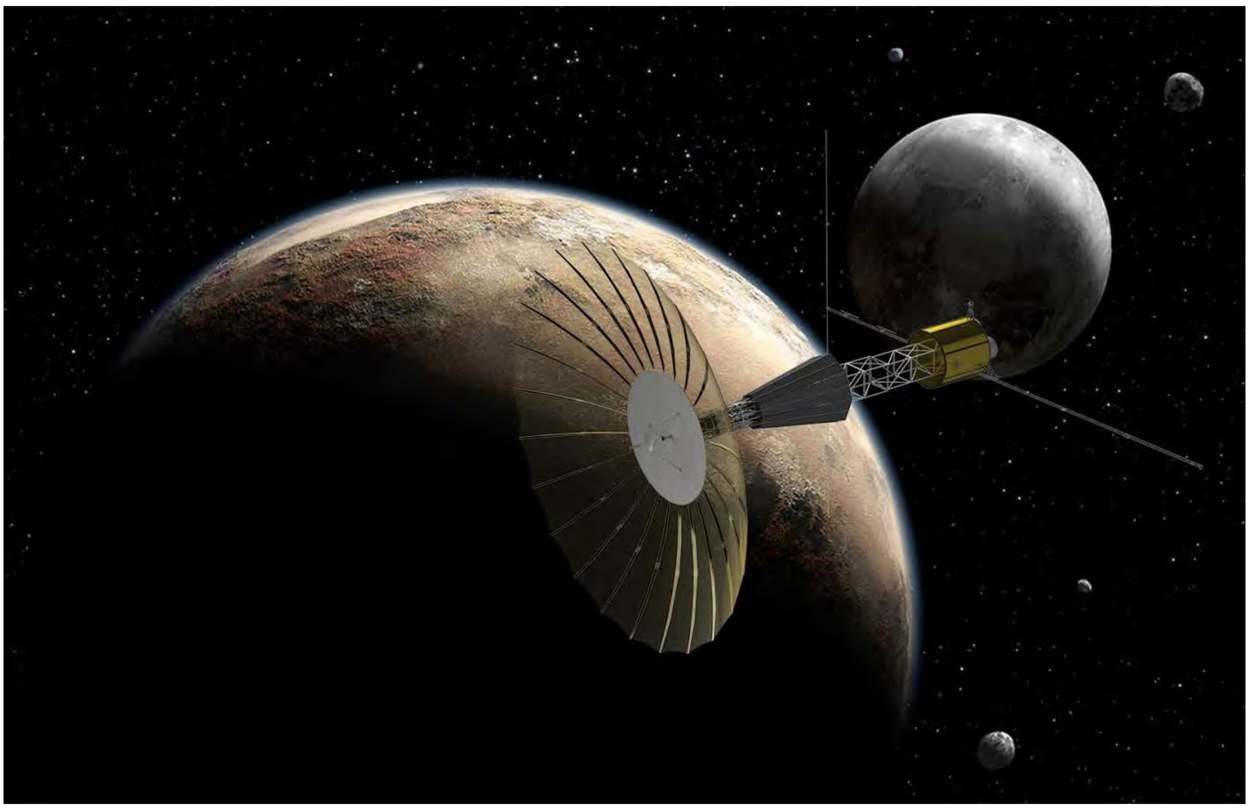
PROSERPINE

Pluto **R**esearch **O**rbiter **S**tudying **E**xperimental **R**ocket **P**ropulsion for **I**mproving trans-**N**eptunian **E**xploration



University of Kansas Team: AstroHawks

Jordan Alonzo   Frank Bonet III   Jack Cozzi   Taylor George   Miranda Myer

James Peters   Nathaniel Routh   Bradley Schroeder   Joseph Vincent   Luke Wehrkamp

University of Kansas Team: Supervisor

Dr. Mark Ewing

May 10, 2018

Acknowledgements

Firstly, the AstroHawks would like to express their gratitude to the people who were consulted and who provided invaluable help throughout the design process of PROSERPINE.

University of Kansas

- George Blake
- Dr. Steve Hawley
- Dr. Crag McLaughlin
- Bailey Miller
- Arno Prinsloo
- Alex Sizemore
- Sam Yang

Industry

- Michael Altenhofen (SpaceX)
- Tiffany Finley (Southwest Research Institute)
- Steve Mance (SpaceX)
- Dr. Jerry Manwiler (Fundamental Technologies)
- Adam Sexton (Ball Aerospace)
- Dr. Kelsi Singer (Southwest Research Institute)

Secondly, the AstroHawks would like to express their gratitude to the families, friends, and significant others who provided much-needed support throughout the design process of PROSERPINE. Thirdly, the AstroHawks would like to express their gratitude to Piper. Continuously brightening everyone's day, your puppy-dog eyes, generous amounts of kisses, and contagious energy were fundamental to the success of PROSERPINE.

University of Kansas Team: AstroHawks

Jordan Alonzo	Frank Bonet III	Jack Cozzi
922198	510671	665505
Taylor George	Miranda Myer	James Peters
665629	922531	922291
Nathaniel Routh	Bradley Schroeder	Joseph Vincent
922286	665621	922420
Luke Wehrkamp	Dr. Mark Ewing	
834256	100975	

Table of Contents

List of Figures

List of Tables

List of Symbols

AD&CS .... Attitude Determination and Control System
AIAA .... American Institute of Aeronautics and Astronautics
BWG .... Beam Waveguide
CCSDS .... Consultative Committee for Space Data Systems
C&DHS .... Command and Data Handling System
CER .... Cost Estimating Relationship
CM .... Center of Mass
CommOPs .... Communication Operations
DAEP .... Deep Space Network Aperture Enhancement Project
DART .... Double Asteroid Redirection Test
DSN .... Deep Space Network
e .... Electric
EIS .... Europa Imaging System
FGM .... Fluxgate Magnetometer
HGAS .... High Gain Antenna System
HGAX .... High Gain Antenna X-Band
HPSC .... High Performance Spaceflight Computing
Hz .... Hertz
I .... Second Moment of Inertia
IEM .... Integrated Electronics Module
IMU .... Inertial Measurement Unit
k .... Kilo

kg ... Kilogram
KRUSTY ... Kilopower Reactor Using Stirling Technology
LGH ... Low Gain Horn
LHC ... Left Hand Circular
LORRI ... Long Range Reconnaissance Imager
m ... Meter
MASPEX ... Mass Spectrometer for Planetary Exploration
MOC ... Mission Operations Center
MOps ... Mission Operations
MS ... Margin of Safety
N ... Newton
NASA ... National Aeronautics and Space Administration
NEP ... Nuclear Electric Propulsion
NEXT-C ... NASA Evolutional Xenon Thruster (Commercial)
NICM ... NASA Instrument Cost Model
NSTAR ... NASA Solar Technology Application Readiness
NTP ... Nuclear Thermal Propulsion
PMS ... Propellant Management System
PPS ... Power Processing System
PPU ... Power Processing Unit
POSSE ... Pluto and Outer Solar System Explorer
RDT&E ... Research, Development, Test, and Evaluation
REASON ... Radar for Europa Assessment and Sounding: Ocean to Near-Surface
REP ... Radioisotope Electric Propulsion
RF ... Radio Frequency
RFP ... Request for Proposal
RHC ... Right Hand Circular
ROIC ... Read-Out Integrated Circuit
RTG ... Radioisotope Thermoelectric Generator
RTP ... Radioisotope Thermal Propulsion
s ... Second
SEP ... Solar Electric Propulsion
SFCG ... Space Frequency Coordination Group
SNAP ... Space Nuclear Auxiliary Power
SOC ... Science Operations Center
SPDT ... Single-Pole Double-Throw
SRU ... Shunt Regulator Unit
SSPA ... Solid-State Power Amplifiers
SSR ... Solid State Recorders
SWIA ... Solar Wind Ion Analyzer
SWAS ... Submillimeter Wave Array Spectrometer
TFU ... Theoretical First Unit
TRL ... Technology Readiness Level
TWTA ... Traveling Tube Wave Amplifiers
USO ... Ultra-Oscillator
UVS ... Ultraviolet Imaging Spectrometer/Spectrograph
ΔV ... Velocity Change
VMC ... Vehicle Management Computer

V/SHM .... Vector/Scalar Helium Magnetometer
W .... Watt
WBS .... Work Breakdown Structure
CER .... Cost Estimating Relationships
X .... X-Coordinate/Direction
Y .... Y-Coordinate/Direction
Z........ Z-Coordinate/Direction
ZBO .... Zero Boil-Off

1. MISSION DESCRIPTION

The following is a description of the request for proposal (RFP) solicited by the American Institute of Aeronautics and Astronautics (AIAA) for the 2017-2018 Undergraduate Team Space Transportation Design Competition [1]. The objective of this RFP was to propose an orbiting mission of the Pluto-Charon System with a spacecraft minimum nominal instrument load matching the New Horizons spacecraft [1].

The New Horizons mission completed a flyby of the Pluto-Charon system in 2015, information from which has shown Pluto to be much more geologically active than anticipated, as well as an incredibly dynamic Pluto-Charon system. Further characterization of atmosphere and geology of Pluto as well as system dynamics may be scientifically valuable. An orbiting mission is necessary to gather sufficient data for these goals, and the proposal outlines such an orbiting mission. Technical details of the mission architecture as well as major design elements must be addressed. Also documented should be major design decisions and characteristics of the design, as well as risks and challenges associated with the proposed design.

Explicit constraints are as follows, New Horizons instrumentation will serve as a baseline for the proposed orbiter with alterations or substitutions to the instrument load, justified on scientifically justified; the mission must be limited to a single launch; in-space propulsion systems must be TRL 6 or higher at launch; the primary mission length is limited to 25 years with at least one year in orbit at Pluto. Additional considerations include scientific merit, mission economics, reliability, affordability, and extended missions. Proposed missions with longer orbit times at Pluto are preferred. The proposed design will be multidisciplinary in nature, requiring the expertise of many disciplines within aerospace.

