Table

Appendix B: Subsystem Analysis
B.1 Payload
B.1.1 OSIRIS REX SamCam/TAGCAMS Specifications

Analyzed Parameter	**Specification**
Mass	~4 kg
Volume	~0.02 m^3
Power Draw	~10 W total
Spectral Range	400-800 nm (Visible)
Field of View	Up to 80° (TAGCAM); MAPCAM: 4°
Frame Rate	~2.8 fps
Integration	Modular; part of OSIRIS-REx OCAMS suite
Radiation/Low-Light Suitability	High; deep-space proven

Table 13: OSIRIS REX SamCam/TAG CAMS Specifications

B.1.2 PCB Piezotronics Model 350B41 Specifications

Analyzed Parameter	**Specification**
Mass	27 g
Shock Tolerance	Up to 1,000,000 g peak
Sensitivity	~0.1 mV/g
Frequency Range	4-10,000 Hz
Operating Temperature	-23°C to 66° C
Type	Piezoelectric Charge Accelerometer
COTS Availability	Yes

Table 14: PCB Piezotronics Model 350B41 Specifications

B.1.3 Artius Seismometer Specifications

Analyzed Parameter	**Specification**
Mass	1.4 kg
Frequency Range	0.03-200 Hz
Sensitivity	500 V•s/m; 145 dB dynamic range
Tilt Tolerance	±5°
Power Consumption	~600 mW
COTS Availability	Yes

Table 15: Artius Seismometer Specifications Specifications

B.1.4 Lake Shore Cryotronics Cernox RTD Specifications

Analyzed Parameter	**Specification**
Operating Temperature Range	0.1 K – 420 K
Power Usage	~1 mW
Thermal Response Time	50 ms – 1 s at 77 K
Measurement Reproducibility	±0.5 mK at 4.2 K, ±16 mK at 77 K
Radiation Tolerance	Maximum ΔT error up to 100 K: ~40 mK
COTS Availability	Yes

Table 16: Lake Shore Cryotronics Cernox RTD Specifications Specifications

B.1.5 Payload Requirements

System ID	Description	Parent Source	Verification Method
PAY_1	The payload shall fit within the secondary system structure	SYS_CN_1, SYS_CN_6	Inspection
PAY_2	All payload instruments shall be COTS and have a TRL level of 6 and above	SYS_CN_3	Inspection
PAY_3	All necessary payload instruments shall remain operational during their requisite mission phases	SYS_CR_4	Analysis/Test
PAY_4	All necessary payload supporting electronics shall remain operational during their requisite mission phases	SYS_CR_4	Analysis/Test
PAY_5	The payload subsystem shall have necessary radiation resistance to survive in all mission radiation environments	SYS_SD_7	Analysis/Test
PAY_6	The payload subsystem shall survive and be operational following all launch and deployment environments (i.e. shock, vibe, acoustics)	SYS_CR_3, SYS_CR_6	Analysis/Test
PAY_7	The payload subsystem should survive and be operational following impact shock environments (i.e. impact)	SYS_FR_5	Analysis/Test
PAY_8	The system shall support continuous operation to all instruments, as well as the CDH and TT&C subsystems during its descent	SYS_FR_1	Analysis/Test
PAY_9	The system shall shield the imagers from dust and other hazards during approach during descent	SYS_FR_4	Analysis/Test
PAY_10	The system should record acceleration data to determine the deceleration profile at the time of impact	SYS_FR_6	Analysis/Test
PAY_11	The system shall maintain a high signal-to-noise ratio across all onboard instruments	SYS_FR_9	Analysis/Test
PAY_12	The system should conduct high-frequency temperature measurements following impact	SYS_FR_11	Test
PAY_13	The system shall measure and transmit [TBR] bits of data of scientific temperature measurement data for each data type	SYS_FR_13	Analysis/Test

Table 17: Payload Subsystem Requirement Table

B.2 TT&C

B.2.1. TT&C Requirements

System ID	Description	Parent Source	Verification Method
TTC_1	The TT&C subsystem shall have >[TBR] and <20.1 kbps data rate	SYS_CN_8, SYS_FR_12	Analysis/Test
TTC_2	The TT&C subsystem shall have > [TBR] Hz frequency in UFH range	SYS_CN_1, SYS_CN_8, SYS_FR_12	Analysis/Test
TTC_3	The TT&C subsystem shall have COTS components with TRL of 6+	SYS_CN_3	Inspection/Analysis
TTC_4	The TT&C subsystem shall have necessary radiation resistance to survive in all mission radiation environments	SYS_SD_7	Analysis/Test
TTC_5	The TT&C subsystem shall be operational during descent	SYS_FR_1	Analysis/Test
TTC_6	The TT&C subsystem should survive and be operational following impact	SYS_FR_5	Analysis/Test
TTC_7	The TT&C subsystem shall survive all pre-operation phases	SYS_CR_7, SYS_CR_8	Analysis/Test

Table 18: TT&C Subsystem Requirement Table

B.2.2 Link Budget Calculations

Item	Symbol	Units	Value	Source	Explanation
Frequency	*f*	GHz	2	Input Parameter	Refer to Frequency Selection table in Communication_Systems_ Sep_25-27 lecture
Transmitter Power (DC)	*P*	W	5	Input Parameter	Assume antenna power = 25% of total bus power
Transmitter Power Amplifier Efficiency	*hp*	-	0.6	Input Parameter	Assume TWTA
Transmitter Power (RF)	*P*	W	5	*R*hp*	
Transmitter Power (RF)	*P*	dbW	6.99	10*log(P)	
Transmitter Line Loss	L_l	dB	1	Input Parameter	Assume based on Table 16-13 from New SMAD
Transmit Antenna Beamwidth	θ_t	deg	70	Input Parameter	New equation provided in Slack ; theta = 21/(f/c)*D
Transmit Antenna Efficiency	h_t	-	0.6	Input Parameter	
Peak Transmit Antenna Gain	G_{pt}	dBi	7.72	Eq. 16-10 from New SMAD	
Transmit Antenna Diameter	D_t	m	0.15	Input Parameter	
Transmit Antenna Pointing Error	e_t	deg	5	Input Parameter	
Transmit Antenna Pointing Loss	L_{pt}	dB	0.061	Communication_Systems_Sep_25-27 lecture (slide 46)	
Transmit Antenna Gain (net)	G_t	dBi	7.66	Gpt + Lpt (Sample_Link_Budget)	Gain after considering pointing loss
Equiv. Isotropic Radiated Power	*EIRP*	dBW	5.67	Equation 16-20 from New SMAD	

Propagation Path Length	S	km	1000	Input Parameter	Distance between transmitter and receiver antennas
Space Loss	L_s	dB	158.5	Eq. 16-22 from New SMAD	
Propagation & Polarization Loss	L_a	dB	0	Circular Polarization	Assume no loss in power
Receive Antenna Diameter	D_r	m	1	Input Parameter	Orbiter and probe document, pg. 19
Receive Antenna Efficiency	h_r	-	0.6	Input Parameter	
Peak Receive Antenna Gain	G_{rp}	dBi	15.0	Eq. 16-9 from New SMAD	
Receive Antenna Beamwidth	θ_r	deg	10.5	Input Parameter	New equation provided in Slack ; theta = 21*(f/c)*D
Receive Antenna Pointing Error	e_r	deg	1.05	Input Parameter	Typically 10% of the beamwidth if tracking is used (also account for GNC limitations)
Receive Antenna Pointing Loss	L_{pr}	dB	0.12	Communication_Systems_Sep_25-27 lecture (slide 46)	
Receive Antenna Gain (net)	G_r	dBi	14.9	Sample_Link_Budget	
System Noise Temperature	T_s	K	70	Input Parameter	Assume Ground Station at room temperature
Data Rate	R	bps	1024	Input Parameter	1024 bps = 1 kbps, 1024^2 bps = 1 Mbps, 1024^3 bps = 1 Gbps, 1 byte = 8 bits
Modulation Rate	--	-	976	Input Parameter	For LoRa, R = BW/2^SF, assume SF7
Computer Implementation Efficiency	--	-	0.9	Input Parameter	Computer efficiency typically around 90%
Effective Data Rate	R	bps	1137.8	Calculated	

E_b/N_o (1)	E_b/N_o	dB	40.6	Fig. 16-16 from New SMAD	Assume modulation method (8PSK - 3)
Carrier-to-Noise Density Ratio	C/N_o	dB-Hz	71.2	Eq. 16-31 from New SMAD	
Bit Error Rate	*BER*	-	1.00E-05	Communication_Systems_Sep_25-27 lecture (slide 46)	Refer to Eq. 16-19 from New SMAD
Required E_b/N_o (2)	Req E_b/N_o	dB	8	Table 16-12 from New SMAD	Assume 8PSK at code rate = 0.8 and spectral efficiency = 2.4 (Table 16-11 from New SMAD)
Implementation Loss (3)	---	dB	-2	Table 16-11 from New SMAD	Typically between -1 and -2 dB
Rain Attenuation (4)	---	dB	0	Input Parameter	
Margin	---	dB	30.6	Calculated	
Bandwidth	*B*	MHz	0.125	Input Parameter	Assumed range - Verify with main UOP mission
Carrier-to-Noise Ratio	*C/N*	dB	80.2		Assumes noise bandwidth = nominal bandwidth
Carrier-to-Noise Ratio	*C/N*	-	104739576.4		
Maximum Data Rate	*Rmax*	bps	3.3		
Exceeds Shannon Limit?	---	---	NO		
Received Power	Prx	dBW	-139.1		
Received Power	Prx	dBm	-109.1		
Above Sensitivity Limit?	---	---	NO		

Table 19. Link Budget

B.3 CD&H

B.3.1 CD&H Requirements

System ID	Description	Parent Source	Verification Method
CDH_1	The CD&H system shall have [TBR] of data storage	SYS_SD_5, SYS_FR_12	Analysis/Test
CDH_2	The CD&H system shall have [TBR] of data processing	SYS_FR_12	Analysis/Test
CDH_3	The TT&C subsystem shall have COTS components with TRL of 6+	SYS_CN_3	Inspection/Analysis
CDH_4	The TT&C subsystem shall have necessary radiation tolerant to survive in all mission radiation environments	SYS_SD_7	Analysis/Test
CDH_5	The TT&C subsystem shall be operational during descent	SYS_FR_1	Analysis/Test
CDH_6	The TT&C subsystem should survive and be operational following impact	SYS_FR_5	Analysis/Test
CDH_7	The TT&C subsystem shall survive all pre-operation phases	SYS_CR_7, SYS_CR_8	Analysis/Test

Table 20. CD&H Subsystem Requirement Table

B.3.2 Ionizing Radiation Exposure

The penetrator will experience the most ionizing radiation when leaving Earth orbit and passing through the Van Allen belts. To mitigate this radiation exposure, a model was generated using SPENVIS which illustrates the total received dose in rads compared to the thickness of an aluminum shield to protect the CDH subsystem, shown in figure 7 below.

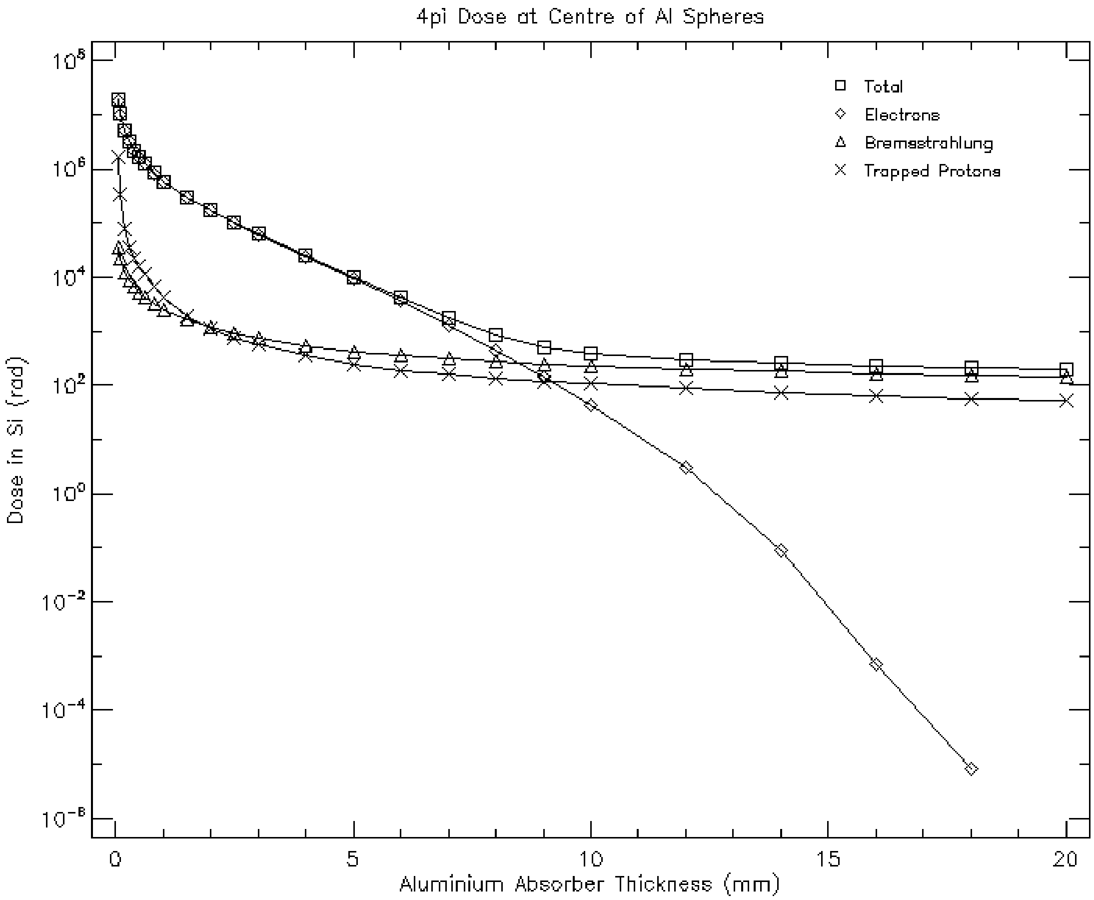

Figure 7: SPENVIS-modeled ionizing radiation dose leaving Earth orbit

Numerical modelling reveals that with a 10mm thickness absorber, the craft would absorb another further 5krad on its trip to the Uranian planetary system. Should a gravitational assist near Jupiter be used, an additional 10-15krad dose will be incurred, depending on flyby altitude.