2. EXECUTIVE SUMMARY

The Pluto Research Orbiter Studying Experimental Rocket Propulsion for Improving trans-Neptunian Exploration (PROSERPINE), shown in Figure 1, will start the journey to Pluto by launching on an Atlas V rocket which will give the spacecraft the needed C3 of 144 km^2/s^2 to reach its destination. The proposed launch window will be from February 17 to March 21, 2031, as this allows Jupiter to be used for a flyby. The trajectory utilized is a simple Earth to Pluto flight path which includes a gravity assist at Jupiter to decrease the overall time of flight. To round the trip off and allow PROSERPINE to achieve an orbit about Pluto, a deceleration with respect to the Sun is done by utilizing multiple ion propulsion engines to provide the required reverse thrust. The calculated trajectory has a total time of flight of 18.3 years, well within the allotted 25 year timeline.

Upon arrival in the Pluto-Charon system, PROSERPINE will perform a plane change to enter a polar mapping orbit of the system. Once science objectives in the close vicinity of Pluto have been achieved, the spacecraft will raise its orbit to allow for successive flybys of the various moons.

PROSERPINE uses a nuclear-electric propulsion system for deceleration and mission operations at Pluto. The spacecraft will use three NASA Evolutionary Xenon Thruster (NEXT) units with flight heritage tankage. Carrying 1350 kg of propellant with a dry mass of 4083 kg, PROSERPINE has over 16.5 km/s of ΔV.

Electrical power for all PROSERPINE systems will be supplied by a NASA Kilopower Reactor sized to output 8.5 kWe. This value was determined to be the power required to operate the spacecraft systems during any stage of the mission, with a 10% margin of safety (MS). The reactor mass is 1,390 kg.

The power system uses three solid-state lithium-ion (Li-Ion) batteries, to provide power to the spacecraft subsystems during launch and Earth departure, which have a dry mass of 16.3 kg. The battery system was sized for a total load of 1.5 kW. The solid-state Li-Ion batteries are divided into three different banks. Battery banks were sized such that in the event of a single bank failure, the remaining units will continue to provide subsystems with the necessary power. The complete mass and power budget are tabulated in Table I.

The PROSERPINE spacecraft will utilize passive thermal methods to both dissipate and contain heat generated throughout the mission. The thermal methods used on PROSERPINE are designed for unmanned spacecraft, which tend to use less electrical power, are more cost effective, and do not use up as much of the mass budget as active control subsystems. The thermal control system used incorporates carbon fiber radiators, Haynes 230 alloy/Sodium heat pipes, and Kapton-Mylar multilayer insulation.

The command and data handling system (C&DHS) of PROSERPINE will be contained within two Integrated Electronics Modules (IEM). The two IEMs are identical, with the second only used in the case of primary IEM failure. Each IEM will house the three command and data processors, multiple solid state recorders, and an ultra-stable oscillator. The processors will be Boeing Chiplet processors which are dual quad-core processors under development for the High Performance Spaceflight Computing NASA program. The solid state recorders will be from Southwest Research Institute which are radiation hardened and allow direct downlink in Consultative Committee for Space Data Systems (CCSDS) format to the telecommunications system. Science instrument data will be recorded, compressed, and encoded before transmission back to Earth. Additionally, functionality to rewrite onboard software will be included. This functionality allowed the New Horizons spacecraft team to revive the system after a failure and was part of what helped make the mission a success.

Table I: Mass and Power Budget

Component	Mass (kg)	Max. Power (W)
Science Payload	87	141
Structure & Mechanisms	595	10
Propulsion	201	7250
Power & Reactor	1424	36
Thermal Control	61	-
AD&CS	63	134
C&DHS	43	26
Cabling	91	-
Telecommunications	168	6000
On-Orbit Dry Mass	2733	-
Propellant	1350	-
On-Orbit Wet Mass	4083	-
Launch Vehicle Adaptor	359	-
Total Launch Mass	4442	-

The RF telecommunications system will utilize X-band for primary uplink and downlink with a redundant S-band system. Communications will feature a 12.6-m deployable parabolic antenna with a 4.2-m solid center section capable of uplink and downlink duties in the event that the antenna does not deploy. It will also use an S-band low gain horn for near Earth communications. With the high-gain antenna deployed, the PROSERPINE will be capable of 16900 kbps downlink and 144 kbps uplink upon arrival at Pluto.

The attitude determination and control system (AD&CS) aboard PROSERPINE determines attitude through sun trackers, star trackers, and inertial measurement units (IMUs). The orientation of the spacecraft is controlled primarily through an assembly of four reaction wheels. Momentum dumping and supplemental attitude control will be provided by the primary propulsion system and six additional ion thrusters. This type of system is sized to perform all intended maneuvers and counteract expected external and internal disturbances.

The parametric cost estimating method was used to approximate the cost of PROSERPINE. The total cost of the mission including research and development, integration to the launch vehicle, 22 years of transit, and one year of orbiting Pluto is $3.3 Billion (FY18$).

PROSERPINE will continue to be upgraded as the launch date approaches. This includes the possibility of improved scientific instruments and additional science payloads. If the spacecraft mass is held sufficiently low, this may include a Compact, Low-Yield Dwarf Explorer (CLYDE) impactor probe to obtain close images of the surface and expose the subsurface terrain to subsequent orbital imaging.

At the end of the mission life, PROSERPINE will meet a similar fate. All scientific data will downlink to Earth, at which point the spacecraft will use its remaining propellant to put itself on a collision course with Pluto or Charon. PROSERPINE will then transmit high-resolution data of the body as it approaches the surface, providing useful scientific information till the very end.

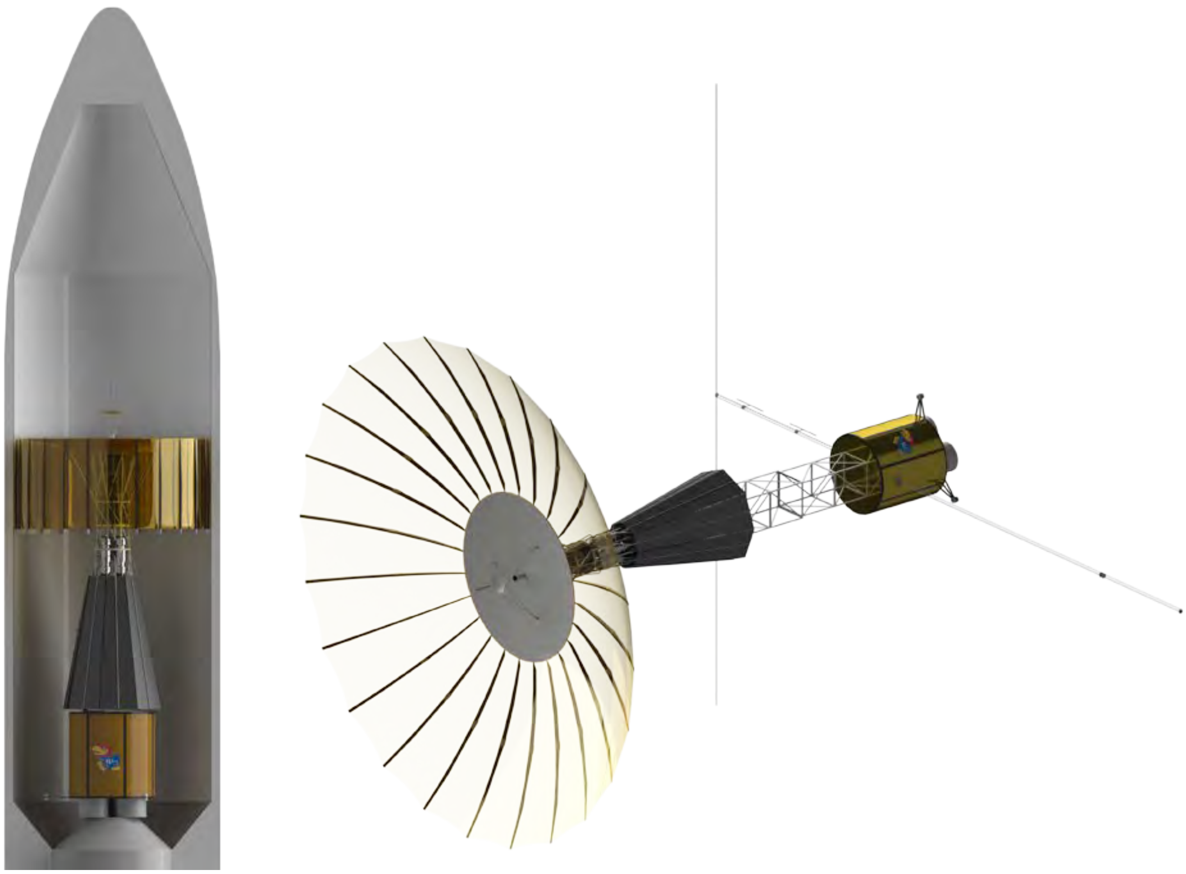


Figure 1: Rendering of PROSERPINE within Atlas V 552 and Deployed

3. HISTORY

“Between Jupiter and Mars, I place a planet.”

- Johannes Kepler, *Mysterium Cosmographicum*

Since antiquity, stargazers knew of five planets which wandered the night sky independently of the constellations. After the invention of the telescope, it became clear that these were finite bodies rather than point sources of light. Planets were permanently distinguished from stars, which led to significant cosmological turmoil.

By 1783, however, the implications of the heliocentric Solar System had largely been accepted. On March 13th, 1783, the British astronomers William and Caroline Herschel spotted the planet Uranus while measuring stellar parallax. The discovery was an accident, and initially Uranus was mistaken for a large comet. Soon, however, astronomers realized that it was a planet in its own right.

On the first day of 1801, another planet was discovered by Giuseppe Piazzi of Sicily. The moving, star-like object was eventually named Ceres and three other planets were soon discovered: Pallas, Juno, and Vesta. For many years there were eleven planets in the Solar System. In 1845, however, Karl Ludwig Hencke of Germany discovered a new body, and dozens more were spotted in the following years. Astronomers soon reclassified these objects as asteroids to avoid adding hundreds of planets to the Solar System.

Uranus and Ceres are visible in the night sky under the correct conditions but are so dim that their motion was overlooked by early astronomers. After their discovery, the orbits were mapped in considerable detail. With time, astronomers noticed deviations from Uranus’ predicted position. John Couch Adams in Britain and Urbain Le Verrier in France independently concluded that these discrepancies could be explained by the existence of another planet. German astronomer Johann Gottfried Galle spotted Uranus at the location that Le Verrier predicted in September of 1846, with less than an hour of searching.

As time went on, astronomers noticed that Neptune did not explain all of the discrepancies in the orbit of Uranus. Many concluded that this implied another planet beyond Neptune, including Percival Lowell of canal fame. Lowell spent the last years of his life searching for Planet X and died unsuccessful in 1922. After a lengthy estate battle, the Lowell Observatory hired autodidact astronomer Clyde Tombaugh in 1929 to continue the search. Tombaugh spotted Pluto near one of the locations which Lowell had predicted on the night of February 18, 1930.

The object was near the ecliptic, but this proved to be a coincidence. Tombaugh continued searching the ecliptic until he was satisfied that no comparable trans-Neptunian existed. The orbit of Pluto has a 17° inclination and

248-year period with eccentricity 248, but the characteristics of the object itself were difficult to determine. Assuming a low albedo, the mass is comparable to the mass of Earth. Lowell, however, had predicted seven Earth masses.

As the 20th Century progressed, mass estimates drifted downwards. In the 1970s, astronomers discovered methane ice, implying a high albedo. Soon thereafter, John Christy at the US Naval Observatory identified Charon, the largest moon of Pluto. This settled the question of mass of Pluto at 0.2% that of Earth. Some astronomers resumed searching for Planet X after the discovery of Charon, without success. The perturbations on Uranus were finally explained after Voyager 2 flew by Neptune in 1989, enabling a small but sufficient revision in the mass estimate that eliminated any unexpected perturbations.

In the following decades, numerous small bodies were discovered outside the orbit of Neptune in what is now known as the Kuiper Belt. After the turn of the century, several increasingly large Kuiper Belt Objects put the planetary status of Pluto in question. This culminated in 2005 when Eris was estimated to have a mass greater than that of Pluto. Some argued that Eris should be considered a planet, while other believed that neither should be.

Following the precedent set by the asteroids, the International Astronomical Union laid out a formal definition of planet which excluded Pluto. It, along with Ceres and Eris, would be considered dwarf planets while similarly-sized objects not orbiting the Sun in debris fields would be considered planets in their own right. Additionally, dwarf planets in orbital resonance with Neptune would be known as Plutoids, after the prototypical member. Many have criticized these definitions, but few better alternatives have been presented.

Pluto flybys were first seriously considered as a secondary target for the Voyager missions. However, the trajectory needed was incompatible with the more valuable flybys of Titan, Uranus, and Neptune. Another mission to observe Pluto would be necessary. In the 1990s, NASA studied a fast flyby mission known as the Pluto-Kuiper Express. This mission was cancelled, however, after budgetary overruns.

Several other missions were studied by NASA and industry teams during the same time. A large-scale Mariner Mark II concept was proposed, and competed with a much smaller Pluto 350 concept, so named based on its expected mass budget. Neither of these concepts were approved due to high perceived risks and costs. In 2001, NASA resumed study of a Pluto mission, soon narrowing the proposals down to New Horizons and the Pluto and Outer Solar System Explorer (POSSE). New Horizons won and, after an administrative budgetary battle, was launched in 2006.

Constructed by John Hopkins University Applied Physics Laboratory and the Southwest Research Institute, New Horizons lifted off aboard an Atlas V 551 on January 19, 2006. After a flyby of Jupiter in late February of 2007, the spacecraft cruised until January 2015. At that point, flight controllers activated the spacecraft and began taking readings of the Pluto-Charon system.

Closest approach occurred on July 14, 2015 as the spacecraft blazed through the system at nearly 14 km/s, passing within 12,500 km of Pluto and 28,800 km of Charon. New Horizons remained in radio silence during the flyby to maximize scientific observations and only re-aligned its dish with Earth on departure. The spacecraft imaged the bodies of the system in unprecedented detail, though in the cases of the minor moons, this remained mere splotches of color. Furthermore, the spacecraft was only able to image one side of Pluto and Charon in significant detail, as neither body rotates rapidly enough to be entirely visible over such short durations.

Following its departure, New Horizons began downlinking the massive quantity of data gathered during the flyby, a process which took over a year to complete. The spacecraft then adjusted its trajectory to ensure a flyby of Kuiper Belt object 2015 MU69 on January 1, 2019. During its extended flight through the Kuiper Belt, New Horizons will also use its long-range imager to map distant objects with improved accuracy.

No follow-up mission to New Horizons is currently planned. An orbit is the next logical choice in the general sequence for planetary exploration. PROSERPINE proposes to fulfill this role, entering orbit around the Pluto-Charon system and mapping the various bodies in great detail.

4. SCIENTIFIC OBJECTIVES AND INSTRUMENTS

Within this chapter, the scientific objectives and instruments for PROSERPINE will be discussed.

4.1 Downselection

The selection of the scientific areas of interest was conducted by first looking at those of New Horizons to get a base line of what PROSERPINE should include. Next, individuals from the New Horizons team were consulted. Tiffany Finley and Dr. Kelsi Singer from the Southwest Research Institute, both of whom were on the New Horizons team, were asked to discuss which instruments they thought would be most beneficial to include on an orbiting mission to Pluto. For example, the consulted individuals stated, knowing what they know now, they would have included an ice penetrating radar. The combination of questions left unanswered by the flyby of New Horizons served as the foundation that evolved into the selection of scientific objectives and thus the instruments required to complete these objectives.

Once a list of possible objectives was created, a downselection process was conducted to finalize the scientific objectives of PROSERPINE. The objectives that clearly satisfied the RFP were weighted more heavily than others [1]. Additional metrics such as data collected over the duration of the mission, level of preceding scientific research, and professional opinion were weighted as well. Time, structural, and physical complexity of instruments required to complete each scientific objective was also considered in the objective selection process. The final objectives are outlined in the following section.

4.2 Scientific Objectives

The five main scientific focuses for PROSERPINE are:

- Terrain of Pluto
- Atmosphere of Pluto
- Orbital Mechanics of Pluto-Charon System
- Moons of Pluto
- Planetary Flybys en Route to Pluto

4.2.1 Terrain of Pluto

The instruments that New Horizons carried to complete terrain related science objectives included the Long Range Reconnaissance Imager (LORRI) and Ralph, a visible and infrared imager. The high definition imagery of

LORRI allowed for the determination of number and size of craters. In addition, the instrument looked for activity such as geysers [2]. Discoveries made by Ralph included the characterization of the global geology and morphology of Pluto as well as mapping of the surface composition [3]. Although New Horizons was able to vastly expand the knowledge of terrain of Pluto, the spacecraft data led to more questions that have been left unanswered. Many observations of terrain of Pluto point to mysteries under the surface.

Mountains in the eastern half of Tombaugh Regio, better known as the 'Heart', are made of water ice. Planetary geologists theorize that these mountains were created by a mantle of water ice that has burst through the surface. This theoretical water ice mantle is presumed to be located under frozen nitrogen crust of Pluto [4]. The creation of these mountains has led to another interesting feature in the region. Fragments of these ice mountains have broken away and become floating hills. Because water ice is less dense than the surrounding nitrogen ice, these hills float and move like icebergs [5].

The interesting geological features of Tombaugh Regio are not just confined to the eastern half but appear in the west as well. Hexagonal patterns crisscross the western portion of the Regio, indicating fluid transfer of heat. This convection process has caused the plain of frozen nitrogen to slowly, constantly churn over millions of years. The reason for temperatures greater than 40 Kelvin to produce this type of behavior has not yet been discovered. The constant churning combined with the lack of impact craters in the area suggest that Tombaugh Regio is young compared to the rest of the planet. Decay processes or residual heat from the formation of Pluto is currently the best theory [4].

Looking into why the Regio did not simply freeze long ago adds to the mysteries as to what could be lying underneath the surface, including an underground ocean of liquid water. One theory as to why the Regio is not fully frozen is due to a massive impact in the early stages of life of Pluto that never healed. The additional weight of the nitrogen snow that covers the planet depressed the basin, which also lies in the most shaded part of the planet due to Pluto and Charon being tidally locked. For this scenario to work, Pluto must have had a liquid water ocean underneath the crust. This, in combination, with Pluto being relatively warm, geologists have concluded there may be a layer of liquid water under the surface today [4].

Additional evidence in the terrain of Pluto points to the possibility of an underground ocean. Photographs of the surface show tectonic evidence of the crust stretching and breaking. Crevasses indicate a slow freezing of a subsurface water ocean, causing the crust to expand. Had the ocean froze entirely, models indicate the ice structure should have been transformed from Ice I to the 30% denser Ice II. Because Ice II has not yet formed, the ocean has not completely frozen [6].

Another mysterious terrain feature on Pluto is the possibility of cryovolcanoes, better known as ice volcanoes. Wright Mons and Piccard Mons are two large mountains on Pluto. Both have a single crater in their center, indicating they may be volcanos. Nitrogen might be warmed by the interior of the planet, allowing it to exist in its liquid state. This nitrogen would then erupt back to the surface through the cryovolcanoes where it ultimately would freeze [6]. Wright and Piccard Mons could potentially be a part of a larger system of volcanos that was beyond the field of view of New Horizons [7].