B.4 ADCS

B.4.1: ADCS Subsystem Requirements

System ID	Description	Parent Source	Verification Method
ADCS_1	The ADCS subsystem shall provide accurate pointing ≤[TBR]° during the deployment phase to support communications with the orbiter	SYS_SD_2	Analysis/Test
ADCS_2	The ADCS subsystem shall all be COTS with a TRL of 6+	SYS_CN_3	Inspection /Analysis
ADCS_3	The ADCS subsystem shall provide accurate pointing ≤1° of the surface of the moon during the impact phase	SYS_SD_2	Analysis/Test
ADCS_4	The ADCS subsystem shall have a pointing stability of [TBR]°/sec during descent	SYS_FR_3	Analysis/Test
ADCS_5	The TT&C subsystem shall have necessary radiation tolerant to survive in all mission radiation environments	SYS_SD_7	Analysis/Test

Table 21: ADCS Subsystem Requirement Table

B.4.2 ADCS Control System and Sizing

The 4 primary disturbance torques (atmospheric, magnetic, solar, and gravitational) were analyzed based on the relationships in SMAD and current CAD data. The atmospheric, magnetic, and solar torques were determined to be negligible. The moons have no atmosphere and therefore will impose extremely minimal to no drag on the spacecraft. None of the moons are suspected of having a magnetic field and the closest moon is over 129,000km from Uranus which makes the magnetic field from the planet negligible as well. Additionally, Uranus is almost 3 billion km away from the sun (19AU) which makes any solar disturbance torque also negligible. The gravity gradient from the largest moon, Titania, and Uranus was analyzed and calculated to be $1.5*10^{-9}$ Nm. This value is incredibly small and is insignificant.

Therefore, the ADCS was sized based on 3 major events: Deployment de-tumble, SRB ejection slew, and final impact approach. With the greatest torque on the spacecraft being the SRB ejection slew, the ADCS system will have to be able to stop and correct the maneuver in the case of failure and will be sized with margin based on this. A trade study was conducted to determine the most optimal mode of attitude control between reaction wheels, momentum wheels, magnetorquers, thrusters, and control moment gyros. Based on this, the selected attitude control type will be reaction wheels due to it satisfying the do no harm requirements to the flagship orbiter. All other types would/could have an effect on the orbiter which the goal is to minimize or prevent.

The total torque that the ACS will have to stop is based on the maximum slew rate of the spacecraft. This was taken from the Propulsion slew requirement of 10 degrees per second of the SRB ejection. Neutralizing this spin, based on the current specifications of the spacecraft, would take an ACS system of about 30mNms 5 minutes. This was deemed allowable as it ensures that the 1 degree pointing to the orbiter, for active communication and data upload, can still take place for the vast majority of the descent time frame. To account for the other modes of torque, a 2x margin/safety factor will be included bringing the size requirement for the reaction wheels to 60mNms. Finally, a source for this specification can be found. Rocket Lab provides this exact specification with a TRL 6+ COTS available with the specifications provided in Table 22.

Reaction Wheel	Maximum Torque	Mass per Unit	Size	Operating Lifetime	Nominal Power Draw	Peak Power Draw
Rocket Lab: 60mMms	SRB Slew Rotation: ~10°/s	288 g	77 mm x 65 mm x 38 mm	5 hours	1.8 Watts	60 Watts

Table 22: Rocket Lab kinda mid Reaction Wheel Data Table

B.4.3 Spin Up/Spin Down

The purpose of a spin up and spin down maneuver is to create gyroscopic stability during SRB firing to minimize the impact of any offset torques that may arise from any of the 4 motors. This would also then ensure that the impact trajectory calculation will be more precise as there is only 1 shot to getting it right with no ability for corrections.

The spin up phase will likely be done via small solid rocket motors mounted on the main body, tangent to the spacecraft. The sizing will be done to create just enough angular momentum in the yaw axis of the spacecraft. This will ensure that any offset SRB thrust magnitude between the 4 motors will equalize and the trajectory will remain consistent. Once the spacecraft confirms SRB shut down through the accelerometer data, a Yo-Yo desaturation mechanism will be deployed. The Yo-Yo consists of 2 strings and 2 masses on opposite sides of the spacecraft which will rotate outward in the opposite direction of the spacecraft spin to neutralize the yaw spin. This spin up and spin down method has been utilized on several space missions, primarily during the deployment burn for interplanetary missions, and has a very high heritage.

B.5 Structures

B.5.1 Structures Subsystem Requirements

System ID	Description	Parent Source	Verification Method
STRUCT_1	The structure subsystem shall contain and protect all components during the launch, transit, deployment, and descent phases	SYS_CR_3, SYS_CR_5 SYS_CR_6 SYS_CR_7	Analysis/Test
STRUCT_2	The structure subsystem should contain and protect all components during the impact phase.	STRUCT_2	Analysis/Test
STRUCT_3	The structure subsystem shall fit within the allocated payload static envelope for the launch vehicle while attached to the orbiter with [TBD] margin	SYS_CN_1	Inspection/Analysis/Demonstration
STRUCT_4	The structure shall shield the imagers from dust and other hazards during approach and descent	SYS_FR_4	Analysis/Test
STRUCT_5	The structure subsystem shall have COTS components with TRL of 6+	SYS_CN_3	Analysis/Test
STRUCT_6	The structure subsystem shall do no harm to the UOP orbiter	SYS_CN_7	Analysis/Test
STRUCT_7	The structure subsystem shall produce less than [TDB] Nm of total torque onto the flagship orbiter during the deployment phase	SYS_TL_3	Analysis/Test
STRUCT_8	The structure subsystem shall produce less than [TDB] m/s of total delta-v onto the flagship orbiter during the deployment phase.	SYS_TL_3	Analysis/Test

Table 23: Structures Subsystem Requirement Table

B.5.2 Nose Cone Shape Trade Study Overview

The shape of the nose cone plays an important role in the penetrator's behavior during impact with the surface. Ideally, the nose cone would allow the penetrator to achieve extremely deep penetration without deflection, yaw, or deformation of the structure. Deeper penetration of the surface increases the time under which the penetrator decelerates, and therefore reduces the loads experienced by the rest of the penetrator and its instruments. Both yaw and deflection (occuring during and after initial penetration of the surface respectively) are deviations from the initial path of the penetrator prior to impact due to the ground closer to the surface being less able to resist deformation than the ground further from the surface. During penetration, deviation from the penetrator's initial path will likely cause significant shear stresses on the craft, which could deform or otherwise damage the craft’s components, and potentially compromise the mission.

Literature review indicated that a blunt, or hemispherical tipped penetrator will be less likely to deflect upon impact with the surface when compared to a conical, or ogive tipped penetrator. This can be seen in figure 8, which shows how the forces acting on a conical-tipped penetrator act to push it upward, whereas the forces on a hemispherical-tipped penetrator are less directed.

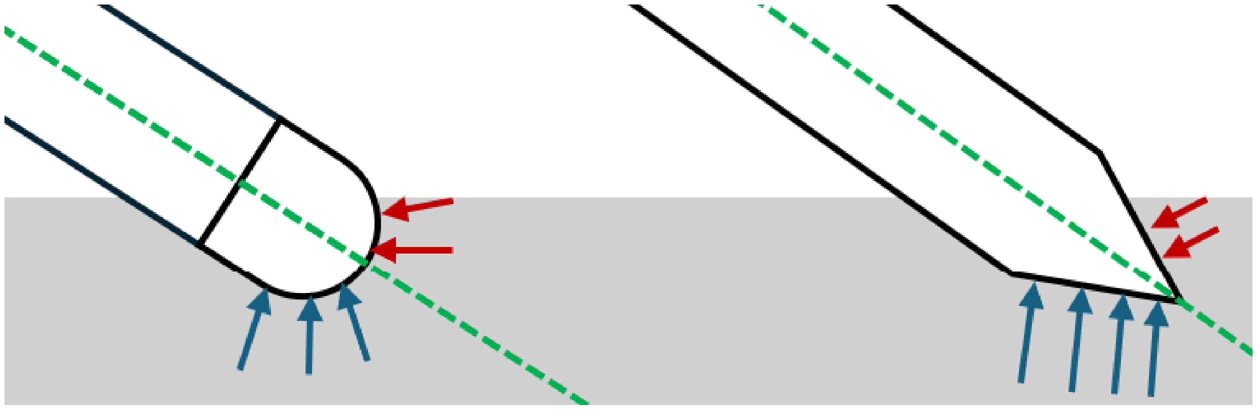

Figure 8: Diagram showing the direction of forces acting on hemispherical-tipped (left) and conical-tipped (right) long-rod penetrators just after impact. The arrows representing the forces are not to scale.

However, after initial penetration of the surface has occurred, ogive tipped penetrators perform much better both in terms of overall penetration, and in terms of yaw resistance [1]. To find a balance between these two nose shapes, an ogive nose cone with a hemispherical tip was selected, shown in figure 9 below. This is because it is a shape that allows for very good surface penetration, while addressing the shortcomings of the sharp-pointed ogive nose cone shape. Those shortcomings being its poor performance when impacting the surface (its tendency to deflect on impact), and its relatively weak tip, which may deform on impact [1].

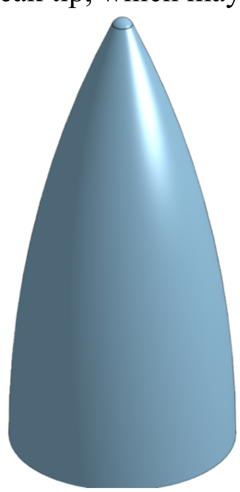

Figure 9: Hemispherically tipped ogive nose cone

B.5.3 Impact Force and Depth Calculations

Two models were used to calculate the impact force and/or depth upon impact of Umbriel. Multiple models were used in order to establish a range of expected values. These models include:

1. Tate-Alekseevski model - Rigid-Body Penetration [3]
2. Spherical Cavity-Expansion Approximation - Ogival Nose Rod [5]

For the Tate-Alekseevski model, the projectile hitting the targets assumes a rigid rod undergoing eroding penetration. The rigid deceleration can be separated from the portion of the projectile undergoing plastic deformation via a modified Bernoulli equation in Equation B.5.3.1.

$$\frac{1}{2}\rho_p(v - u)^2 + Y_p = \frac{1}{2}\rho_t u^2 + R_t \qquad \text{B.5.3.1}$$

Important variables to consider in this equation include the dynamic stress Y_p and the target's resistance to penetration R_t. The dynamic stress of the projectile can be estimated from the Johnson-Cook Method, shown by Equation B.5.3.2.

$$\sigma = \left(A + B\varepsilon^n\right)\left(1 + C\ln\frac{\dot{\varepsilon}}{\dot{\varepsilon}_0}\right)\left\{1 - \left(\frac{T - T_{ref}}{T_{melt} - T_{ref}}\right)^m\right\} \qquad \text{B.5.3.2}$$

Where A is the material yield stress, B is the straining hardening modulus, ε is the deformation strain, n is the strain hardening coefficient, C is the strain rate sensitivity coefficient, $\dot{\varepsilon}$ is the strain rate, $\dot{\varepsilon}_0$ is the reference strain rate, T is the operating temperature, T_{ref} is the reference temperature, T_{melt} is the melting temperature, and m is the thermal softening coefficient. Assuming a solid Titanium rod of grade Ti-6Al-4V, the following material properties were determined or assumed from Deb et. al., 2022 [2], shown in table 24.

Property	Value
T_{ref}	308 K
T_{melt}	1878 K
$\dot{\varepsilon}$	0.2
$\dot{\varepsilon}_0$	0.01
ε	0.0067
n	0.88
A	971.63 MPa
B	650.67 MPa
m	0.7468

Table 24: Titanium Ti-6Al-4V Assumed Properties [2]

Once the dynamic stress is calculated, it can be related by a scaling law represented by the variable λ, a constant independent of velocity and accounting for material effects. For the Tate-Alekseevski model, a value of 0.7 was determined to be best for dynamic effects. This relationship is shown by Equation B.5.3.3.

$$Y_p = (1 + \lambda)\sigma_{Yp} = 1.7\sigma_{Yp} \quad \text{B.5.3.3}$$

Where Y_p is the dynamic flow stress, and σ_{Yp} is the flow stress of the material. Solving gives a dynamic flow stress of the projectile at 0.8 GPa. In order to estimate the target's resistance to penetration, R_t, the following relation in Equation B.5.3.4 can be used, derived from the Tate solenoidal model.

$$R_t = \sigma_{Yt}\left[\frac{2}{3} + ln\left(\frac{0.57E_t}{\sigma_{Yt}}\right)\right] \quad \text{B.5.3.4}$$

Where E_t is the target's modulus of elasticity, and σ_{Yt} is the target flow stress. Given there is little information on the material properties of water ice at temperatures as low as 70 K, the assumed values were derived from ice in polar ice caps. Using Glen's flow law for ice given by Equation B.5.3.5, where $\dot{e}_t$ is the strain rate of the ice, A is the flow rate factor, assumed to be 1.3E-24 [4], and n is the strain hardening coefficient, assumed to be the standard value of 3. The ice upon impact will have a very high strain rate as it tries to get out of the path the penetrator takes. Thus, we can equate it to the ratio between the longitudinal and transverse strain rate at the moment of impact. For the impact velocity on Umbriel, this is approximately 4.

$$\sigma_{Yt} = \frac{\dot{e}_t}{A}^{1/n} \qquad \text{B.5.3.5}$$

Solving for the flow stress of ice gives an approximate value of 0.15 GPa. Plugging this value into equation B.5.3.5, and assuming a modulus of elasticity of 9.1 GPa for typical glacier ice, the target's resistance to penetration is calculated to be approximately 0.6 GPa. From here, the rigid-body deceleration on the body is then given by Equation B.5.3.9.

$$u = \frac{v - \mu\left(v^2 + A\right)^{1/2}}{1 - \mu^2} \qquad \text{B.5.3.6}$$

$$\mu = \left(\frac{\rho_t}{\rho_p}\right)^{1/2} \qquad \text{B.5.3.7}$$

$$A = \frac{2\left(R_t - Y_p\right)\left(1 - \mu^2\right)}{\rho_t} \qquad \text{B.5.3.8}$$

$$\frac{dv}{dt} = -\frac{1}{\rho_p L}\left(\frac{1}{2}\rho_t v^2 + R_t\right) \qquad \text{B.5.3.9}$$

Where ρ_p is the density of the projectile, ρ_t is the density of the target, v is the impact velocity (for rigid body penetration, velocity is given by u in Equation B.5.3.6), and L is the length of the projectile. Plugging these values into Equations B.5.3.6, B.5.3.7, B.5.3.8, and B.5.3.9 gives an instantaneous deceleration of approximately 94000 m/s^2, or approximately 9600 G's. Assuming a mass of 110 kg (subtracting approximately 40 kg of SRBs from the total mass of around 150 kg), this translates to a force of approximately 10.4 MN.