To satisfy this specific scientific objective, the instrument payload for PROSERPINE will be capable of:

- Confirming the existence of and mapping the underground water ice mantle.
- Discovering the cause of internal warming.
- Confirming the existence of and mapping the underground liquid water ocean.
- Confirming the existence of and mapping locations of cryovolcanoes.
- Mapping and determining the movement patterns of floating water ice hills.

4.2.2 Atmosphere of Pluto

The discoveries of New Horizons gave the world a new perspective on the way the atmosphere of Pluto was viewed. The instruments aboard the spacecraft used to complete atmospheric scientific objectives were ALICE which determined atmospheric composition and structure, Solar Wind at Pluto (SWAP) which was used for solar wind imagery and atmospheric escape, Pluto Energetic Particle Spectrometer Science Investigation (PEPSSI) which was an energetic particle spectrometer, and REX which measured atmospheric temperature and composition [8]. These instruments measured a variety of never before seen phenomena that opened up the gateway to new questions for the future.

It was found that Pluto has a nitrogen rich atmosphere that extends a thousand miles above the surface, greater than predicted. The atmosphere was not degraded as severely as had been previously thought, which suggests something is feeding the atmosphere as the atmospheric escape should have deteriorated the atmosphere to a further extent. The atmospheric escape rate was also determined to have been lower than expected, though this alone does not account for all of the retained atmosphere. The temperature was determined to be colder than previous estimations by 70 degrees Fahrenheit [9].

Other discoveries include two layers of haze above the surface of Pluto about 30 miles and 50 miles high respectively [10]. This is what is responsible for the reddish hue of the body. Another discovery was an unexpectedly low surface pressure which is possible evidence of past changes to the pressure of the atmosphere [9]. Possible changes could involve the atmosphere beginning to freeze to the surface as the body moves further away from the sun. An "ion tail" was also discovered to have been trailing behind Pluto, however the details of how and what exactly the composition is remain unknown.

An area of interest that emerged from the New Horizons flyby which PROSERPINE will solve is possibility of clouds on Pluto. While orbiting Pluto, the overall structure, composition, and dynamics will be studied which could definitely answer this question along with the question of what is feeding atmosphere of Pluto. The ion tail trailing behind Pluto will be examined to determine composition and geometry as well. Further detailing of the atmospheric thermal properties as the body moves further away from the sun will also be considered on the mission. New Horizons did not include a magnetometer because of structural difficulty and budget concerns. For this reason, detailed mapping of the magnetosphere was not obtained by the flyby. The interplanetary magnetic field will thus be investigated during the orbit with inclusion of a magnetometer.

To satisfy this specific scientific objective, the instrument payload for PROSERPINE will be capable of:

- Determining the structure and dynamics of the atmosphere of Pluto throughout the duration of the orbit.
- Measuring the atmospheric thermal properties of Pluto throughout the duration of the orbit.
- Mapping the magnetosphere of Pluto.
- Determining the composition and cause of the ion tail that is being emitted by Pluto.

4.2.3 Orbital Mechanics of Pluto-Charon System

Mark Showalter, one of the co-investigators on the New Horizons spacecraft mission, had the following remarks on the interaction of the bodies of the Pluto-Charon system, "The way I would describe the system is not just chaos, but pandemonium" [11]. This chaotic behavior has not been observed in any other satellite system. Answers to unexplained anomalies, such as this, are fundamental in answering larger questions about origins the entire solar system and universe.

Most satellites within the solar system are in synchronous rotation with their central body. However, this is not true for the orbiting bodies within the Pluto-Charon system. For example, Hydra, the furthest known satellite of Pluto, rotates 89 times per single revolution around Pluto [12]. An increase in this rotational velocity could cause material from the moon to be ejected due to centrifugal force. This behavior is speculated to be the result of gravitational effects of the binary system [12]. However, more evidence would be needed to provide sufficient credibility to this theory. Scientists speculate that the system operates peculiarly because of a merger between two or more of the moons within the system.

If it were discovered that there were more satellites of the Pluto-Charon system, it would be beneficial to define their orbits about the system. This would help further understanding of the physical properties of the system as a whole. The last moon was discovered in 2013 by New Horizons with only a flyby [12]. An entire sweep of the sphere of influence of the system, provided by PROSERPINE, would determine within a high level of confidence whether or not there are more celestial bodies.

4.2.4 Moons of Pluto

Prior to New Horizons, little was known of Pluto. It was originally thought that the moons of Pluto were frozen, cratered rocks. However, after New Horizons, detailed images of Charon and the other moons show complex bodies that should be investigated further. Scientists determined that Charon is covered in deep fractures and canyons, as well as smooth plains. Some canyons are estimated to be up to nine kilometers deep. There is also a surprising lack of craters on Charon. In the southern hemisphere, there is a relatively smooth area that is theorized to be the result of geological processes [13].

New Horizons also led to discoveries regarding the smaller moons of Pluto, especially the formation of the smaller moons. It was initially assumed that all the moons would be of similar composition. However, after New Horizons, it was found that Styx, Nix, and Hydra were brighter than expected indicating an icy surface while Kerberos was much darker than expected. Furthermore, it was found that all the smaller satellites of Pluto are ellipsoids. The albedo of the smaller moons indicate that they are possibly icy bodies [14].

While New Horizons provided new insights to the moons of Pluto, it also gave rise to new questions. Further investigations should be performed into the geological processes acting on Charon. Improved surface imaging of the moons of Pluto could provide insight into the formation of the moons as well as any resurfacing events that occur. Improved surface imaging will also help confirm the presence of water ice on the smaller moons.

Overall, New Horizons answered many questions yet also raised more. Scientific instruments chosen for the purpose of confirming these hypotheses. To satisfy this specific scientific objective, the instrument payload for PROSERPINE will be capable of:

- Mapping the moons of Pluto.
- Determining the geological mechanisms of the moons.
- Determining the origin of the moons.
- Confirming water ice.

4.2.5 Planetary Flybys en Route to Pluto

The two primary trajectory types are a direct flight, or a flight which utilizes planetary flybys. In the case of a direct trajectory, the flight typically takes a long time and the spacecraft systems are in a hibernation state until it reaches its destination. With the use of planetary flybys and gravity assists, the spacecraft can build up speed and considerably shorten the flight time. Because of a mission duration limit of 25 years and at least a one year orbit of the Pluto - Charon system, PROSERPINE will be utilizing a planetary flyby of Jupiter. Also, the spacecraft can use the flyby planet as a staging area to test any of the instruments on board. This would help to ensure the instruments are working, and do not have to stay in a hibernation state for a long period of time.

New Horizons utilized a Jupiter gravity assist on its way to Pluto. By using Jupiter for a gravity assist, New Horizons reduced the time of flight to Pluto by up to three years [15]. The time New Horizons spent on the Jupiter gravity assist allowed for the instruments to be tested, and investigations were performed to answer the most pressing scientific questions about the largest planet in our solar system [16]. Using the on-board remote sensing instruments, New Horizons was able to collect data on the cloud structure, composition, and capture the first ever close-up images of Jupiter Red Spot. New Horizons was also able to look at how the magnetosphere of Pluto caused fluxes of charged particles to create the immense auroras at the poles of Jupiter [17].

The data and images NASA was able to collect at Jupiter, from New Horizons, show how detailed of an analysis can be achieved with a flyby. Based on conversations with members of the New Horizons team, PROSERPINE will investigate the atmospheric conditions of the Jovian system, and the magnetosphere around Jupiter, and perform additional surface imaging of Jupiter. The data and images collected at the Jovian system will show what the system looks like when PROSERPINE passes Jupiter. From the flyby of Jupiter by PROSERPINE, observations can be made about how the system has changed over time. In the atmosphere, one point of interest to be focused on is the interaction of particles with the magnetosphere of Jupiter, which causes extremely intense auroras. Also, when the imagers capture photos of the atmosphere of Pluto, an observation should be made on the size of red spot of Jupiter, because it is shrinking at an incredible rate and is predicted to be gone within 20 years [18]. The imagers on PROSERPINE will also be utilized to collect images of the changes of the atmosphere of Jupiter. REASON will be used to collect data on the possibility of an underground ocean on one of the moons of Jupiter.

4.3 Science Instruments

Table II gives a summary of the specifications for the instruments selected for PROSERPINE.

Table II: Instruments and Specifications

Instrument	TRL	Mass (kg)	Operational Power Intake (W)	Bit Rate (bps)
Europa Imaging System (EIS)	8	5.3	5.8	1.60×10^6
Ice Penetrating Radar (Reason)	8	32.2	55.0	3.20×10^5
Submillimeter Wave Array Spectrometer (SWAS)	5	7.00	12.0	2.33×10^6
Magnetometer (FGM)	9	5.00	6.50	2.41
Magnetometer (V/SHM)	9	7.10	6.00	2.41
Solar Wind Ion Analyzer (SWIA)	9	2.62	1.75	7.04×10^2
Mass Spectrometer for Planetary Exploration (MASPEX)	5	11.7	35.4	1.40×10^4
Ultraviolet Imaging Spectrometer/Spectrograph (UVS)	9	4.40	4.40	2.80
Europa Thermal Emission Imaging System (E-THEMIS)	8	11.2	14.0	1.00×10^6

4.3.1 Europa Imaging System

The Europa Imaging System (EIS) is a camera suite developed to investigate the geology of Europa, ice shell, and potential for current activity. EIS builds upon the heritage design of LORRI with improved capability and flexibility. The original science objectives for the Europa Clipper mission that are to be satisfied by EIS include the following:

- Characterization of the ice shell and any subsurface water including heterogeneity, ocean properties, and surface-ice-ocean exchange nature.
- Understand habitability of the ocean on Europa through composition and chemistry.
- Understand the formation of surface features, including sites of recent or current activity.
- Characterization of high science interest locations.

The objectives above reflect and closely align with those of PROSERPINE and will therefore be included on the Pluto orbiter.

4.3.2 Radar for Europa Assessment and Sounding: Ocean to Near-Surface

Radar for Europa Assessment and Sounding: Ocean to Near-surface (REASON) is a dual frequency, ice penetrating radar designed for the upcoming Europa Clipper mission. REASON is equipped to conduct four key measurements as follows [20]:

- Sounding: To probe an icy shell of the object of interest.
- Altimetry: To determine surface elevations.
- Reflectometry: To study surface composition and roughness.
- Plasma and Particles: To characterize the ionosphere for detection of active plumes.

REASON was included on the Europa Clipper mission to conduct terrain related science objectives. These objectives are closely aligned with those of PROSERPINE and thus the instrument can be included on the Pluto orbiter with little to no modification. The science objectives REASON will explore for the Europa Clipper mission include:

- Characterization of shallow, subsurface water distribution.
- Search for ice-ocean interface while characterizing the global structure of the ice shell.
- Investigate material exchange processes of ocean, ice shell, surface, and atmosphere.
- Constrain the amplitude and phase of the gravitational tides.
- Characterize scientifically compelling sites and hazards for potential future landed missions.

4.3.3 Submillimeter Wave Array Spectrometer

The Submillimeter-Wave Spectrometer instrument developed by the Jet Propulsion Laboratory provides unique data on atmospheric physicals and composition. The wave spectrometer will provide the knowledge of how global atmospheric circulation patterns of planets differ from those of Earth and Mars. Information of processes, reactions, and chemical cycles controlling the chemistry of atmospheric levels will also be provided. The instrument allows sub-ppb sensitivity for trace species, direct temperature, wind measurements, and pressure [21].

4.3.4 Magnetometers

To fulfill the science objectives that have been outlined, the science instrumentation payload will include a set of two magnetometers including: one Fluxgate Magnetometer (FGM), and one Vector/Scalar Helium Magnetometer. Based on recommendations from two members of the New Horizons team, the magnetometers were included in the design of PROSERPINE. The magnetometers will be used to study the magnetosphere and ionosphere at the Jovian system, and determine the existence of any magnetic field at the Pluto - Charon system. Adding the magnetometers to PROSERPINE will allow for new scientific discoveries about how the system formed and how Pluto and its moons interact with each other. Because magnetometer sensors are sensitive to electric currents and the spacecraft instruments and system emit a micro- magnetic field, the magnetometers will be located on an extendable boom away from the rest of the spacecraft. This will help to mitigate the interference.

The design for the magnetometer suite used for PROSERPINE is based on the magnetometer designs for the Juno mission currently orbiting Jupiter, as well as Cassini. The first part of the magnetometer suite is the fluxgate magnetometer. The fluxgate magnetometer will be located five meters from the rest of the spacecraft. A fluxgate magnetometer measures the background magnetic field, of a system, and develop a 3D model of the magnetosphere of a planet. The characteristics are summarized in Table III [22].

Table III: Fluxgate Magnetometer Data [22]

Characteristic	Value
Frequency (Hz)	30
Boom Length (m)	5
Volume (m^3)	$6.48e^{-3}$
Data Rate (bit/s)	2.41
High-Sensitivity Range Resolution	± 64 nT ± 0.5 nT
Low-Sensitivity Range Resolution	± 320 nT ± 2.5 nT

The second magnetometer, onboard PROSERPINE, is a Vector/Scalar Helium Magnetometer (V/SHM), a magnetometer that can function in a vector mode where the magnitude and direction of the magnetic vector can be determined. It can also function in a scalar mode, where the magnetometer will only measure the magnitude of the magnetic field. The V/SHM will be located at the end of an 11 meter boom. Its characteristics are summed in Table IV [22, 23].

Table IV: V/SH Magnetometer Data [22]

Characteristic	Value
Frequency (Hz)	10
Boom Length (m)	11
Volume (m^3)	$6.48e^{-3}$
Data Rate (bit/s)	2.48
SHM Resolution	36 pT ± 256 - 16384 nT
High-Sensitivity Range Resolution	3.9 pT ± 32 nT
Low-Sensitivity Range Resolution	31.2 pT ± 256 nT

4.3.5 Solar Wind Ion Analyzer

The Solar Wind Ion Analyzer (SWIA) was launched on the MAVEN mission and will measure the solar wind ion flows around Mars. SWIA will be used to aid in the characterization of the upper atmosphere of Mars. The instrument will also aid in the parameterization of escaping atmospheric gases to extrapolate the total loss to space throughout the lifetime of Mars. These objectives for the MAVEN mission are easily transferable to the Pluto orbiter to accomplish the atmosphere science objectives of PROSEPINE.

To accomplish the scientific goals, SWIA is equipped with toroidal energy analyzer and electrostatic deflectors to provide a 360 x 90 degree field of view.

4.3.6 Mass Spectrometer for Planetary Exploration

The Mass Spectrometer for Planetary Exploration (MASPEX) is to be used aboard PROSERPINE to replace the duties previously assigned to the PEPSSI instrument for the New Horizons mission. The MASPEX will determine the composition of the atmosphere of Pluto by taking gas samples through cryotrapping and then analyzing for any unique compounds not previously found by the New Horizons flyby [25]. MASPEX will also have an emphasis on satisfying the science objective of determining the ion tail composition trailing Pluto. The device is set for its first launch in 2022 aboard the Europa Clipper mission. This instrument is currently at a TRL level 6 and when flown for the first time will be the most sensitive instrument of its kind by a factor of 100,000 [26]. The MASPEX being a newer and much more sensitive device is why the PEPSSI device has been chosen to be replaced for this orbiter mission. The instrument is predicted to have an incredibly high resolution and throughput as it can store 100,000 ions to be extracted at a rate of 2 kHz [27].

4.3.7 Ultraviolet Imaging Spectrometer/Spectrograph

The ultraviolet imaging spectrometer (UVS) to be used aboard PROSERPINE is the same spectrometer used aboard New Horizons which is known as ALICE. The ALICE instrument will determine the abundance of certain chemical compounds of the atmosphere of Pluto within the wavelengths of 520 and 1870 Å [28]. The chemical compounds, the gradient of the upper atmosphere, the atmospheric haze optical depth, and the escape rate of the atmosphere will all be addressed in further detail by ALICE [28]. The device had its first launch in 2006 aboard the New Horizons mission. This instrument is currently at a TRL 9 as a result of this.

4.3.8 Europa Thermal Emission Imaging System

The Europa Thermal Emission Imaging System (E-THEMIS) was built through a partnership of Arizona State University and Ball Aerospace. The combination of high spatial resolution, large area imaging, and high precision temperature determination allows for the mapping of temperature anomalies. E-THEMIS is also able to interpret surface properties and processes by mapping thermophysical properties. The microbolometer detector included on E-THEMIS is radiation tolerant with additional radiation hardened Read-Out Integrated Circuits (ROIC) [29].

The science objectives from the Europa Clipper mission that E-THEMIS was designed to accomplish are transferrable to PROSERPINE. The E-THEMIS Clipper objectives include:

- Detect and characterize thermal anomalies on the surface that are indicative of recent or active resurfacing and venting events.
- Identify active plumes
- Determine the regolith particle block abundance, size, and subsurface layer for possible hazard-free landing regions.

4.4 Summary

This chapter covered the various scientific objectives and instruments for PROSERPINE. Descriptors of the five main science objectives are given within this chapter as well as the necessary instrumentation required to complete these objectives. The science objectives that were completed upon the New Horizons missions were described to give the background of what has been completed at Pluto. Given this background, the objectives that need to be conducted or completed allowed for down selection methods to select which science objectives were the best fit for PROSERPINE. The science objectives selected for this mission are as stated below:

- Terrain of Pluto
- Atmosphere of Pluto
- Orbital Mechanics of Pluto-Charon System
- Moons of Pluto
- Planetary flyby

The instruments used to complete these science objectives were also described within this chapter. The instruments that were used upon the New Horizons missions were described to give the background of what has been previously been used at Pluto. Given this background, instruments were chosen to provide more recent derivations of the instruments of New Horizons if available. The following instruments were chosen to conduct the science objectives for PROSERPINE:

- Europa Imaging System (EIS)
- Ice Penetrating Radar (Reason)
- Submillimeter Wave Array Spectrometer (SWAS)
- Magnetometer (FGM)
- Magnetometer (V/SHM)
- Solar Wind Ion Analyzer (SWIA)
- Mass Spectrometer for Planetary Exploration (MASPEX)
- Ultraviolet Imaging Spectrometer/Spectrograph (UVS)
- Europa Thermal Emission Imaging System (E-THEMIS)

5. TRAJECTORY

Within this chapter, the trajectory for PROSERPINE will be discussed.

5.1 Launch

A MATLAB code, based off Tsiolkovsky's rocket equation, was written to simulate the launches of different rockets. This code was used to conduct a trade study to identify a launch vehicle that provided adequate hyperbolic excess velocity (ΔV) for the given payload mass [32]. The trade study accounted for both gravity and drag loss [30-33]. It assumed that the first stage of the launch vehicle would reach a parking orbit of 165 km, and was fully expendable. The code accuracy is shown in Figure 2. The figure shows a drag loss of 8.5% and 10%. For conservative calculations it was assumed to have a 10% drag loss [32]. Table V shows the requirements for the launch vehicle. The study focused on the requirements for the 2031 launch window.