The Tate-Alekseevski model also allows penetration depth to be estimated by Equation B.5.3.10.

$$\frac{P}{L} = \frac{\rho_p}{\rho_t} ln\left(1 + \frac{\rho_t V^2}{2R_t}\right) \qquad \text{B.5.3.10}$$

Where P is the penetration depth and V is impact velocity. Plugging appropriate values in gives a depth of approximately 3.4 meters [3].

As mentioned previously, to establish edge cases for the impact, a second model was used. This model, called Ogival Nose Rods with Spherical Cavity-Expansion Approximation, takes into account the profile of the projectile's nose. For the purposes of this analysis, rods with an ogive nose of caliber radius head (CRH) of 5 will be considered. As with the Tate-Alekseevski model, it takes into account the material properties of the target, and the frictional resistance at the projectile-target interface. The force applied longitudinally along the penetrator's length can be calculated using Equation B.5.3.10.

$$F_z = \alpha_s + \beta_s V_z^{\ 2} \qquad \text{B.5.3.11}$$

Where F_z is the longitudinal force and V_z is the rigid body penetration velocity (for this analysis, the same rigid body penetration velocity calculated for the Tate-Alekseevski model can be substituted for this value). The variables α_s and β_s can be calculated from Equations B.5.3.12 and B.5.3.13.

$$\alpha_s = \pi a^2 K A_s \left[1 + 4\mu\psi^2\left(\pi/2 - \theta_0\right) - \mu(2\psi - 1)\left(4\psi - 1\right)^{1/2}\right] \qquad \text{B.5.3.12}$$

$$\beta_s = \pi a^2 \rho B_s \left[\frac{(8\psi - 1)}{24\psi^2} + \mu\psi^2\left(\pi/2 - \theta_0\right) - \frac{\mu(2\psi-1)\left(6\psi^2+4\psi-1\right)(4\psi-1)^{1/2}}{24\psi^2}\right] \qquad \text{B.5.3.13}$$

$$\theta_0 = sin^{-1}\left(\frac{2\psi - 1}{2\psi}\right) \qquad \text{B.5.3.14}$$

$$A_s = \frac{2Y}{3K}\left\{1 + ln\left[\frac{(1-2v)K}{(1-v)Y}\right]\right\} \qquad \text{B.5.3.15}$$

Where a is the radius of the projectile, Y is the target flow stress, K is the bulk modulus of the target, A_s is a constant given by Equation B.5.3.15, μ is the sliding friction coefficient of the target, v is Poisson's ratio of the target, ψ is the CRH of the nose cone, θ_0 is a geometric angle of the ogive nose determined from the CRH, and B_s is an experimentally determined value that for this calculation, will be omitted.

Assuming a solid titanium penetrator with a 0.058 meters, the same flow stress as calculated for the previous model, a standard bulk modulus of 10.84 GPa for ice at 70 K (Neimeier 2018, [7]), a slide friction coefficient of 0.03, and a Poisson's ratio of 0.33, Equations B.5.3.12, B.5.3.13, B.5.3.14, and B.5.3.15 can be solved and plugged into Equation B.5.3.11 to get the estimated force on the nose cone. The force calculated is approximately 5.4 MN.

Lastly, similar to the Tate-Alekseevski model, the depth of penetration can be estimated by Equation B.5.3.16.

$$P = \frac{m}{2\beta_s} ln\left[1 + \frac{\beta_s V_0^{\ 2}}{\alpha_s}\right] \qquad \text{B.5.3.16}$$

Where P is final penetration depth, m is the mass of the penetrator and V_0 is the impact velocity. Plugging in appropriate values gives a depth of approximately 8.3 meters [5].

Observations of these results show the range of impact forces and penetration depths is large, but is expected given the number of unknowns regarding the material properties of the surface, and the varying assumptions made by both models. The Spherical Cavity-Expansion Approximation has a much lower force for a few reasons: it is not dependent on the penetrator mass, just its geometry. Secondly, since it takes into account a nose cone profile geometry, the force absorbed by the structure would be less than the Tate-Alekseevski model, which assumed the total cross-sectional area of the rod striking the surface at normal incidence. Furthermore, the Spherical Cavity-Expansion has a higher penetration depth for the same reasons. Taking into account the ogival nose shape allows the rod to penetrate deeper than if it has a flat incident surface.

Thus, the expected impact force range is 5.4-10.4 MN and the penetration depth is between 3.4 and 8.3 meters.

B.6 Thermal

B.6.1 Thermal Subsystem Requirements

System ID	Description	Parent Source	Verification Method
THERM_1	The thermal subsystem shall not impede on the main UOP mission	SYS_CN_7	Analysis / Test
THERM_2	The thermal subsystem shall maintain all components within their survival temperature limits with ±5°C margin for relevant mission modes	SYS_TL_5, SYS_FR_1	Analysis / Test
THERM_3	The thermal subsystem shall maintain all components within their operational temperature limits with ±5°C margin for relevant mission modes	SYS_TL_5, SYS_FR_1	Analysis / Test
THERM_4	The thermal subsystem shall survive and be operational following all launch and deployment environments (i.e. shock, vibe, acoustics)	SYS_CR_5, SYS_CR_8	Analysis / Test
THERM_5	The thermal subsystem should survive and be operational following impact shock environments (i.e. impact)	SYS_FR_5	Analysis / Test
THERM_6	The thermal subsystem shall use COTS and TRL 6+ technology	SYS_CN_3	Inspection / Analysis
THERM_7	The thermal subsystem shall not allow greater than [TBD] W of heat flow to/from the flagship.	SYS_TL_4	Analysis / Test
THERM_8	The thermal subsystem shall have necessary radiation tolerance to survive in all mission radiation environments	SYS_SD_7	Analysis / Test
THERM_9	The thermal subsystem should maintain all components within their operational temperature limits with ±5°C margin for 30 minutes in low temperatures moon surface embedded environment at 30K	SYS_SD_1, SYS_FR_11	Analysis / Test

Table 25: Thermal Subsystem Requirement Table

B.6.2 Thermal Analyzes
B.6.2.1 1-Node Thermal Analysis

Model Setup

A 1-node thermal model was created in Thermal Desktop for this thermal analysis. The model file is "box_around_Uranus.dwg", located in the team Google drive here. The representation of the spacecraft consists of a 1-node rectangular cube solid object. The dimensions of the object are 1.6m x 1.06m x 0.96m, representing the maximum SIMPLEx volume limits. These dimensions give the maximum allowable surface area to maximum teh radiation out surface area. The object's node was put in the SPACECRAFT submodel. The object has MLI turned on, with the MLI node set to a different node ID and submodel, the submodel being MLI. Thermal Desktop automatically connects the base object node to the MLi node and will calculate the heat transfer based on the MLI property's effective emissivity (e*) value. There will also be one case where the MLI option is turned off.

The SPACECRAFT's outer faces are added to the radiation analysis group BASE (Thermal Desktop's default radiation group). The face's optical properties are varied based on the case. The optical properties used for this analysis are shown in Table 26 below.

Name	**Solar Absorptivity, α**	**IR Emissivity, ε**	**α/ε**	**Source**
AL Coated PET	0.14	0.035	4.0	Shadahl Redbook [8]
AL Tape	0.1	0.03	3.3	Spacecraft Thermal Control Handbook [9]
Gold Polyamide	0.3	0.03	10.0	Shadahl Redbook [8]

Table 26. Thermal Model Optical Coatings

The SPACECRAFT object's material property was set to Al 6061, the values of which come from the *Spacecraft Thermal Control Handbook* [9]. The MLI was set to the MLI property. Table 2 below shows the material properties used in the thermal model. The e* value in the MLI value is a symbol called "e_star". The e* values are derived from Figure 5.4 in the *Spacecraft Thermal Control Handbook* [9, pg 165]. Each coverage area for each side of the spacecraft is going to be about 1 m^2, so the e* values will range from the historic missions with similar coverage area, giving a range from 0.03-0.005. Three values are used for this analysis: 0.03, 0.01, and 0.005. For context, the lower the MLI's e*, the better it is at preventing heat transfer through the MLI. The thermophysical properties used for this analysis are shown in Table 27.

Name	K [W/m/°C]	ρ [kg/m³]	Cp [J/kg/°C]	e*	Source
Al 6061	155.8	2770	961.2		Spacecraft Thermal Control Handbook [9]
MLI	1	1	1	e_star	e_star = {0, 0.03, 0.01, 0.005}

Table 27. Thermal Model Material Properties

A node heater object is attached to the SPACECRAFT node. The heater power is set to a very high value, 200 W in this case. For now, this value doesn't matter for how the heater object is used. The heater is set to proportional control mode with on/off temperatures set to +/- 0.1°C of the set target value. This set target value is -10°C for this model, which is a common minimum operating temperature for batteries we saw in datasheets, -20°C, + 10°C analysis uncertainty margin. Thus the on temperature is -10.1°C, and the off temperature is = 9.9°C. With the heater set up like this, during steady state analyses, the heater's input load will converge to a value that results in the heater node being between the on and off temperatures, so approximately -10°C. In post-processing, we can record this value, which will be the minimum internal heat we are looking for in this analysis.

Case Setup

The cases for this analysis will all be steady-state analyses, as transient solutions are not necessary for the current goal. The cases will be set up to represent an absolute worst-case cold scenario, where the spacecraft is radiating heat out to space while receiving no environmental heating from sources like the Sun or Uranus. In each case, the external face optical property and MLI e* value will be set to that analysis's settings, resulting in 7 total cases. Table 28 lists out each case and the associated coating and e* value.

Case #	**e***	**Outer Coating**
1	0.03	Gold Polyamide
2	0.01	Gold Polyamide
3	0.005	Gold Polyamide
4	0.03	AL Coated PET
5	0.01	AL Coated PET
6	0.005	AL Coated PET
7	-	AL Tape

Table 28. One Node Thermal Analysis Case Setup Values

Results

Table 29 shows the results for all cases below. The lowest required internal heat was 7.9 W from case 5, where e* = 0.005 and the coating was gold polyamide. The highest required internal heat was 54 W from case 7, where no MLI was used and the coating was aluminum tape. The best result from this analysis, 7.9 W (rounded up to 8 W) will be used as an input into the power analysis. The results also show how important MLI will be for passive thermal management of the spacecraft, and will have to be considered as integral to the spacecraft's design moving forward.

Case #	**e***	**Outer Coating**	**Internal Power Load (W)**
1	0.03	Gold Polyamide	27.4
2	0.01	Gold Polyamide	13.7
3	0.005	Gold Polyamide	7.9
4	0.03	AL Coated PET	29.5
5	0.01	AL Coated PET	14.2
6	0.005	AL Coated PET	8.1
7	-	AL Tape	54

Table 29. Feasibility Thermal Analysis Results

B.6.2.2 70-Node Thermal Analysis

Model Setup

A multi-node thermal model was created in Thermal Desktop for this thermal analysis. The representation of the spacecraft consists of a 70-node rectangular cube solid object. The model represents the best estimate of the penetrator at the time of modeling. Figure 10 below shows the model. In a second model, a representation of the orbiter was added for transit cases requiring the orbiter. The external surfaces have MLI turned on, with the MLI node set to a different node ID and submodel, the submodel being MLI. Thermal Desktop automatically connects the base object node to the MLi node and will calculate the heat transfer based on the MLI property's effective emissivity (e*) value. There will also be one case where the MLI option is turned off.

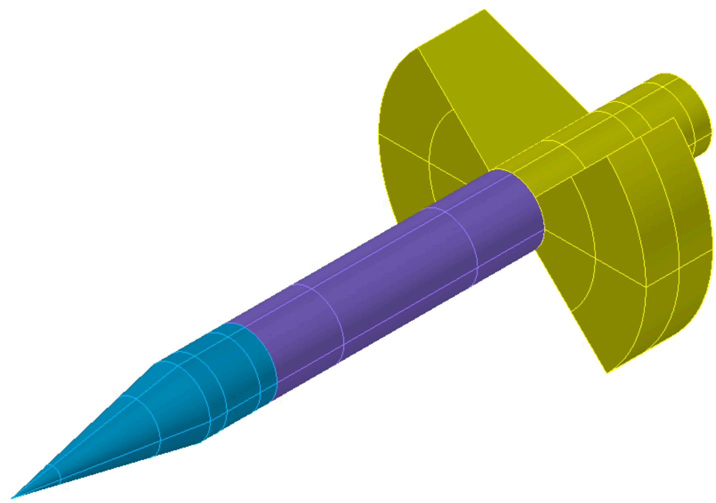

Figure 10 70-Node Thermal Desktop Model

The exterior surface faces are added to the radiation analysis group EXTERNAL, while interior faces are added to the INTERNAL group. The face's optical properties are varied based on the case. The optical properties used for this analysis are shown in Table 30.

Name	**Solar Absorptivity, α**	**IR Emissivity, ε**	**α/ε**	**Source**
AL Coated PET	0.14	0.035	4.0	Shadahl Redbook [8]
Bare_AL	0.1	0.03	3.3	Spacecraft Thermal Control Handbook [9]
Gold Polyamide	0.3	0.03	10.0	Shadahl Redbook [8]

Table 30. Thermal Model Optical Coatings

All objects assume the material property Al 6061, the values of which come from the *Spacecraft Thermal Control Handbook* [5]. The MLI was set to the MLI property. Table 2 below shows the material properties used in the thermal model. The e* value in the MLI value is a symbol called "e_star". The e* values are derived from Figure 5.4 in the *Spacecraft Thermal Control Handbook* [5, pg 165], where a conservative value of 0.01 was selected. The thermophysical properties used for this analysis are shown in Table 31 below.

Name	K [W/m/°C]	ρ [kg/m³]	Cp [J/kg/°C]	e*	Source
Al 6061	155.8	2770	961.2		Spacecraft Thermal Control Handbook [9]
MLI	1	1	1	0.01	Spacecraft Thermal Control Handbook [9]

Table 31. Thermal Model Material Properties

A node heater object is attached to the BATTERY object. The heater power is set to a very high value, 100 W in this case. The heater is set to proportional control mode with on/off temperatures set to +/- 2.5°C of the set target value. This set target value is -41.5°C for this model, which is above the operating temperature for batteries assumed, -55°C, +10°C analysis uncertainty margin. Thus the on temperature is -44°C, and the off temperature is -39°C. The ice embedment cases are unique as the battery heater objects are replaced with constant heat load objects. The heat load was adjusted to match the desired operating time. This is to determine the time for the batteries to fall below operating temperature limits.

Case Setup

A total of seven cases were performed using this model, shown in table 32. Four transit cases in Uranian orbit in hot and cold environments and with or without attachment to the orbiter (set to -20°C) were considered. Figure 11 below shows the estimated Uranian orbit used. A hot flyby case around Jupiter was also performed. Figure 12 below shows the estimated jupiter flyby path used. Finally, two ice embedment cases.

Case #	**Connected to Orbiter?**	**Orbit**	**Solar Constant (W/m^2)**	**Albedo**	**IR Planetshine**	**Surface Ice Temp. [K]**
Uranus Orbit, Hot, Oribter	Yes	Uranus Orbit 1	3.7	0.9	0.8	-
Uranus Orbit, Hot, No Oribter	No	Uranus Orbit 1	3.7	0.9	0.8	-
Uranus Orbit, Cold, Oribter	Yes	Uranus Orbit 1	3.7	0.9	0.5	-
Uranus Orbit, Cold, No Oribter	No	Uranus Orbit 1	3.7	0.9	0.5	-
Jupiter Flyby	Yes	Jupiter Flyby	50.1	0.3	14.1	-
Ice Embedment, 70K	No	-	-	-	-	70
Ice Embedment, 30K	No	-	-	-	-	30

Table 32. Feasibility Thermal Analysis Case Setup Values

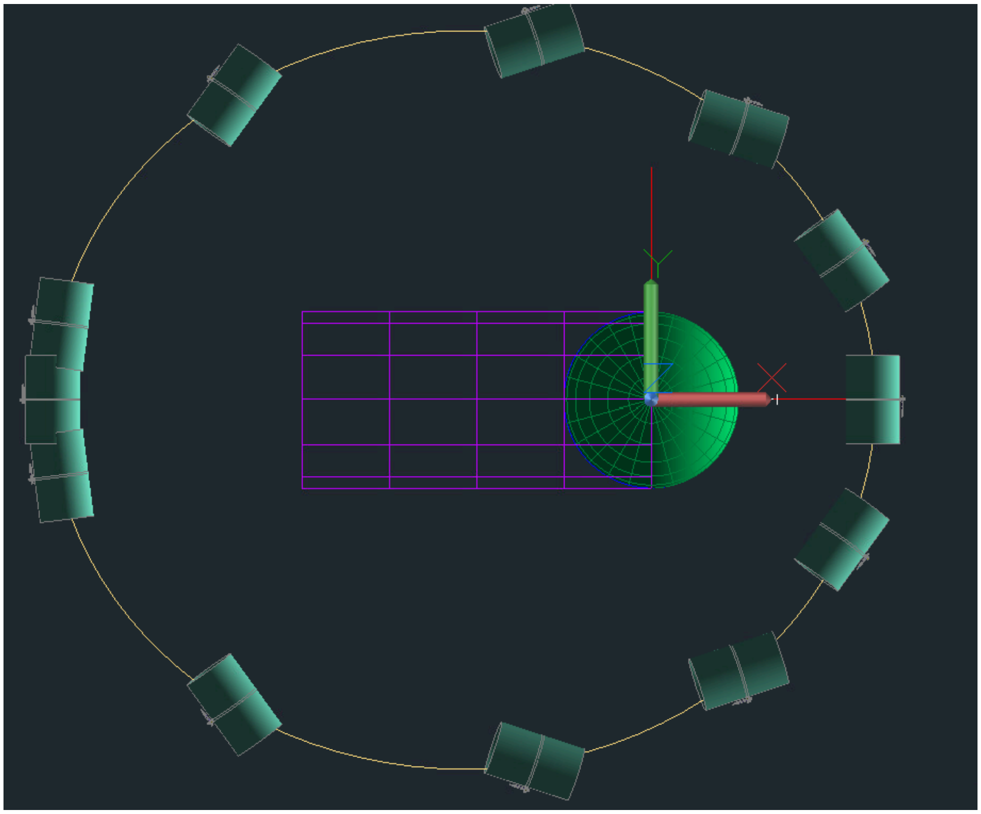

Figure 11. Uranus Orbit 1

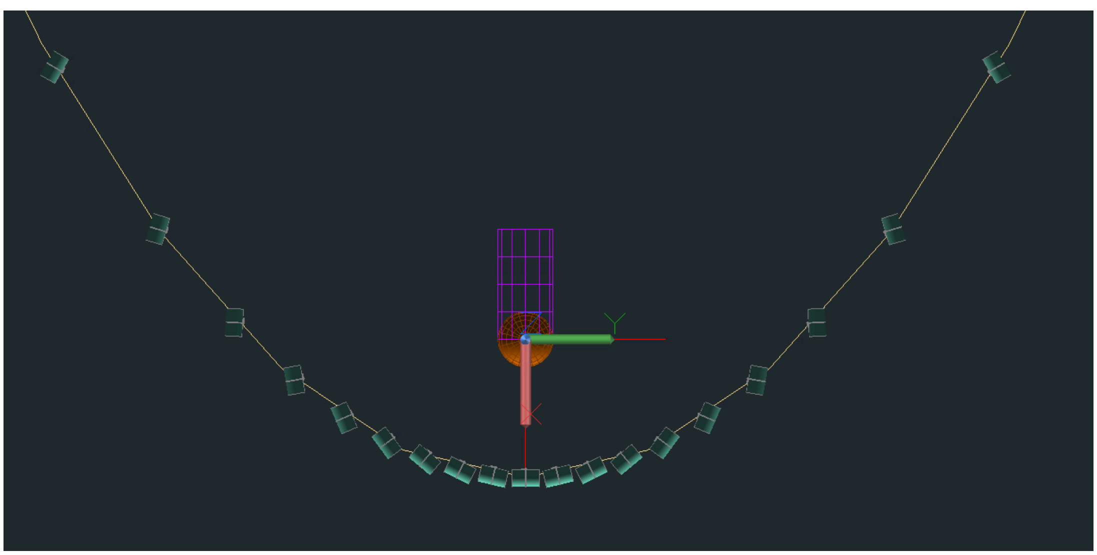

Figure 12. Jupiter Flyby

Results

Table 33 shows the results for all orbital cases below. The maximum average heater power required is <1 W. This is a good indicator for the overall lifetime as there were concerns this heater power would have needed to be higher. The thermal isolation assumed drove the heater power required down for the initial 1-node estimates. The battery operating temperature limits were maintained in all cases.

Case #	**Min. Battery Temperature [°C]**	**Max. Battery Temperature [°C]**	**Avg. Heater Power [W]**
Uranus Orbit, Hot, Oribter	-20.2	-20.2	0
Uranus Orbit, Hot, No Oribter	-39.0	-39.0	0.64
Uranus Orbit, Cold, Oribter	-20.2	-20.1	0
Uranus Orbit, Cold, No Oribter	-39.0	-39.0	0.64
Jupiter Flyby	24.7	24.9	0

Table 33. 70-Node Orbital Thermal Analysis Results

The ice embedment cases results are shown in Table 34. After 10 minutes post-impact, it requires significant operational heater power to keep the batteries within their operating temperature limits. To meet the requirement of operating for 30 minutes post-impact, at least 135 W of operational heater power is required, rounded up to 140 W.

Survival Time	Ice embedment, 70K Internal Heat (W)	Ice Embedment, 30K Internal Heat (W)	Notes
1 min	0	0	From initialized temps, survives w/o heater power
10 min	9	32	
30 min	102	135	
1 hour	128	163	
6 hours	147	184	
12 hours	148	185	
1 day	148	185	
1 week	148	185	
1 month	148	185	

Table 34. Ice Embedment Analysis Results

B.6.2.3 Orbiter Mounting Analysis

Model Setup

Offshoot models of the 70-node model were used. Four models total were created to represent four ways of mounting to the orbiter. The four are shown in figure 13 below. Three models represent if the penetrator was mounted in an inset within the orbiter. The final model represents if the penetrator was side mounted, as assumed in the regular 70-node model.

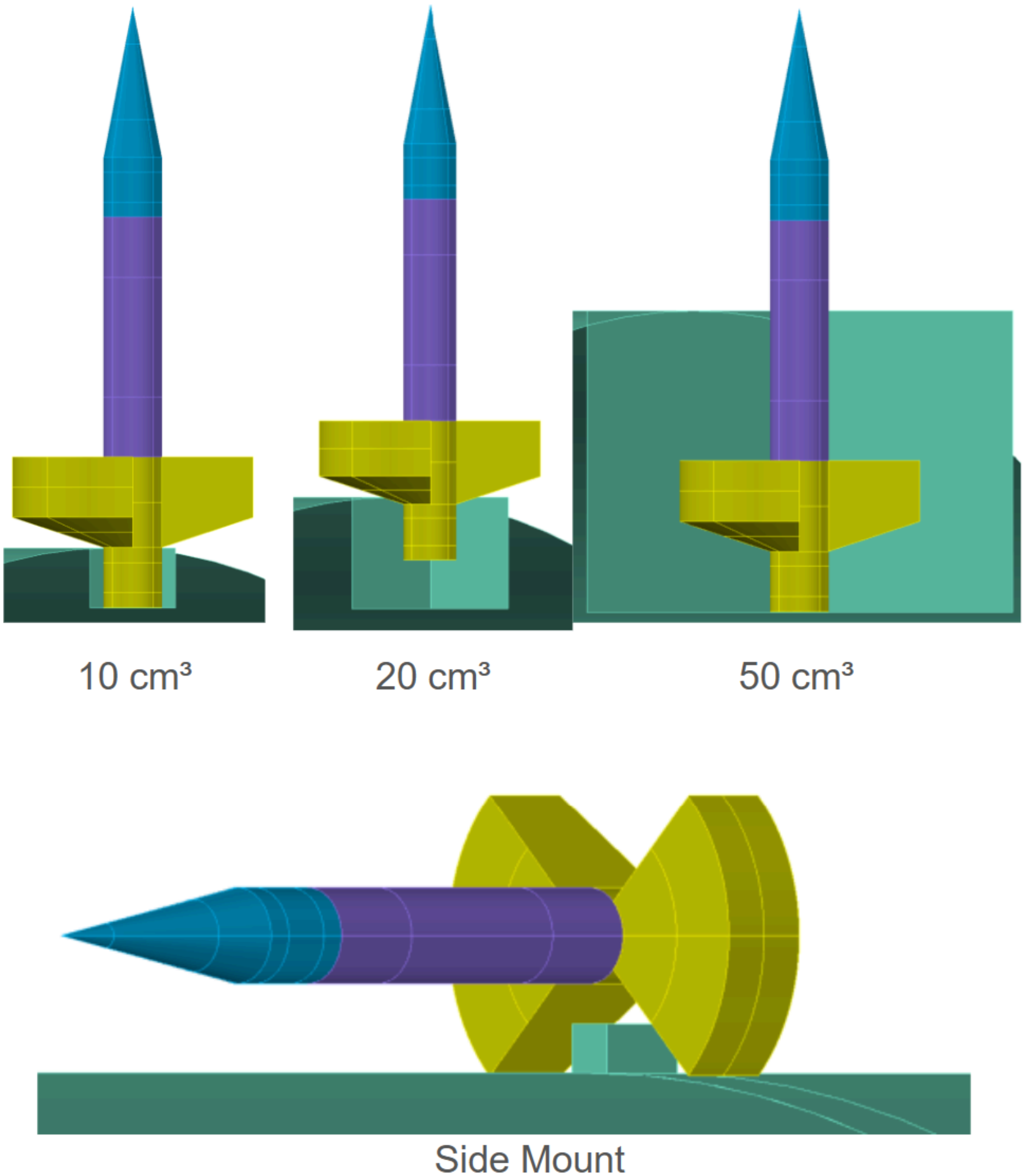

Figure 13: Mounting Configurations onto the Orbiter

Case Setup

Three cases were run for each mounting configuration. The cases changed the external surface temperature of the orbiter to the following values: -40 °C, -20°C, and 0°C. This is to account for the uncertain spacecraft temperature of the orbiter. A base case for compassion values were calculated by hand.

Results

Table 35 shows the results for all cases below. The mounting configuration with the least change to the orbiter's radiating heat out is the side mount, increasing the heat out up to 1 W. In the -40°C, heat was actually being drawn from the penetrator to the spacecraft. Thus, it is recommended that the penetrator use the side mounting configuration.

Output	Battery Heater Steady State Power (W)			dQ Orbiter Out, w/ Penetrator (W)			dQ Orbiter Out, w/o Penetrator (W)		
Spacecraft Ext Temperature (°C)	-40	-20	0	-40	-20	0	-40	-20	0
Base	-	-	-	0.00	0.00	0.00	0.00	0.00	0.00
Inset Mount 10cm^3	3.55	0.00	0.00	-2.90	0.90	1.22	0.23	0.33	0.44
Inset Mount 20cm^3	3.55	0.00	0.00	-2.42	1.57	2.12	0.94	1.30	1.77
Inset Mount 50cm^3	3.54	0.00	0.00	1.43	6.89	9.35	5.88	8.17	11.08
Side Mount	3.55	0.00	0.00	-3.01	0.74	1.01	0.11	0.16	0.21

Table 35. Mounting Comparison Thermal Analysis Results

B.6.3 Thermal Management System

B.6.3.1 Passive Control

The radiative passive control batteries being covered in MLI. The MLI will need to be good quality. The current design uses MLI with an effective emissivity of 0.01, with gold polyamide outer coating with a IR emissivity of 0.03 for external faces, and aluminum coated PET outer coating with a IR emissivity of 0.035 for battery faces.

The conductive passive control currently assumes a normal thermal resistance between the three sections of the penetrator, using 10 W/m/K (industry standard assumption) along the mounting edges. There is a good thermal isolation assumption for the battery-structure mounting, ~ 1W/K derived from 10 W/m²/K, about 1/10th the assumed value of a good bolted connection and a contact area of 0.1 m².

There is no need for a radiator as the system is in safe mode during the hottest environmental cases, confirmed by the 70-node analysis in Appendix B.6.2.2 above.

B.6.3.2 Active Control

Survival and operational heaters will be required for this spacecraft. Survival heaters will be sized to provide 1W of heat to temperature sensitive components in transit and safe modes. Operational heaters are predicted to provide a maximum of 140W of heat to necessary components (i.e. scientific instruments).