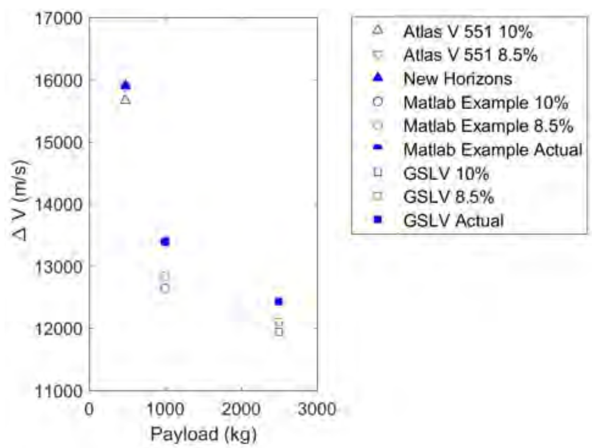


Figure 2: Validation and Accuracy of ΔV Code

The trade study was conducted on 18 different rocket configurations stemming from six different rocket families [34-41]. All of the rockets that were analyzed are shown in Figure 3. The Atlas V family was also analyzed with a third stage, which consisted of a Star 48-B shown in Figure 4 [35]. The Star 48-B was used as the third stage for the New Horizons mission. All Atlas V vehicles have the capability to use the Star 48-B as a third stage [31,36]. During an interview with two New

Table V: Launch Vehicle Requirements

Launch Window	ΔV	Payload Mass	Fairing Diameter	Cost
2031	12 km/s	4442 kg	≥ 4.2 m	Affordable
2032	13.5 km/s	4442 kg	≥ 4.2 m	Affordable

Figure 3: Investigated Launch Vehicles

Horizons team members the Atlas V and Vulcan were suggested launch vehicles for the proposed mission [42]. There was no official published data for the Falcon Heavy, New Glenn, and Vulcan. Other sources were used in place of official documents [39-41].

Figure 4: Star 48-B (3rd Stage of Atlas V 551)

The initial trade study is shown in Figure 5. This figure analyzes the ΔV capability for a range of payload masses. The actual New Horizon data and code calculated data are present on this chart. The required ΔV and payload mass are marked with a star. It was concluded that third stage of the Atlas V rocket becomes burdensome for ΔV capabilities for payloads over 500 kg. The third stage Atlas V rockets were removed from further study. The Delta IV M, Delta IV M+ (4,2), Delta IV M+ (5,2), Delta IV M+ (5,4), and Delta

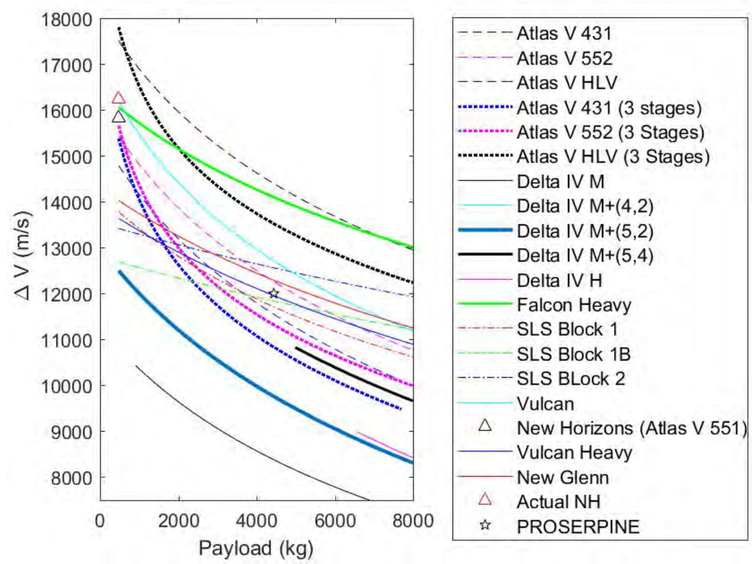


Figure 5: Initial Trade Study for Launch Vehicles

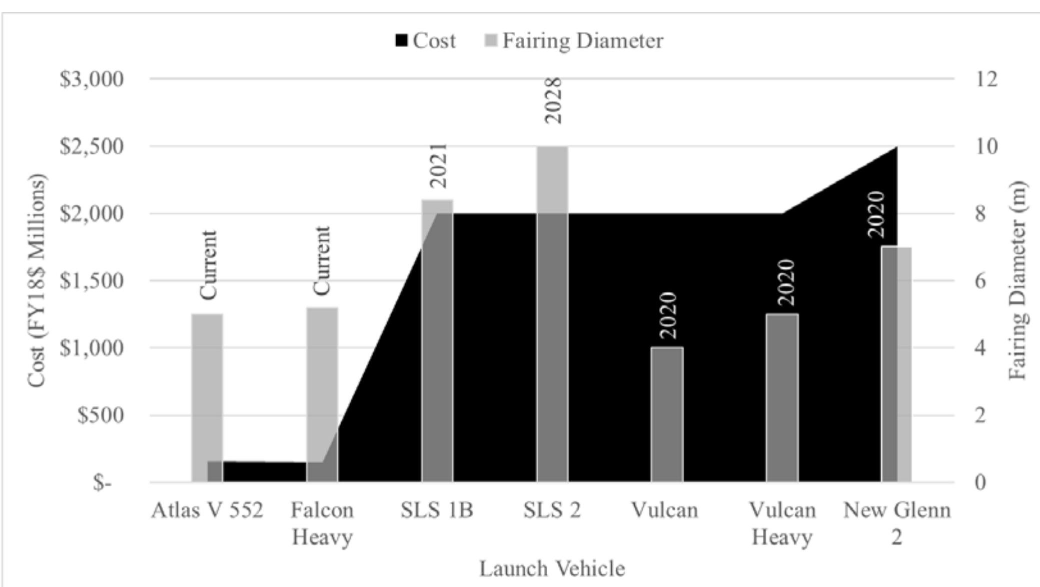


Figure 6: Cost and Fairing Trade Study

Heavy were found to be insufficient and were removed from the study. The Atlas V HLV and the SLS Block 1 have both been canceled and were removed from the study. Due to insufficient data the Falcon Heavy, Vulcan, and New Glenn vehicles were removed. The staging masses were determined not reliable enough to keep them in the study. The cost and fairing diameter study is shown in Figure 6 [42-47].

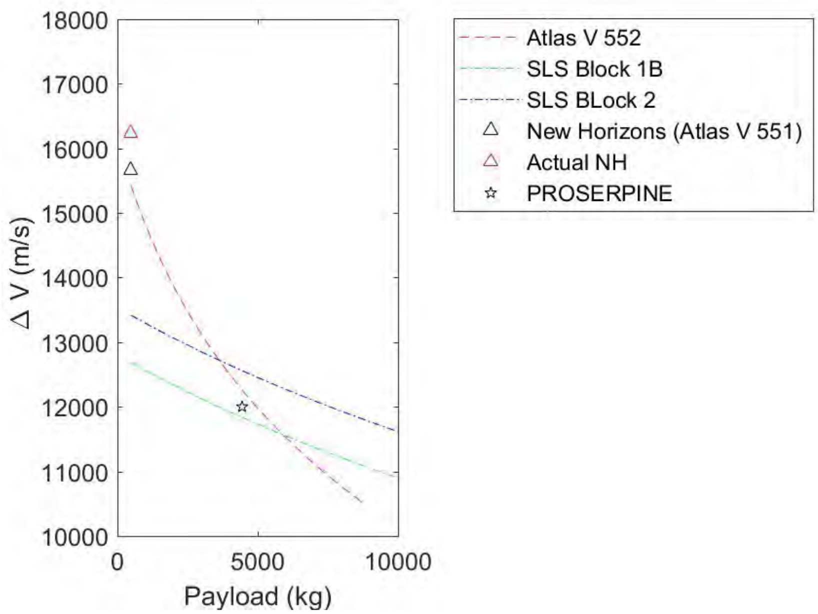


Figure 7: Final Trade Study for Launch Vehicles

The final trade study is shown in Figure 7. Based on this study it was determined that both the SLS block 2 and the Atlas V 552 are capable of launching PROSERPINE in the 2031 window. Due to the high cost of the SLS Block 2, the Atlas V was chosen as the launch vehicle for PROSERPINE.

5.2 Interplanetary Flight Path

Within this section, the interplanetary flight path for PROSERPINE will be discussed.

5.2.1 Downselection

Overall, three general concepts were considered for the interplanetary flight path from Earth to Pluto. These concepts were:

- Flight Paths without a Flyby
- Flight Paths with a Flyby to Accelerate
- Flight Path with a Flyby to Decelerate

Firstly, the feasibility of a flight path without a flyby was determined. In astrodynamics, Hohmann transfers are the simplest and most efficient transfers between two circular or elliptical orbits [49]. Base calculations for a Hohmann transfer from the orbit of Earth to the orbit of Pluto were performed. Assuming the mean distances of Earth and Pluto from the Sun, the time of flight for a Hohmann transfer from Earth to Pluto would be around 44.6 years. Since Pluto is approaching the apoapsis of the orbit around the Sun, this time of flight would increase in the near- and long-term future. In addition, this type of flight path does not include the necessary plane change between orbit of Earth and the orbit of Pluto. This plane change would result in added time as well as added propellant. Thus, due to the constraint of 25 years for the primary mission length listed in the RFP [1], a Hohmann transfer cannot be used.

Furthermore, base calculations for the lowest-energy flight path, without a flyby, between Earth and Pluto for a specific time of flight were performed using a simulation written in MATLAB. This simulation was based on the Lambert-Battin Method for solving Lambert's Problem [48, 49]. For these calculations, the specific time of flight was chosen to be 24 years, due to the constraint of a one-year orbital duration listed in the RFP [1]. As depicted in Table VI, which shows the results of these calculation. The ΔV requirement at Earth was 11.7 km/s equating to a time of flight of 24 years. With regards to launch vehicles, this ΔV requirement is feasible. However, the arrival velocity at Pluto was 2.8 km/s. This means deceleration would be necessary to achieve orbital insertion at Pluto. Deceleration, of course, would result in a longer primary mission length. Decreasing the time of flight would not solve this problem since the ΔV requirement at Earth and the arrival velocity at Pluto would increase as a result. In addition, this type of flight path does not include the plane change between orbit of Earth and orbit of Pluto, which would result in added time as well as added propellant. Thus, the lowest energy flight path, without a flyby, cannot be used.