B.7 Power
B.7.1 Power Requirements

System ID	Description	Parent Source	Verification Method
PWR_1	The power subsystem shall provide adequate power during all mission phases	SYS_CR_2, SYS_FR_1	Analysis/Test
PWR_2	The power subsystem shall provide adequate power to keep the batteries within their survival temperatures during all mission phases	SYS_TL_5, SYS_CR_2	Analysis/Test
PWR_3	The power subsystem shall provide adequate power to keep the batteries within their operating temperatures during the required mission phases	SYS_TL_5, SYS_CR_2	Analysis/Test
PWR_4	The power subsystem shall have COTS components with TRL 6+	SYS_CN_3	Inspection / Analysis
PWR_5	The power subsystem's chemical battery cells shall not break containment from housings during the launch, transit, or deployment phases	SYS_CN_7, SYS_CR_3, SYS_CR_5, SYS_CR_6, SYS_CR_7	Inspection / Analysis
PWR_6	The power subsystem shall have necessary radiation resistance to survive in all mission radiation environments	SYS_SD_7	Analysis/Test
PWR_7	The power subsystem should provide adequate power post impact for 30 minutes	SYS_SD_1	Analysis/Test
PWR_8	The power subsystem should survive and be operational following impact shock environments (i.e. impact)	SYS_FR_5	Analysis/Test

Table 36: Power Subsystem Requirement Table

B.7.2 Power Budget

ON DESCENT/IMPACT ONLY			
Instrument	Worst-Case Power Draw (W)	Duty Cycle	Average Power Draw (W)
Simera Sense Multispectral Imager	5	1	5
Side-View Imager	5	1	5
Uniaxial Impact Accelerometer	0.25	1	0.25
ADCS	60	0.035	2.1
MEMS Seismometer	1.5	0.01	0.015
Temperature Probe	0.01	1	0.01
Heating Coils	140	0.07	9.8
C&DH	2	1	2
Comms	5	1	5
	TOTAL POWER DRAW (W)		29.175
PROLONGED MISSION ONLY			
Instrument	Worst-Case Power Draw (W)	Duty Cycle	Average Power Draw (W)
Simera Sense Multispectral Imager	5	0	0
Side-View Imager	5	0	0
Uniaxial Impact Accelerometer	0.25	0	0
ADCS	60	0	0
MEMS Seismometer	1.5	1	1.5
Temperature Probe	0.01	1	0.01
Heating Coils	140	1	140
C&DH	2	1	2
Comms	5	0.05	0.25
	TOTAL POWER DRAW (W)		143.76
		CELL VOLUMETRIC ENERGY DENSITY (Wh/L)	248
		CELL GRAVIMETRIC ENERGY DENSITY (Wh/kg)	330
	TOTAL ENERGY (J)		705145.5
	TOTAL MASS (kg)		0.59356
	TOTAL VOLUME (L)		0.78981
		4s48p configuration for safe current distribution:	192 cells total needed
		FINAL MASS (kg)	4.032
		FINAL VOLUME (L)	4.1472

Table 37: Power Budget

B.8 Trajectory

B.8.1: Trajectory Subsystem Requirements:

System ID	**Description**	**Parent Source**	**Verification Method**
TRJ_1	The trajectory subsystem shall be able to target the Uranian moons of Umbriel and Titania	SYS_SD_6	Analysis
TRJ_2	The trajectory subsystem shall maintain a continuous line of sight access to the orbiter throughout descent and impact	SYS_FR_2	Analysis

Table 38: Trajectory Requirements Table

B.8.2 Flyby Data Table

Body	Date	Phase (deg)	Altitude (km)	Speed (km/sec)	Lat (deg)	Lon (deg)
Titania	08/11/45	13.77	384.69	2.97	34.11N	91.07E
Titania	11/14/45	44.58	72.37	4.05	60.94N	64.46E
Titania	02/01/46	33.62	62.76	4.1	51.78N	81.93E
Titania	04/03/46	43.21	142.76	3.74	59.92N	75.09E
Titania	05/25/46	38.73	26.69	4.28	55.68N	84.30E
Titania	07/07/46	24.37	25.75	4.28	39.17N	98.21E
Titania	08/11/46	73.16	25.77	4.28	75.42N	3.27E
Titania	09/15/46	73.68	38.36	4.22	77.21N	3.27E
Titania	10/20/46	74.12	53.42	4.15	78.98N	3.56E
Titania	11/24/46	75.26	93.27	3.95	80.43N	359.60E
Titania	12/28/46	74.96	71.65	4.05	82.40N	4.92E
Titania	02/01/47	75.53	76.19	4.03	84.10N	5.20E
Titania	03/08/47	75.73	86.29	3.99	85.86N	11.02E
Titania	04/12/47	76.42	96.79	3.94	87.48N	12.85E
Titania	05/17/47	93.12	175.55	3.61	72.71N	294.23E
Umbriel	06/22/47	118.51	5354.46	0.52	1.79S	77.31E
Oberon	07/28/47	106.02	1038.56	1.62	4.16S	66.71E
Ariel	07/29/47	126.31	3786.72	0.76	0.18S	82.60E
Ariel	08/30/47	53.31	1821.57	1.39	0.19N	264.17E
Umbriel	10/03/47	118.14	3652.93	0.73	2.61N	76.52E
Titania	11/07/47	159.86	6505.63	0.47	1.88S	292.06E
Oberon	12/09/47	105.44	462.15	2.4	8.53N	65.78E
Ariel	01/10/48	44.24	1478.66	1.66	45.15N	99.27E

Figure 14: Fly-bys provided by the UOP concept study

B.9 Propulsion

B.9.1: Propulsion Subsystem Requirements:

System ID	Description	Parent Source	Verification Method
PROP_1	The propulsion subsystem shall provide a minimum of 450 m/s of delta-V during all relevant mission phases	SYS_CR_3	Analysis/Test
PROP_2	The propulsion subsystem shall have COTS components with TRL 6+	SYS_CN_3	Inspection
PROP_3	The propulsion subsystem shall reduce to a relative velocity of X m/s during descent, and should reduce to a relative velocity of X m/s prior to impact	SYS_SD_4, SYS_FR_7, SYS_FR_8, SYS_FR_13	Analysis
PROP_4	The propulsion subsystem shall be able to target the Uranian moons of Umbriel and Titania	SYS_SD_6	Analysis
PROP_5	The propulsion subsystem shall do no harm to the UOP orbiter	SYS_CN_7	Analysis/Test
PROP_6	The propulsion subsystem shall survive and be operational following all launch and deployment environments (i.e. shock, vibe, acoustics)	SYS_CR_3, SYS_CR_6	Analysis/Test
Prop_7	The propulsion subsystem shall be radiation tolerant to to survive in all mission radiation environments	SYS_SD_7	Analysis/Test

Table 39: Propulsion Requirements Table

B.9.2: Propulsion Selection

The propulsion method trade study was conducted weighing the difference between solid rocket motors, cold gas, hydrazine, electric, and liquid chemical propulsion. The criteria for wants and needs were on the basis of low mass, low volume, low cost, high thrust, high efficiency, short ignition time, high impulse, starting complexity, storage lifetime, power requirements, temperature limitations, and do no harm requirements. The do no harm, long storage time, and low mass allocation requirements held the highest weight as those 3 are major limiting factors for the system. The scores for each item under the 5 propulsion types were set and totalled with the winner being solid rocket motors having a score of 699 out of 830.

With the propellant type selected, we moved to selecting and sourcing a rocket motor from heritage missions with a TRL level of 6 and up. After research into different options, the Star 5D solid rocket motor, previously used on the NASA Pathfinder Mars Mission, was found and analyzed for mass and volume given the initial data sheet. Each motor has a total mass of approximately 10kg while fully loaded, approximately 3kg empty, and a total volume of 9250cm^3. These values meet the maximum requirement values allowed for the propulsion system therefore the DeltaV requirement can be analyzed.

B.9.3 DeltaV Calculations

To calculate the DeltaV that the Star 5D can provide, the DeltaV rocket equation (equation B.9.3.1) will be employed provided below:

$$\Delta v = ISP * g_0 ln \frac{M_0}{M_f}$$

Equation B.9.3.1: DeltaV Equation

All of the key variables are given in the data sheet or can be calculated given the projected mass budget. Taken from the data sheet, the ISP, or efficiency of the engine, is 256 seconds and g_0 is 9.81m/s^2. M_0 and M_f are the initial and final masses of the total spacecraft respectively where M_0 is the mass with the fully loaded SRBs (10kg) and M_f is the mass of the spacecraft with empty SRBs (3kg). Given the volume requirements imposed on the spacecraft and design of the aft flare, only four SRBs are able to fit properly on the spacecraft. Therefore, if the DeltaV requirements cannot be met with 4 SRBs, a different rocket motor will have to be used.

The DeltaV calculation was run using 1, 2, 3, and 4 rocket motors to predict the number of motors that would be required to meet the DeltaV requirements of 450 m/s and the table to results is provided in Table 40.

# of Motors	ΔV [m/s]
1 Motor	139
2 Motors	265
3 Motors	379
4 Motors	476

Table 40: DeltaV Calculations for 1, 2, 3, and 4 Motor Configurations

As shown in the table, using 4 motors allows the spacecraft to achieve the 450 m/s requirement with some minor margin. Finally, these DeltaV values, albeit accurate, are still preliminary and will need to be further refined as the spacecraft design and mass budget are further matured.

B.9.4 SRB Ejection

To mitigate the risk of damage to the main body and nose cone by the solid rocket motors exhaust, the SRBs will be mounted to the spacecraft flare with the nozzles pointing aft. This means to achieve the retrograde maneuver, the nose cone tip will have to be pointing opposite of the direction of travel. This means that after the SRBs exhaust their propellant and eject, the attitude control system will have to flip the spacecraft around to have the tip forward during the impact phase. Therefore, to avoid potential ADCS reaction wheel saturation the SRBs will eject in a staggered manner.

Upon SRB shutdown, the two motors on one side of the spacecraft will eject via mounted, spring based mechanism and induce a 180 degree slew where then the other two motors will eject to stop the rotation. This maneuver must not exceed 10 degrees per second to ensure that the internal reaction wheels can still stop the rotation without the risk of saturation in the case of spring lock up/failure.

Appendix C: Other

C.1 Previously considered designs

C.1.1 Orbiter

The orbiter concept was intended to focus on the properties of the magnetosphere around Uranian moons. Equipped with a magnetometer, cameras, and plasma analyzers, it would continuously collect data during fly-bys.

C.1.2 Impactor

The impactor concept was designed to collide with the surface of Uranian moons to collect scientific data. The information collected would be transmitted during descent, while the impactor would not survive. Any plume generated by impact could be analyzed by instruments on-board the main vehicle to determine the chemical composition of the moon's surface.

C.1.2 Impactor and Penetrator

The impactor and penetrator concept was designed to have one penetrator, which would be launched first from the flyby. Subsequent flybys would release impactors, causing seismic activity that the penetrator would pick up. From this seismic activity, data which describes the inside of the moon could be collected and analyzed to determine features such as density or material composition.

C.1.3 Trade Study

Criteria	Choices: 1 Penetrator		Penetrator + Impactors		Multiple Impactors		Orbiter	
Must Haves	Info (estimate for option)	Y/N	Info	Y/N	Info	Y/N	Info	Y/N
Shall have a maximum mass allocation of 180 kg	<= 180 kg	Yes	<= 180 kg	Yes	<= 180 kg	Yes	<= 180 kg	Yes
Shall have a maximum volume allocation of 1.19 m³	<= 1.19 m³	Yes	<= 1.19 m³	Yes	<= 1.19 m³	Yes	<= 1.19 m³	Yes
Provide novel science return to flagship mission	9	Yes	10	Yes	At least 1	Yes	5	Yes
Does not exceed maximum power draw from orbiter in transit	<= 2 W	Yes	<= 2 W	Yes	<= 2 W	Yes	<= 2 W	Yes

Wants	WT	1 Penetrator: Info	Value	Score	Penetrator + Impactors: Info	Value	Score	Multiple Impactors: Info	Value	Score	Orbiter: Info	Value	Score
Technical Complexity	9	Biggest complexity comes from designing a bus capable of surviving high shocks	0.4	3.6	Biggest complexity comes from designing a bus capable of surviving high shocks from penetrator. Addition design of seperate impactors.	0.2	1.8	Low eng. design complexity, most likely 'dumb systems' post delta-v manuevers.	0.6	5.4	Requires trades to properly allocate mass for either propellant for delta-v versus batteries for longer lifetime	0.6	5.4
			0.5	4.5		0.3	2.7		0.8	7.2		0.7	6.3
			0.6	5.4		0.4	3.6		1	9		0.8	7.2
Operational Complexity	8	High complexity for trajectory to impact moon	0.5	4	Highest complexity, need to precisely coordinate release of penetrator then impactors	0	0	High complexity for trajectory to impact moon	0.5	4	Needs complex trajectory design. Longer CONOPs scheduling design also brings the overall score down.	0.3	2.4
			0.6	4.8		0.05	0.4		0.6	4.8		0.4	3.2
			0.7	5.6		0.1	0.8		0.7	5.6		0.5	4
Amount of Science Objectives Achieved	10	9 science objectives, implies the mission provides highly valuable science data the orbiter can not collect on its own, less objectives if penetrator doesn't survive. Score = (# objectives) / (highest possible objectives over all concepts(10))	0.7	7.0	10 science objectives, provides highly valuable science data the orbiter can not collect on its own, additional science than penetrator alone. Score = (# objectives) / (highest possible objectives over all concepts(10))	0.7	7.0	Only creates 1-2 additional science objectives for the overall mission, but objectives are very novel (is this correct?). Less if created debris can't be observed by orbiter mass spec. Score = (# objectives) / (highest possible objectives over all concepts(10))	0.1	1.0	5 objectives, most if not all of the objectives are already covered by the main mission. Score = (# objectives) / (highest possible objectives over all concepts(10))	0.3	3.0
			0.8	8.0		0.9	9.0		0.2	2.0		0.4	4.0
			0.9	9.0		1.0	10.0		0.2	2.0		0.5	5.0
Survival potential through all mission phases	3	Medium survival potential, large drop in potential comes from the impact and surface survival phases	0.5	1.5	Medium survival potential, large drop in potential comes from the impact and surface survival phases. Only need to worry about the penetrator	0.5	1.5	High survival potential, only need to design for the space environment during flybys and the Uranian system. Do not need to design for the impact phases. Only complexity would be their relatively small size to survive the transit phase	0.6	1.8	Highest survival potential, need to design for the space environment around Jupiter and Uranus	0.75	2.25
			0.7	2.1		0.7	2.1		0.9	2.7		0.9	2.7
			0.8	2.4		0.8	2.4		1	3		1	3
Number of possible targets	4	Lowest number of targets, can reasonably impact only one moon while maintaining mass, volume, and power constraints. Score = (# of possible targets)/(max # of targets of all concepts)	0.25	1	Lowest number of targets, can reasonably impact only one moon while maintaining mass, volume, and power constraints. Score = (# of possible targets)/(max # of targets of all concepts)	0.25	1	High number of target moons, can be sent to multiple targets, will be constrained by delta-v of each impactor. Score = (# of possible targets)/(max # of targets of all concepts)	0.5	2	targets include ring and 3 possible moons, but likely cam send to only one target. Score = (# of possible targets)/(max # of targets of all concepts)	0.5	2
			0.5	2		0.5	2		0.75	3		0.75	3
			0.75	3		0.75	3		1.0	4		1	4
Opportunities to collect and transmit science	6	Plausible, low margin	0.2	1.2	Plausible, low margin, lower score due to additional minimum time associated with impactor offset	0.05	0.3	Definetly possible on an impactor, just need lifetime to impact	0.8	4.8	Plausible, trajectory and line of sight calculations were not performed	0.4	2.4
			0.3	1.8		0.2	1.2		0.9	5.4		0.6	3.6
			0.5	3		0.25	1.5		1	6		0.8	4.8
Influence on orbiter	7	Demonstrated low power draw and thermal influence, small release mechanism area/volume required could possibly have multiple units increasing influence on orbiter. Orbiter instruments OPTIONAL for additional science goal	0.5	3.5	Demonstrated low power draw and thermal influence, small release mechanism area/volume required, multiple releases necessary. Orbiter instruments OPTIONAL for additional science goal.	0.25	1.75	Demonstrated low power draw and thermal influence, small release mechanism area/volume required, multiple releases necessary. Lower mass impactors increases the number and complexity of the release mechanism. Orbiter instruments NECESSARY for any science collection	0.1	0.7	Demonstrated low power draw and thermal influence, ~ESPA ring release mechanism area/volume required	0.1	0.7
			0.75	5.25		0.5	3.5		0.2	1.4		0.25	1.75
			0.9	6.3		0.75	5.25		0.3	2.1		0.4	2.8
Previous Heritage Missions	5	Mars 96 Deep Impact	0.75	3.75	Heritage of both penetrators and impactors, but non-together	0.25	1.25	Deep Impact Deep Space 2 DART	1	5	Multiple heritage small sat missions for lunar reconassaince. Heritage explanetary recon missions, but no small sat form.	0.5	2.5
			0.75	3.75		0.25	1.25		1	5		0.5	2.5
			0.75	3.75		0.25	1.25		1	5		0.5	2.5
Total Score		Minimum		25.55			14.60			24.70			20.65
		Nominal		32.20			22.15			31.50			27.05
		Maximum		38.45			27.80			36.70			33.30

Table 41. Concept Selection Trade Study

C.2 Subsystem Trade Studies

C.2.1 Camera Selection Trade Study

Trade Matrix

PERCI Instrument Trade Study: Cameras

Choices Criteria	DRACO Imager (Baseline)		OSIRIS REX SamCam/TAGCAMS (ECAM M50)		Deep Impact Impactor Targeting Sensor (ITS)		SDL Deep Space Imager		Crystal Space Cubsat Camera	
Must Haves	Info	Y/N	Info	Y/N	Info	Y/N	Info	Y/N	Info	Y/N
Shall have a maximum mass of ~5 kg.	9.55 kg, but this is also one of the primary instruments for DRACO so they were probably able to provide more mass/volume for this	No	4 kg, SAMCAM interfaces with the TAGCAM so lumped the mass together	Yes	10 kg, approximate mass, the HRI/MRI mass is supposed to be 90 kg. and the MRI is much smaller and the same as ITS with no focal wheel	No	1 kg, small cubesat like imagers that are rad tolerant	Yes	50 g, we can stack these but not sure if they are rad tolerant	Yes
Shall have a maximum volume of ~0.02 m^3.	0.03 m^3, rough guess based on the diameter and length	No	0.02 m^3, making an estimate based on the diameter and length of both cams	Yes	0.02 m^3, making an estimate based ont he diameter and length	Yes	0.002 m^3	Yes	0.0001 m^3	Yes
Does not exceed maximum power draw of the secondary payload. (~10W)	Uses 4.95 W power for electronics and 5.2 W for heaters, so barely making it	No	~10 W. total power draw of camera suite is 30.2 W	Yes	30 W, the MRI uses 40W, so I assumed that it would be less	No	2 W	Yes	1 W	Yes
Shall capture and store or transmit imaging data prior to impact		Yes		Yes		Yes		Yes		Yes

Wants	WT	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score
Survivability (Radiation / Thermal)	10	Deep space tested, good sensitivity and thermal stability	0	0	Deep space tested, and rad tested within the last 10 years	0	0	Designed for deep space, and survived impact, and rad hardened	0	0	Explicitly states it is rad hardened, designed for LEO, GEO, and long term orbits	0	0	Rad hardened but not proven in deep space, good for LEO	0	0
	10		1	10		1	10		1	10		1	10		0.6	6
	10		1	10		1	10		1	10		1	10		1	10
Operation in Low Light Conditions	9	Optimized for imaging dark asteroids surfaces	0	0	Large optics? (f/3.5) and adjustable exposure, good low light ability	0	0	navigating and imaging in a dim low light environment	0	0	and integration time, which is good for low light	0	0	No mention of control or binning, or a sensitiviy to low light readings	0	0
	9		1	9		1	9		0.9	8.1		1	9		0.4	3.6
	9		1	9		1	9		1	9		1	9		1	9
Capabilities (OF: frequency value is the max observed frequency and normalized to min: 400 nm and	9	400-1000 nm, Visible to NIR	0	0	400-800 nm, bandpass, six position filter wheel, visible	0	0	400-700 nm, visible	0	0	Optional RGB Bayer or hyperspectral filters add to DISC's wide range of	0	0	400-700 nm, probably basic visible	0	0
	9		1	9		0.6666666667	6		0.5	4.5		1	9		0.5	4.5
	9		1	9		1	9		1	9		1	9		1	9
Provide ADCS Support During Transit	8	Used in DART's SMARTNAC guidance system	0	0	The point of TAGCAM is to provide navigation and ADCS for the impactor.	0	0	Was used for ADCS when in transit to the comet	0	0	Can perform centroiding and tracking but nothing about navigation	0	0	No onboard nav features that use this camera, seems like its mainly to take pictures and not for tracking	0	0
	8		1	8		1	8		1	8		0.5	4		0.3	2.4
	8		1	8		1	8		1	8		1	8		1	8
Wide FOV	7	(0.29 deg), but high resolution and not optimized for	0	0	with the 3.85 mm focal length and if	0	0	and enough for guidance so probably good enough?	0	0	compact form factor and less than that of the	0	0	view, less than the 80 deg that OCAM offered	0	0
	7		0.4	2.8		0.7	4.9		0.7	4.9		0.3	2.1		0.5	3.5
	7		1	7		1	7		1	7		1	7		1	7
High Frame Rate	6	Sample every 5 seconds. Good at prox but not high speed vid	0	0	2.8 fps (200 Mbit/s SpW), looked this up its not high speed but enough for navigation and helpful for ADCS	0	0	Not focused on high frame rate, more on navigation imaging	0	0	Says it is adjustable and not fully defined, but the system appears to be very flexible	0	0	Captures data at 70 deg/rotation, seems to be optimized for rapid capture	0	0
	6		0.4	2.4		0.6	3.6		0.3	1.8		0.5	3		1	6
	6		1	6		1	6		1	6		1	6		1	6
Ease of Integration	5	Large, heaviest in this suite, can't just be used as a standalone	0	0	modular, this could be a good thing for us, we can pick and choose the	0	0	enough to remove the filter wheel, this could be useful for us	0	0	lightweight and modular, looks like it can be used for cubesats and	0	0	small, low power, seems to adapt to multiple different	0	0
	5		0.1	0.5		0.7	3.5		0.9	4.5		0.9	4.5		1	5
	5		1	5		1	5		1	5		1	5		1	5
Previous Heritage	4	Most recent deep space impact type mission	0	0	Recent, and successful flight on OSIRIS REX,	0	0	mission and provided science relevant data	0	0	credibility, no deep space flight related to this instrument	0	0	Claims heritage but only in LEO, not for deep space	0	0
	4		1	4		1	4		1	4		0.5	2		0.4	1.6
	4		1	4		1	4		1	4		1	4		1	4
EMI/EMC Compatibility	3	Likely rated, unsure, should be okay with SEIS	0	0	Likely rated, unsure, should be okay with SEIS	0	0	instruments like HRI so assuming its probably good	0	0	Designed with rad tolerance so most likely compliant but	0	0	Low power helps, not sure if its certified	0	0
	3		0.6	1.8		0.6	1.8		0.6	1.8		0.6	1.8		0.5	1.5
	3		1	3		1	3		1	3		1	3		1	3
Onboard Processing	2	Some info on compression filtering etc, but prob not fully	0	0	Some onboard processing? Not fully all there	0	0	like it went directly to the primary spacecraft	0	0	and tracking, seems like it explicitly has more onboard	0	0	No onboard processing, will require reliance on external	0	0
	2		0.5	1		0.5	1		0.4	0.8		1	2		0.3	0.6
	2		1	2		1	2		1	2		1	2		1	2
Total Score		Minimum		0	Minimum		0	Minimum		0	Minimum		0	Minimum		0
		Nominal		**48.5**	Nominal		**51.8**	Nominal		**48.4**	Nominal		**47.4**	Nominal		**34.7**
		Maximum		55	Maximum		55	Maximum		55	Maximum		55	Maximum		55

Table 42. Camera Selection Trade Study

C.2.2 Accelerometer Selection Trade Study

PERCI Instrument Trade Study: Accelerometer																
Choices		HBK TYPE 8309			CRL Model 383			CRL Model 876			PCB Piezotronics Model 350B41			PCB Piezotronics Model 250B42		
Links		TYPE 8309 Piezoelectric Charge Shock Accelerometer, Integral Side Cable			Columbia Research Laboratories - Model 383			Columbia Research Laboratories - Model 876			Model 350B41			Model 350B42		
Must Haves		Info		Y/N	Info		Y/N	Info		Y/N	Info		Y/N	Info		Y/N
Commercial off the shelf (COTS)		COTS		Yes	COTS This sensor is built by a research laboratory - very low inventory		No	COTS This sensor is built by a research laboratory - very low inventory		No	COTS		Yes	COTS		Yes
Piezoelectric (Charge Accelerometer)		Piezoelectric		Yes	Piezoelectric		Yes	Piezoelectric		Yes	Piezoelectric		Yes	Piezoelectric		Yes
Low mass		3 grams		Yes	0.28 Oz = ~7.37 grams		Yes	1.2 Oz = ~34.02 g		Yes	27 g		Yes	27 g		Yes
Wants	**WT**	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score
Low dynamic noise range	6	23 mg	1	6	N/A	0	0	N/A	0	0	N/A	0	0	N/A	0	0
High amplitude range / shock tolerance (>13000 g's)	10	Highest operating shock 15000 g's	1	10	Shock limit 20000 g's	0.7	7	Shock limit 10000 g's (1/2 sine)	0.5	5	1000000 g peak	1	10	50000 g peak	0.8	8
High sensivity - How many output volts per unit of acceleration (~10 mV/g)	7	0.004 pC/g Does not give its capacitance, so I am not sure what its sensitivity in voltage is	0	0	0.5 pC/g = 1.5 mV/g	0.85	5.95	@-420 deg F to -295 deg F 25 pC/g with 5% variability = 1.47 mV/g	0.853	5.971	0.05 mV/g	0.995	6.965	0.1 mV/g	0.99	6.93
High frequency range - within frequency range of SRS (~3 KHz) NOTE: Important to determine frequency response before final selection	7	1-54000 Hz If the instrument can measure the maximum frequency expected, it passes	1	7	2-15000 Hz If the instrument can measure the maximum frequency expected, it passes	1	7	2 - 4000 Hz If the instrument can measure the maximum frequency expected, it passes	0.8	5.6	4 - 10000 Hz If the instrument can measure the maximum frequency expected, it passes	1	7	4 - 10000 Hz If the instrument can measure the maximum frequency expected, it passes	1	7
High usable bandwidth Hard to determine without frequency response	8	N/A		0	N/A		0	N/A		0	N/A		0	N/A		0
Amplitude non-linearity	7	N/A	0	0	1% / 2000 g	0.99	6.93	1% / 2000 g	0.99	6.93	2.5% / 10000 g	0.975	6.825	2.5% / 10000 g	0.975	6.825
Large operating temperature range Ideal: -30 - > 150 C	10	-74 - 180 C Score = (range below op temp - range above max temp spike)/100	1	10	-54 - 175 C	1	10	-267 - 149 C	1	10	-23 - 66 C Operating	0.9	9	-23 - 66 C operating	0.9	9
Vacuum safe Not sure how to look for this information on data sheets	7															
Previous heritage	8															
		Nominal		33	Nominal		36.88	Nominal		33.501	Nominal		39.79	Nominal		37.755