Table VI: Lowest Energy Flight Path without Flyby

Calculation	Result
Time of Flight	24 years
Departure Velocity	41.7 km/s
ΔV Requirement	11.7 km/s
Arrival Velocity	2.8 km/s

Secondly, the feasibility of a flight path with a flyby, specifically around one of the outer planets, to accelerate was determined. Overall, with aid from simulations of the solar system [50], Jupiter was determined be available for a flyby to accelerate around 2030, while Saturn would be available for a flyby to accelerate around 2050 due to the alignment of the planets. Uranus and Neptune, on the other hand, would not be available for a flyby until well after 2100. In addition, this type of flight path does include the plane change between the orbit of Earth and the orbit of Pluto. Thus, a flight path with a flyby to accelerate, specifically a flyby of Jupiter and Saturn, can be used. However, due to the lack of feasibility for the launch date required for a flyby of Saturn, meaning accurate predictions on technological advances cannot be made this far into the future, this type of flight path was eliminated from further consideration.

Thirdly, the feasibility of a flight path with a flyby, specifically around one of the outer planets, to decelerate was determined. The flight paths with a flyby of Jupiter or Saturn to decelerate were eliminated from consideration due to the proximity of these planets to Earth. The expectation was, with deceleration, the primary mission length would be near or over the constraint of 25 years for the primary mission length listed in the RFP [1]. This leaves the flight paths with a flyby of Uranus or Neptune. Uranus and Neptune are available for a flyby to decelerate, with an earlier launch being better [50]. In addition, this type of flight path includes the plane change between the orbit of Earth and the orbit of Pluto. Thus, a flight path with a flyby to decelerate, specifically a flyby of Uranus and Neptune, can be used. However, this type of flight path was eliminated from further consideration due to the lack of precedent for the orbital maneuvers required for a flyby to decelerate at this scale (i.e. around a planet rather than a moon).

5.2.2 Simulation

As determined by the downselection of the interplanetary flight path for PROSERPINE, a flight path with a flyby of Jupiter is the most feasible. Preceding flybys of inner planets were considered. However, existing and future launch vehicles possess the required ΔV capability for the necessary payload capability for a flight path with a flyby of Jupiter. Thus, preceding flybys were eliminated from further consideration as these flybys would increase the amount of orbital maneuvers as well decrease the feasibility of the launch date. Though the synodic period is shorter for inner planets than for outer planets, the duration of viable launch windows for inner planets is shorter than for outer planets.

A simulation, based on the Lambert-Battin Method [48,49], was written in MATLAB to determine the viable launch windows as well as the velocities and time of flights corresponding with each of these launch windows. Overall, the simulation consisted of three distinct parts. In the first part, the Lambert-Battin Method was used in an iterative fashion to calculate the velocity vector at departure and the velocity vector before the flyby for a given time of flight, initially one day. During this portion, the direction of the departure velocity with respect to orbit of Earth was calculated. If the direction of the departure velocity with respect to orbit of Earth was not within the desired tolerance (a 5° half-angle cone), the time of flight was increased by a time step of one day. If the direction of the departure velocity with respect to orbit of Earth was within the desired tolerance, the simulation would move to the next part. In the second portion, the Lambert-Battin Method was used in an iterative fashion to calculate the velocity vector after the flyby and the velocity vector at arrival for a given time of flight, initially one day. During this part, the hyperbolic excess velocities before and after the flyby were calculated. If the magnitude of the hyperbolic excess velocity before the flyby was not within the desired tolerance of 1% for the hyperbolic excess velocity after the flyby, the time of flight was increased by a time step of one day. If the magnitude of the hyperbolic excess velocity before the flyby was within the desired tolerance of 1% for the hyperbolic excess velocity after the flyby, the simulation would move to the next portion. In the third part, the turn angle and the radius of closest approach for the flyby were calculated with the results from the first and second parts. If the radius of closest approach was under the desired value of 32.25 Jupiter radii, the simulation would return to the first part and the time of flight would be increased by a time step of one day. If the radius of closest approach was over the desired value, the simulation would display the results.

This simulation was validated using an example problem based on Voyager 1 as well as data from New Horizons [48]. The velocity vectors and the magnitude of the velocity vectors displayed by the simulation were within reasonable range of those from the example problem or New Horizons, and the time of flight displayed by the simulation was accurate to within days of the time of flight from the example problem or New Horizons.

5.2.3 Results

Since the viability of an interplanetary flight path between two planets is dependent on the alignment of those planets, a shortened version of the simulation was used to determine the exact dates of viable launch windows. As shown in Figure 8, these viable launch windows were plotted using the standard definitions for true (i.e. viable) and false (i.e. not viable).

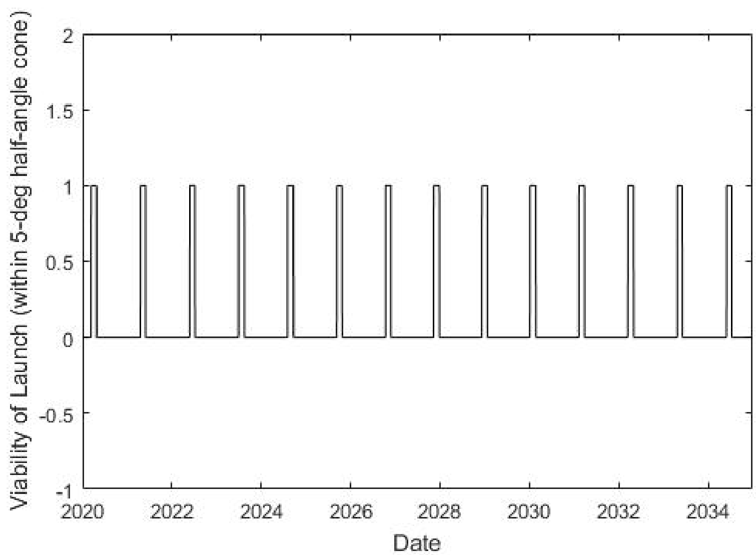


Figure 8: Viable Launch Windows

Once all of the viable launch windows were determined, the full version of the simulation was run for each of these launch windows. By doing so, it was determined that only two of the launch windows had an interplanetary flight path with a time of flight less than the constraint of 25 years for the primary mission length listed in the RFP [1]. These launch windows were from February 10th to March 28th during 2031 and from March 19th to May 1st during 2032. The time of flight, the departure velocity at Earth, and the arrival velocity at Pluto for both of these launch windows are shown in Figure 9 - Figure 14. The launch window during 2031 had a lower time of flight compared to the launch window during 2032 as well as a lower departure velocity at Earth. However, the launch window during 2031 had a higher arrival velocity at Pluto compared to the launch window during 2032. Of note, the values for the arrival velocity at Pluto and the time of flight are before deceleration.

In conclusion, the optimal launch window for PROSERPINE is from February 10th to March 28th during 2031, specifically February 17th to March 11th. The departure velocity at Earth is lower for 2031 than the departure velocity at Earth for 2032 (~12 km/s versus ~13.5 km/s). This ΔV requirement is within the ΔV capability of the Atlas V 552, a reliable and affordable launch vehicle. Although the arrival velocity at Pluto for 2031 was higher than the arrival velocity at Pluto for 2032 (~11 km/s versus ~7.5 km/s), the time of flight for 2031 was lower than the time of flight for 2032 (~15 years versus ~19 years). For both launch windows, deceleration will be necessary to achieve orbital insertion at Pluto, resulting in a longer time of flight. This means, after interplanetary travel, the time remaining for an orbital duration around Pluto will be greater in 2031 than in 2032. Thus, PROSERPINE will launch during the launch window in 2031.

5.3 Deceleration

The mission architecture requires a continuously burning NEXT ion thruster to provide the necessary change in velocity for spacecraft capture at Pluto. The NEXT ion thruster produces a constant thrust of 0.236 N for this simulation. Trajectory determination for this flight segment from Jupiter to Pluto thus requires a numerical solution because the maneuvers are non-impulsive. To incorporate the changing spacecraft mass from the continuous burn, a differential equation was used to model the system. This equation was solved at small time steps. [51]

A trajectory from flyby at Jupiter to capture at Pluto was found using a MATLAB simulation to solve the differential state equation at discrete time steps. The resulting trajectory has a total trip time from Jupiter to Pluto of 16 years. The resulting trajectory is comprised of three phases: a cruise phase, a slowdown burn, and a final cruise. The time of flight for the cruise phase is 12 years and 246 days, slowdown burn is 3 years and 75 days, and final cruise phase is 30 days. The resulting trajectory places the spacecraft 66,300 km from Pluto matching velocity magnitude within 630 m/s. The initial spacecraft mass is 4440 kg at Jupiter, the mass after the three flight phases to reach Pluto is 3380 kg, resulting in a propellant mass of 1060 kg. Figure 15 shows a view of the entire trajectory from Jupiter to Pluto. Figure 16 shows a more detailed view of the slowdown and pointing burn phase.