Table 43. Accelerometer Selection Trade Study

C.2.3 Seismometer Selection Trade Study

PERCI Instrument Trade Study: Seismometer

		Trillium Compact Horizon			Artius			Episensor			Trillium Compact		
Links		https://nanometrics.ca/hubfs/Downloads/Data%20Sheets/trillium_compact_horizon.pdf			https://www.guralp.com/products/artius			https://kinemetrics.com/wp-content/uploads/2017/04/datasheet-episensor-es-t-f			https://nanometrics.ca/hubfs/Downloads/Data%20Sheets/trillium_compact.pdf		
Must Haves		Info		Y/N	Info		Y/N	Info		Y/N	Info		Y/N
Commercial off the shelf (COTS)		COTS		Yes	COTS		Yes	COTS		Yes	COTS		Yes
Low mass		1.1 kg		Yes	1.4 kg		Yes	1.8 kg		Yes	1.2 kg		Yes
Wants	**WT**	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score
Broad spectrum for high and low frequency vibration	8	.008 - 109 Hz	2	16	0.03 (30 s) – 200	5	40	DC – 200	3	24	0.0083 (120 s) – 108	2	16
Large operating temperature range Ideal: -30 - > 150 C	4	-50 - 60C	3	12	-20 - 70	3	12	-20 - 70	3	12	-20 - 60	3	12
Sensitivity	6	754 V·s/m (±0.5%); 2.4 ng/√Hz @10 Hz	5	30	500 V·s/m; 145 dB DR; low noise	3	18	155 dB DR; noise < 1 ng/√Hz (est.)	4	24	753 V·s/m; 156 dB DR	5	30
Tilt tolerance	5	±2.5 (120 s) / ±10 (20 s)	3	15	±5	5	25	N/A	1	5	±2.5 / ±10	3	15
Can be deployed without direct contact with the environment	2	Yes		0	N/A		0	N/A		0	Yes		0
Storage	1	64 GB	5	5	64 GB	5	5	N/A	1	1	64 GB	5	5
Rapid data collection	3	N/A	1	3	1-1000 sps	5	15	N/A	1	3	N/A	1	3
Shock	7	100 g	5	35	N/A	1	7	100 g	5	35	100 g	5	35
Power consumption	5	180-120 mW	5	25	600 mW @ 7-16V	3	15	1,2000 mW @ 12V	2	10	180-120 mW	5	25
		Nominal		81	Nominal		115	Nominal		69	Nominal		81

	STRYDE Node Range+			MEMS Seismometer			GS-1 Seismometer			GeoSIG arolla		
Links	https://dqifccr9a141x.cloudfront.net/specsheet/The-STRYDE-Node_Range-Spe			https://microdevices.jpl.nasa.gov/capabilities/nano-and-micro-systems/			https://www.geospace.com/products/sensors/gs-1-seismometer/?utm_source=cha			https://www.geospace.com/products/sensors/gs-1-seismometer/?utm_so		
Must Haves	Info		Y/N	Info		Y/N	Info		Y/N	Info		Y/N
Commercial off the shelf (COTS)	COTS		Yes	In dev at JPL		No	COTS		Yes	COTS		Yes
Low mass	150 g		Yes	N/A		Yes	1.96kg		Yes	10kg		No
Wants	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score
Broad spectrum for high and low frequency vibration	1 – 125	1	8			0	1 - 100Hz	1	8	0.00833 Hz to 50 Hz	2	16
Large operating temperature range Ideal: -30 - > 150 C	-30 - 70	4	16			0	-40 - 60	4	16	-20 - 60	3	12
Sensitivity	22 ng/√Hz @16 dB	2	12			0	Hiugh sensitivity	4	24	1200V/m/s (±0.5%);	5	30
Tilt tolerance	N/A	1	5			0	±2	4	20	±2	4	20
Can be deployed without direct contact with the environment	N/A		0			0	No	0	0	No	0	0
Storage	64 GB	5	5			0	N/A	1	1	N/A	1	1
Rapid data collection	500-1000 sps	3	9			0	N/A	1	3	N/A	1	3
Shock	High-shock tolerant	3	21			0	N/A	1	7	810g	5	35
Power consumption	N/A	1	5			0	Passive	4	20	N/A	1	5
	Nominal		55	Nominal		0	Nominal		72	Nominal		82

Table 44. Seismometer Selection Trade Study

C.2.4 Thermal Probe Selection Trade Study

PERCI Instrument Trade Study: RTDs

Choices / Criteria	Germanium (GR Series)		Platinum (PT-111)		Cernox RTD		Silicon Diode (CY670 Series)			
Must Haves	Info	Y/N	Info	Y/N	Info	Y/N	Info	Y/N	Info	Y/N
Operating Temp Range (within a range of -223.15°C(50K) to -198.15°C (75K)in the body)	1.4K to 100K	Yes	14 K - 1123.15 K options available	Yes	0.1 K - 420 K options available	Yes	1.4 K - 500 K	Yes		
Commercial off the shelf (COTS) Options Available	COTS	Yes	COTS	Yes	COTS	Yes	COTS	Yes		
Low Power Operation (<10 mW)	10^-7 to 10^-6 W from 10K - 100K		10^-5 W from 10K - 100K	Yes	10-7 to 10-6 W from 10K - 100K	Yes	10 µW (10 µA excitation)	Yes		

Wants	WT	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score	Info	Value	Score
Temperature Errors (delta T / T) in Magnetic Field																0
Range of percent. Assigned Value: < 0.1 % 7-10 0.1% - 1.0 % 3-6 > 1.0 %. 1-2			1	1		4	4		10	10		6	6			0
	1	-3 <> -20 % (20K) (2.5 T)			0.04 % (87K) (2.5 T)			0.002 % (77K) (2.5 T)			-0.3 % (80K) (2 T) Junction perpindicular to field					0
																0
			5	30		4.5778	27.4669		4.8830	29.2980		0	0			0
Radiation Resilient (Largest delta T error up to 100K [magnitude])	6	17 mK			100 mK			40 mK			~1000 mK					0
																0
			4.6938	46.9387		5	50		4.7755	47.7551		0	0			0
Measurement Reproducibility (using 77 K)	10	±0.5 mK reproducibility at 4.2 K, +- 20mK at 77 K			High Reproducibility: +-5mK at 77K			±0.5 mK at 4.2 K, ±16 mK at 77 K			±0.25 K from 2 K to 100 K					0
																0
			0	0		0.8369	5.85872		4.6032	32.2229		5	35			0
Thermal Response (using 77 K)	7	Thermal response time 200 ms at 4.2 K, 3 s at 77 K			2.5 s at 77K			50ms - 1s at 77K based on packaging options (using 250ms SD packaging option)			BR Model: 1 ms @ 4.2 K, 13 ms @ 77 K,					0
											From 25 K to 500 K, a silicon diode has a nearly constant sensitivity of 2.3 mV/K					0
																0
Resitive Sensitivity	4															0
				0			0			0			0			0
				0			0			0			0			0
?	0			0			0			0			0			0
				0			0			0			0			0
				0			0			0			0			0
?	0			0			0			0			0			0
				0			0			0			0			0
				0			0			0			0			0
?	0			0			0			0			0			0
				0			0			0			0			0
				0			0			0			0			0
?	0			0			0			0			0			0
Total Score		Minimum		0			0			0			0			0
		Nominal		77.9387			87.3256			119.276			41			0
		Maximum		0			0			0			0			0

Table 45. Thermal Probe Selection Trade Study

C.2.5 Primary Cell Selection Trade Study

PERCI Primary Cell Selection Trade Study

Author: Sean McClary

Choices / Criteria	Consumer Grade LiFePO4		Defense-Grade Thermal Battery		Deep Space 2 LTC Battery		Traditional LiSOCL2 Battery		Nickel-Hydrogen Battery	
Must Haves	Info	Y/N	Info	Y/N	Info	Y/N	Info	Y/N	Info	Y/N
Cells must exceed 200Wh/L energy density	High, and rechargable	Yes	Very High	Yes	Medium	Yes	High	Yes	Barely meets criterion	Yes
Cells must possess minimal risk of thermal runaway or mitigate risk	Almost zero risk	Yes	Almost zero risk	Yes	Low	Yes	Low	Yes	Very Low	Yes
Total battery weight must not exceed 35kg	Meets requirement	Yes	Meets requirement	Yes	Meets requirement	Yes	Meets requirement	Yes	Meets requirement	Yes
Must survive at least 5,000g impact force	Meets requirement	Yes	Meets requirement	Yes	Meets requirement	Yes	Meets requirement	Yes	Meets requirement	Yes
Cells must survive 10+ years with minimal charge loss or mitigate	Low Charge Loss	Yes	No charge loss	Yes	Negligible charge loss	Yes	Negligible charge loss	Yes	Low charge loss	Yes

Wants	WT	Info (LiFePO4)	Value	Score	Info (Thermal)	Value	Score	Info (DS2 LTC)	Value	Score	Info (LiSOCL2)	Value	Score	Info (Ni-H2)	Value	Score
Volumetric Energy Density	0.25	LiFePO4 batteries have some of the highest energy densities of rechargable cells	0.7	0.175	Thermal batteries allow very high energy density due to rapid chemical reactions	0.95	0.2375	Moderate density, designed for impact survival and low-temperature operation	0.6	0.15	High density, optimized for compact deep-space storage	0.85	0.2125	Low density, limited by gas containment volume	0.1	0.025
			0.75	0.1875		0.98	0.245		0.65	0.1625		0.9	0.225		0.15	0.0375
			0.8	0.2		1	0.25		0.7	0.175		0.95	0.2375		0.2	0.05
Specific Energy (gravimetric density)	0.25	LFP specific energy suffers from the electrode weight and density, although is still high	0.45	0.1125	High specific energy, short-duration power	0.6	0.15	High specific energy	0.9	0.225	Very high specific energy, mass-efficient for long missions	0.85	0.2125	Low specific energy, bulky gas-based system	0.2	0.05
			0.5	0.125		0.65	0.1625		0.95	0.2375		0.9	0.225		0.25	0.0625
			0.55	0.1375		0.7	0.175		1	0.25		0.95	0.2375		0.3	0.075
Availability	0.1	Very available, used in automotive, aerospace, medical, consumer	0.95	0.095	Relatively common from suppliers in defense industry, in active production	0.8	0.08	Custom cell, used only for one mission and related testing. Remanufacture is a possibility	0.1	0.01	Widely available from aerospace and industrial suppliers, used in space, oil/gas, remote sensing, medical	0.85	0.085	Widely considered obsolete, legacy and government programs main clients	0.2	0.02
			0.98	0.098		0.85	0.085		0.15	0.015		0.875	0.0875		0.25	0.025
			1	0.1		0.9	0.09		0.2	0.02		0.9	0.09		0.3	0.03
Impact Tolerance	1	Off-the shelf components can survive 1000-5000g depending on model, but can be ruggedized	0.1	0.1	Depending on design, can handle 50,000-100,000g	0.95	0.95	Specifically designed to tolerate 80,000g+	0.99	0.99	Can possibly survive 20,000g depending on exact design, but unlikely without ruggedization	0.3	0.3	Contains high-pressure storage, can be ruggedized but is catastrophic in event of failure	0.15	0.15
			0.15	0.15		0.975	0.975		0.995	0.995		0.35	0.35		0.2	0.2
			0.2	0.2		1	1		1	1		0.4	0.4		0.25	0.25
Constant and Burst Current Capability	0.75	LFP cells are some of the most tolerant of extreme constant current	0.9	0.675	Can provide high current, but not over long, sustained periods	0.1	0.075	Moderate current capability, designed for stability over longer time periods	0.6	0.45	Ideal for low power draw over long periods, burst power for heater/TX/RX may be an issue	0.4	0.3	Excellent stability under low/moderate load, can tolerate burst power	0.85	0.6375
			0.95	0.7125		0.15	0.1125		0.65	0.4875		0.45	0.3375		0.9	0.675
			1	0.75		0.2	0.15		0.7	0.525		0.5	0.375		0.95	0.7125
Safety Risk	0.5	Thermal runaway is almost impossible, low/moderate risk of hazardous electrolye contamination if breached	0.8	0.4	Very stable; non-reactive until triggered. Generates high heat levels if accidentally triggered	0.6	0.3	Extremely stable under normal operation, small risk of thermal runaway if punctured, moderate chemical hazard	0.5	0.25	Highly corrosive and unstable if containment is breached, very small risk of thermal runaway, increased local hotspots	0.4	0.2	High pressure H2 gas can cause catastrophic unplanned disassembly in event of breach	0	0
			0.85	0.425		0.65	0.325		0.6	0.3		0.5	0.25		0.025	0.0125
			0.9	0.45		0.7	0.35		0.7	0.35		0.6	0.3		0.05	0.025
Long-term Stability / Self Discharge Rate (10 years)	0.9	Potentially 90% charge loss, but could be kept 'topped off' by orbiter during journey	0.1	0.09	Ideal for long storage, inert with no capacity loss until activated	0.95	0.855	Ultra-low self-discharge, designed to survive multi-year deep-space missions.	0.9	0.81	Excellent long-term storage, proven in space, depending on design less specialized than DS2 LTC	0.85	0.765	Estimated 60% loss over 10 years, would require orbiter battery tender	0.2	0.18
			0.15	0.135		0.975	0.8775		0.95	0.855		0.875	0.7875		0.25	0.225
			0.2	0.18		1	0.9		1	0.9		0.9	0.81		0.3	0.27
Nominal Voltage Stability	0.9	Large initial drop from maximum voltage, moderate 'knee' on discharge curve with sharp drop <20%	0.3	0.27	Moderately stable initial voltage, rapid drop-off as heat dissipates	0.2	0.18	Moderate initial drop, almost constant nominal voltage until <5% capacity		0	Very stable nominal voltage throughout discharge range, with a predictable dropoff <5%	0.75	0.675	Voltage highly dependent on load, stable voltage with stable load	0.15	0.135
			0.35	0.315		0.25	0.225			0		0.775	0.6975		0.2	0.18
			0.4	0.36		0.3	0.27			0		0.8	0.72		0.25	0.225
Scalability / Configurability	0.75	Highly configurable due to wide availability in multiple form factors	0.8	0.6	Generally custom sized, not easily scaled or reconfigured	0.2	0.15	Specicfically designed to be stable, ideally in multiple groups of 4 cells	0.9	0.675	Highly configurable and scalable, available in common standards and custom sizes	0.9	0.675	Configuration highly constained by pressure containment systems, but is configurable	0.35	0.2625
			0.9	0.675		0.25	0.1875		0.925	0.69375		0.95	0.7125		0.4	0.3
			1	0.75		0.3	0.225		0.95	0.7125		1	0.75		0.45	0.3375
Total Score		Minimum		2.5175			2.9775			3.56			3.425			1.46
		Nominal		2.823			3.195			3.74625			3.6725			1.7175
		Maximum		3.1275			3.41			3.9325			3.92			1.975

Table 46. Primary Cell Selection Trade Study

C.3 Risk Charts

C.3.1 Current Risk Matrix

Probability x Detectability	Severity: 1 Negligible	2 Low	3 Moderate	4 High	5 Catastrophic
1-5 Very Unlikely	POW--02 POW-05	ADG-05 CDH-04	SCI-04 SCI-08 POW-01 POW-04	THR-02 ADG-03 ADG-04 STR-06 CDH-03 CDH-06 SCI-03 SHD-03 COM-01 POW-03 COM-02 STR-05 SCI-05 STR-01 PRP-02	ADG-01 ADG-02 CST-04 SCI-01 SCI-09 PRP-03 POW-06 SYS-02
6-10 Unlikely			THR-03 STR-02 CST-01 THR-05 SCI-07	PRP-05 PRP-06 SHD-04 CST-05 CST-06 THR-04 PRP-04 STR-04	CDH-01 STR-03 SCI-02 CDH-05 SHD-02 SYS-01
11-15 Possible				THR-01	PRP-01 SCI-06 SHD-01
16-20 Likely					CST-03
21-25 Certain					

Table 47. Current Risk Matrix

C.3.2 Goal Risk Matrix

Probability x Detectability	Severity: 1 Negligible	2 Low	3 Moderate	4 High	5 Catastrophic
1-5 Very Unlikely	POW--02 POW-05	ADG-05 CDH-04	SCI-04 SCI-08 POW-01 POW-04 PRP-06 SHD-04 THR-03	THR-02 ADG-03 ADG-04 STR-06 CDH-03 CDH-06 SCI-03 SHD-03 COM-01 POW-03 COM-02 STR-05 SCI-05 STR-01 PRP-02	ADG-01 ADG-02 CST-04 SCI-01 SCI-09 STR-03 PRP-03 POW-06 SYS-02 SCI-02 CDH-05
6-10 Unlikely			STR-02 CST-01 THR-05 SCI-07	PRP-05 THR-01 CST-05 CST-06 THR-04 PRP-04 STR-04	CDH-01 PRP-01 SHD-02 SYS-01 SCI-06 SHD-01 CST-03
11-15 Possible					
16-20 Likely					
21-25 Certain					

Table 48. Target Risk Matrix

C.3.3 Top 4 Risks Mitigation Chart

FMEA Reference #	Part & Failure Mode	RPN Score	Mitigation Plan	Goal RPN
CST-03	Initial cost estimates inaccurate -> budget overruns	80	• Perform detailed cost estimations • Build a historical cost model • Track high-variance items and assumptions	45
SHD-01	Vendors fail to provide components on time -> schedule overrun	75	• Initiate early vendor communication • Identify potential delays early • Make scheduling issues more detectable	30
SCI-06	Instruments don't turn on -> faulty equipment/connections	60	• Conduct early integration testing • Use engineering models (thermal, vibration, electrical)	30
PRP-01	Impact trajectory compromised -> insufficient delta V	60	• Add margin to SRB delta-V capacity • Run higher-fidelity trajectory analysis • Refine mass budget to improve delta-V estimates	30

Table 49. Top Risks Mitigation Chart

C.3.4 Risk Management Plan

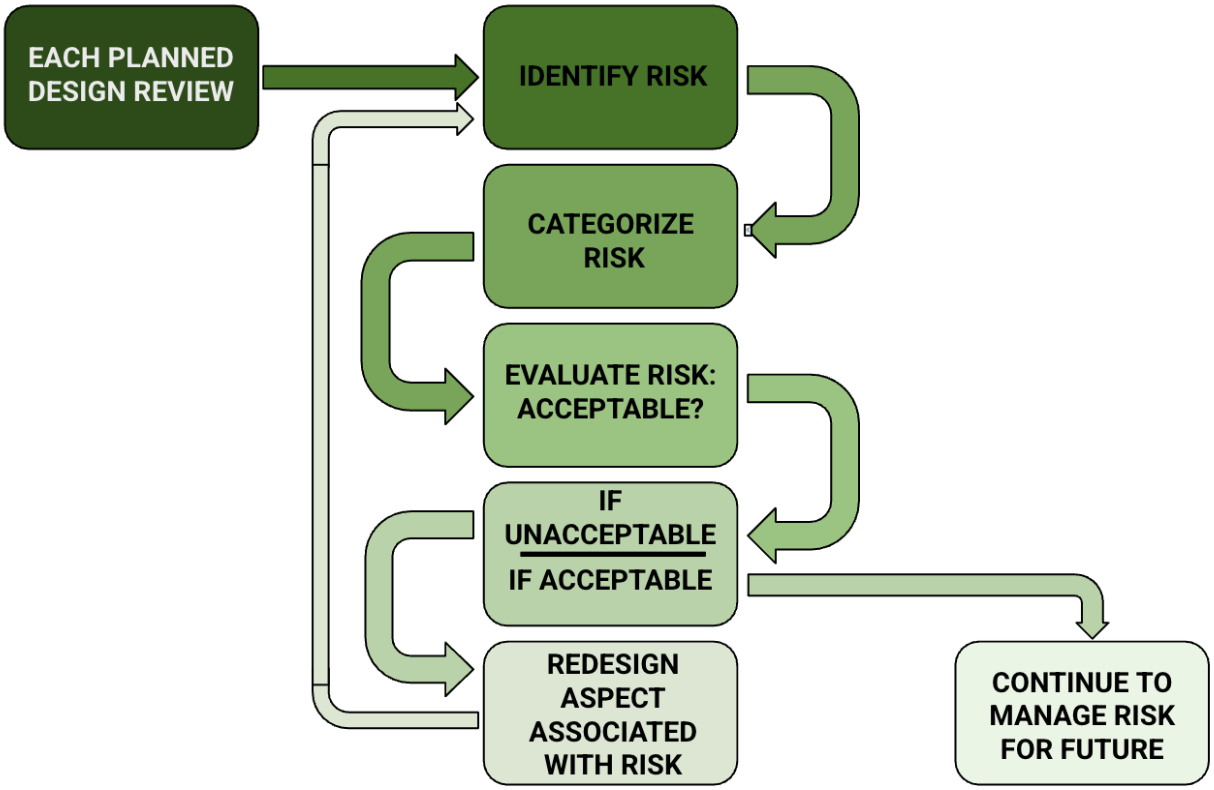

Figure 16: Risk Management Plan Flow Chart

C.4 Schedule

C.4.1 IMS Table

Task #	Subsytem	Task Title	Duration (Days)	Start	Finish	Weight (total PV)	% Complete	Milestone Deliverables
Milestone 1: Fall Final Presentation and Report			88	9/9/24	12/06	184	100.00%	STM, system level requirements, mission concept, budgets, risk analysis, final presentation, final report
1.2.1.1.2	Mission Definition	Decadal Survey Research / Prioritize and Define Science Objectives	18	9/9/2024	9/27/2024	30	100.00%	
1.2.1.1.3	Mission Definition	Cross-reference STM with UOP mission	7	9/27/2024	10/4/2024	56	100.00%	
1.2.1.5.3	Mission Definition	Ring-orbiter Mission Concept	30	10/1/2024	10/31/2024	60	100.00%	
1.2.1.5.4	Mission Definition	Trade mission concepts	10	10/31/2024	11/10/2024	30	100.00%	
1.4.6.1.1	Thermal	One Node Model, Active	10	11/8/2024	11/18/2024	4	100.00%	
1.4.6.2.1	Thermal	One Node Model, Passive	10	11/8/2024	11/18/2024	4	100.00%	
Milestone 2: Mission Concept Decision			36	1/20/2025	2/25/2025	634	100.00%	Final STM, results of additional analyses, trade study results
1.1.1.1.5	Systems	Trace functional requirements from customer needs	14	2/4/2025	2/18/2025	30	100.00%	
1.2.1.1.1	Mission Definition	STM Creation	162	9/9/2024	2/18/2025	120	100.00%	
1.2.1.1.4	Mission Definition	Create Functional Requirements	148	9/23/2024	2/18/2025	148	100.00%	
1.3.1.1.1	Payload and Instrumentation	Instrumentation Research (similar missions)	15	2/3/2025	2/18/2025	10	100.00%	
1.3.1.1.2	Payload and Instrumentation	Seismic Research	15	2/3/2025	2/18/2025	10	100.00%	
1.3.1.1.2	Payload and Instrumentation	Find Specific Instrumentation	14	2/4/2025	2/18/2025	20	100.00%	
1.3.1.3.1	Payload and Instrumentation	Temperature probe trade study	14	2/4/2025	2/18/2025	10	100.00%	
1.3.1.3.2	Payload and Instrumentation	Temperature research	14	2/4/2025	2/18/2025	10	100.00%	
1.3.1.4.2	Payload and Instrumentation	Surface composition research	14	2/4/2025	2/18/2025	15	100.00%	
1.3.1.4.3	Payload and Instrumentation	Accelerometer Trade study	14	2/4/2025	2/18/2025	10	100.00%	
1.4.1.1.1	Power	Find primary battery type	22	1/8/2025	1/30/25	176	100.00%	
1.4.1.1.2	Power	Find Secondary Battery type	22	1/8/2025	1/30/25	30	100.00%	
1.4.1.2.1	Power	Determine number of batteries	22	1/8/2025	1/30/25	44	100.00%	
1.4.1.2.2	Power	Determine size of batteries	22	1/8/2025	1/30/25	176	100.00%	
1.4.2.1.1	Propulsion	Delta V Requirements	3	2/3/2025	2/6/2025	24	100.00%	
1.4.2.1.2	Propulsion	Define Size Constraints	2	2/7/2025	2/9/2025	16	100.00%	
1.4.2.1.3	Propulsion	Define Mounting constraints	1	2/10/2025	2/11/2025	8	100.00%	
1.4.2.2.1	Propulsion	Choose Propulsive Method	4	1/6/2025	1/10/2025	32	100.00%	
1.4.2.2.2	Propulsion	Select Thruster	2	1/12/2025	1/14/2025	16	100.00%	
1.4.2.2.3	Propulsion	Source Thruster	1	1/14/2025	1/15/2025	8	100.00%	
1.4.4.1.1	ADCS	Trajectory Simulation Simulate trajectory of the penetrator.	11	2/1/2025	2/12/2025	16	100.00%	
1.4.4.2.1	ADCS	Trade studies on control hardware	2	2/1/2025	2/3/2025	16	100.00%	
1.4.6.1.2	Thermal	Multi-Node Model, Active	17	1/20/2025	2/6/2025	8	100.00%	
1.4.6.2.2	Thermal	Multi-Node Model, Passive	17	1/20/2025	2/6/2025	8	100.00%	
1.4.6.1.3	Thermal	Penetrator Model, Active	4	2/7/2025	2/11/2025	8	100.00%	
1.4.6.2.3	Thermal	Penetrator Model, Passive	4	2/7/2025	2/11/2025	8	100.00%	
1.4.6.1.4	Thermal	Orbiter Mounting Model, Active	14	2/4/2025	2/18/2025	12	100.00%	
1.4.6.2.4	Thermal	Orbiter Mounting Model, Passive	14	2/4/2025	2/18/2025	12	100.00%	
Milestone 3: Winter Midterm Presentation			19	2/26	3/17	359	100.00%	Final mission concept, midterm presentation
1.1.1.1.1	Systems	Create WBS Dictionary	6	2/22/2025	2/28/25	48	100.00%	
1.1.1.1.2	Systems	Conduct an Earned Value Analysis	1	2/27/2025	2/28/25	8	100.00%	
1.1.1.1.3	Systems	Create an Integrated Master Schedule (IMS)	1	2/27/2025	2/28/25	8	100.00%	
1.3.1.2.3	Payload and Instrumentation	Plume Research	27	2/4/2025	3/3/2025	60	100.00%	
1.4.3.2.1	CD&H	Design TX/RX Antenna	6	2/21/2025	2/27/25	48	100.00%	
1.4.3.1.1	CD&H	Complete a link budget	1	3/10/2025	3/11/2025	2	100.00%	
1.4.3.1.2	CD&H	Make an Updated Link Budget	1	3/11/2025	3/12/2025	4	100.00%	
1.4.5.1.3	Structures	Nosecone shape selection	31	2/1/2025	3/4/2025	30	100.00%	
1.4.5.3.2	Structures	Optimal Orientation and number of SRB's	1	3/3/2025	3/4/2025	6	100.00%	
1.4.5.1.2	Structures	Material Selection	15	2/28/2025	3/15/2025	120	100.00%	
Milestone 4: Winter Final Presentation and Report			27	03/18	04/14	1086	100.00%	Final presentation, final report
1.1.1.1.4	Systems	Develop a Risk Management Plan	3	3/22/2025	3/25/2025	90	100.00%	
1.2.1.2.4	Mission Definition	Impactor-focused Concept	168	10/1/2024	3/18/2025	60	100.00%	
1.2.1.2.2	Mission Definition	Develop multiple mission concepts	28	2/18/2025	3/18/2025	190	100.00%	
1.2.1.2.3	Mission Definition	Penetrator-focused Concept	28	2/18/2025	3/18/2025	60	100.00%	
1.3.1.2.2	Payload and Instrumentation	Seismometer trade study	42	2/4/2025	3/18/2025	10	100.00%	
1.3.1.4.1	Payload and Instrumentation	Multispectral camera trade study	42	2/4/2025	3/18/2025	15	100.00%	
1.4.5.3.4	Structures	CAD with SRBs	4	3/18/2025	3/22/2025	32	100.00%	

Table 50: PERCI’s In-Scope IMS

C.4.2 Future Work Task List

Task ID	Task Title	% Complete
1.1.1.1.6 Systems	Decompose functional requirements into subsystem requirements	50.00%
1.4.3.2.4 CD&H	Determine Polarization Scheme	66.67%
1.4.3.2.5 CD&H	Source Processor Components	50.00%
1.4.3.2.2 CD&H	RF Frontend	0.00%
1.4.3.1.3 CD&H	Define Processing Needs	0.00%
1.4.4.2.2 ADCS	Trade studies on pointing mechanisms	0.00%
1.4.5.3.1 Structures	SRB Jettison	0.00%
1.4.5.1.4 Structures	Impact/Shock Analysis	0.00%
1.4.6.1.5 Thermal	Moon Embedded Model, Active	0.00%
1.4.6.2.5 Thermal	Moon Embedded Model, Passive	0.00%
1.2.1.2.1 Mission Definition	Formulate Final Mission Concept	53.57%
1.4.1.2.3 Power	Design Battery Ring	17.50%
1.4.3.2.3 CD&H	Digital RF Transceiver	0.00%
1.4.4.2.3 ADCS	Control algorithm and simulations	0.00%
1.4.5.3.3 Structures	Vibration test and analysis	0.00%
1.4.5.3.6 Structures	Vibration analysis of launch	0.00%
1.4.5.2.1 Structures	Separation Mechanisms	5.00%
1.4.5.2.2 Structures	Optimal Orbiter Location	0.00%
1.4.5.2.3 Structures	CAD with Orbiter Interface	0.00%
1.4.5.1.1 Structures	Penetrator Design	40.00%
1.4.5.1.5 Structures	Impact Loading FEA	50.00%
1.4.5.1.6 Structures	Finalize CAD	0.00%

Table 51: Future Work Task List

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