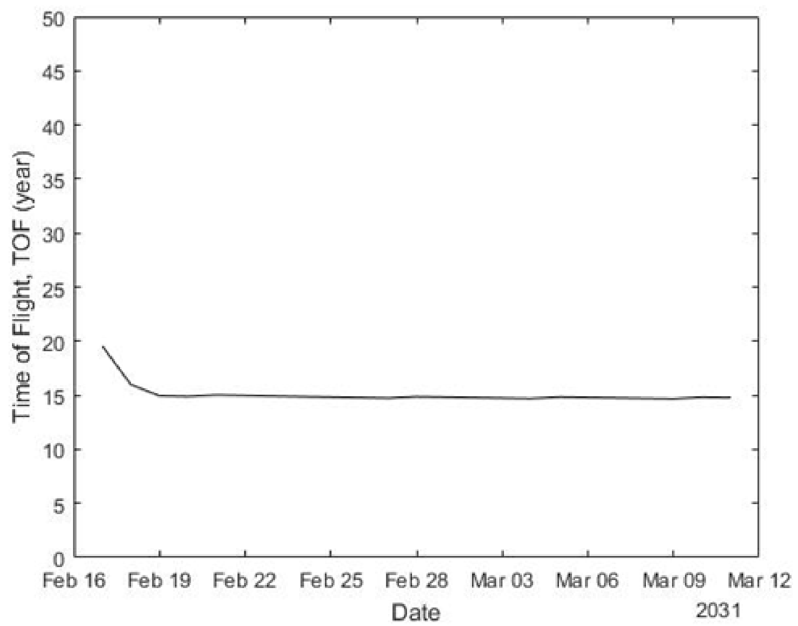


Figure 9: Time of Flight (2031)

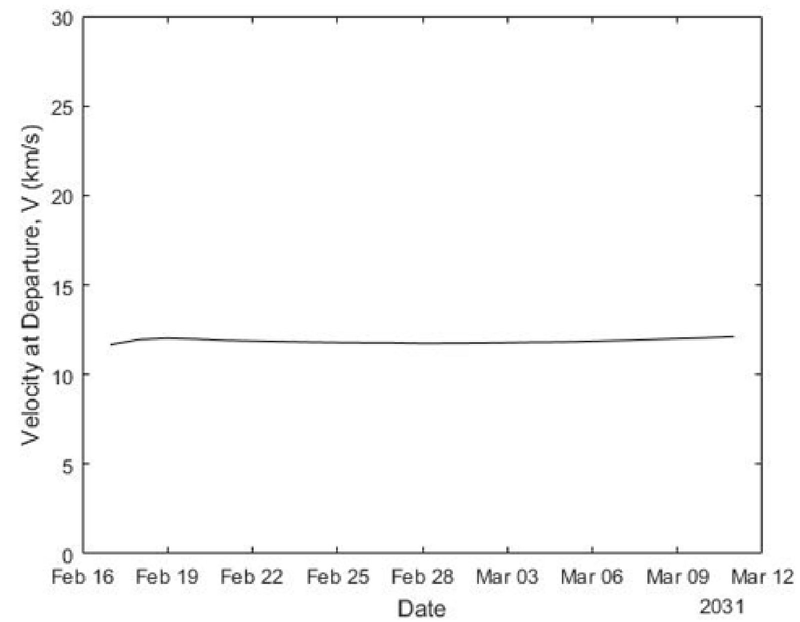


Figure 10: Departure Velocity at Earth (2031)

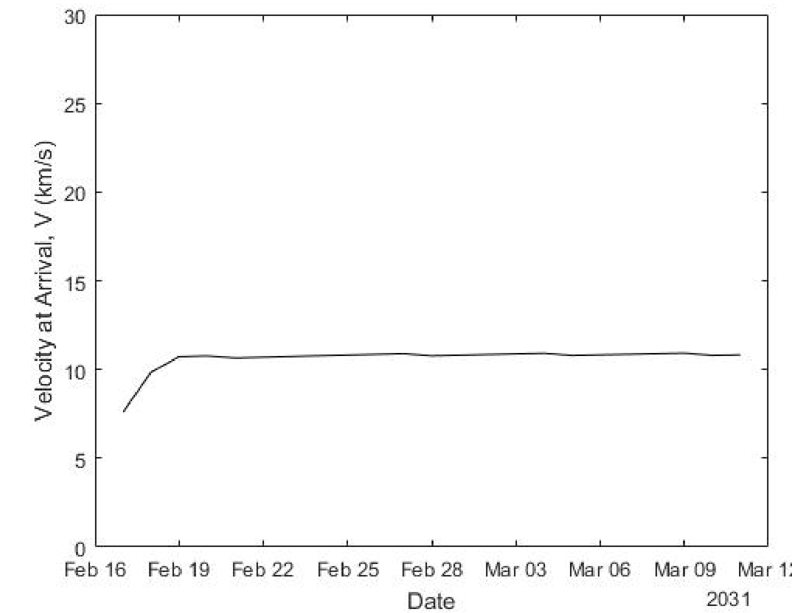


Figure 11: Arrival Velocity at Pluto (2031)

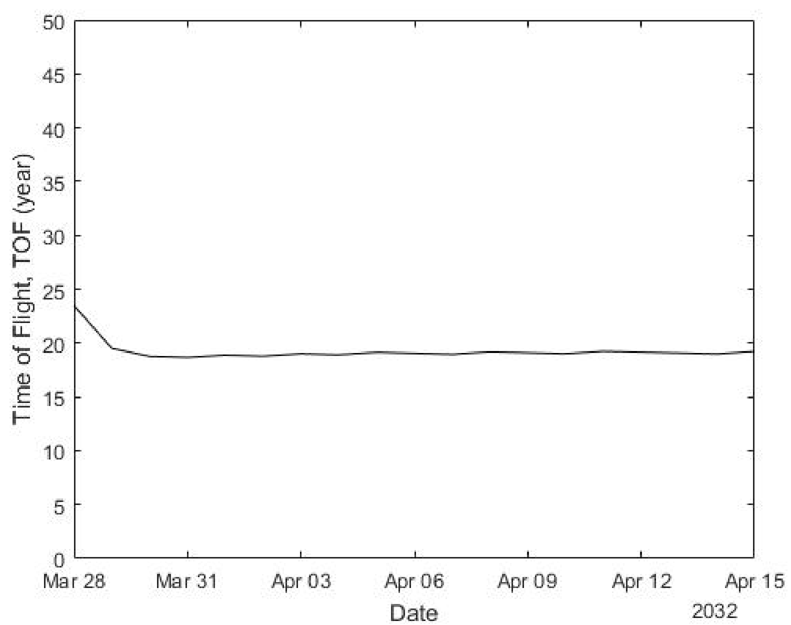


Figure 12: Time of Flight (2032)

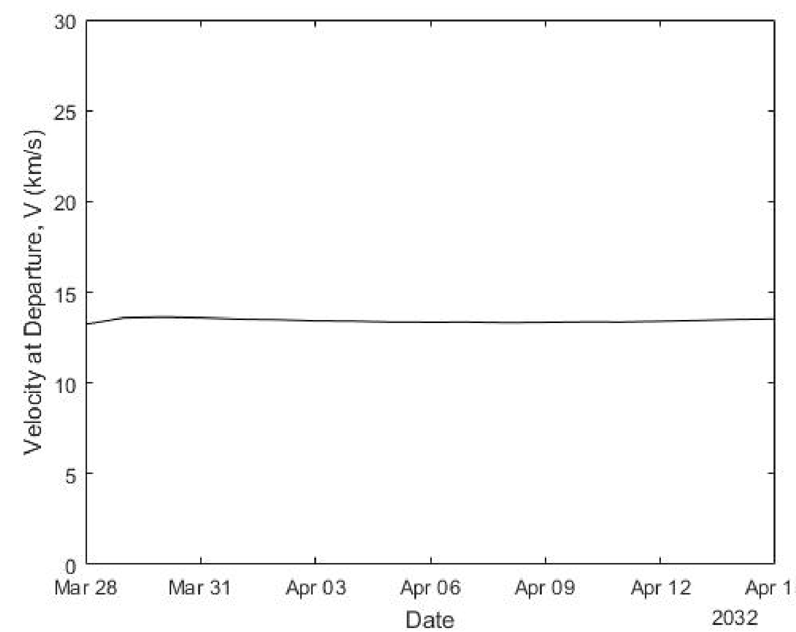


Figure 13: Departure Velocity at Earth (2032)

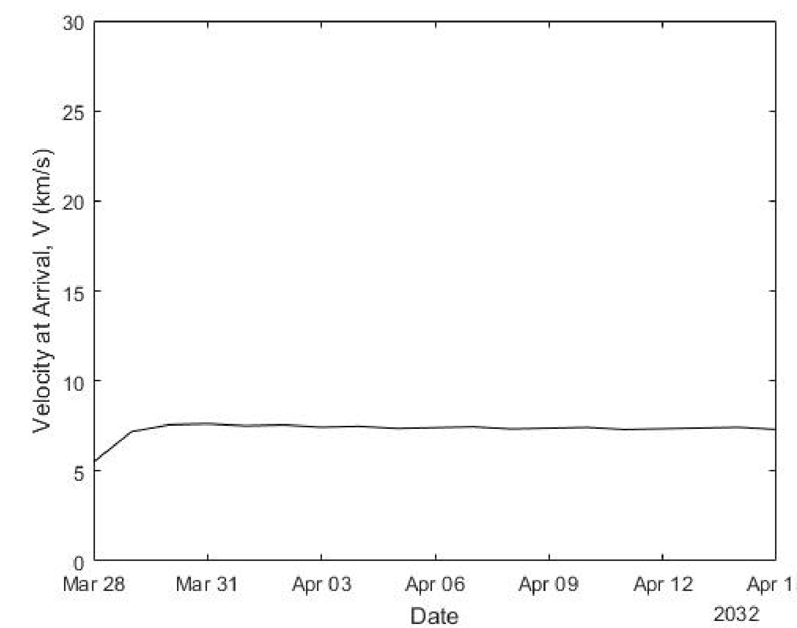


Figure 14: Arrival Velocity at Pluto (2032)

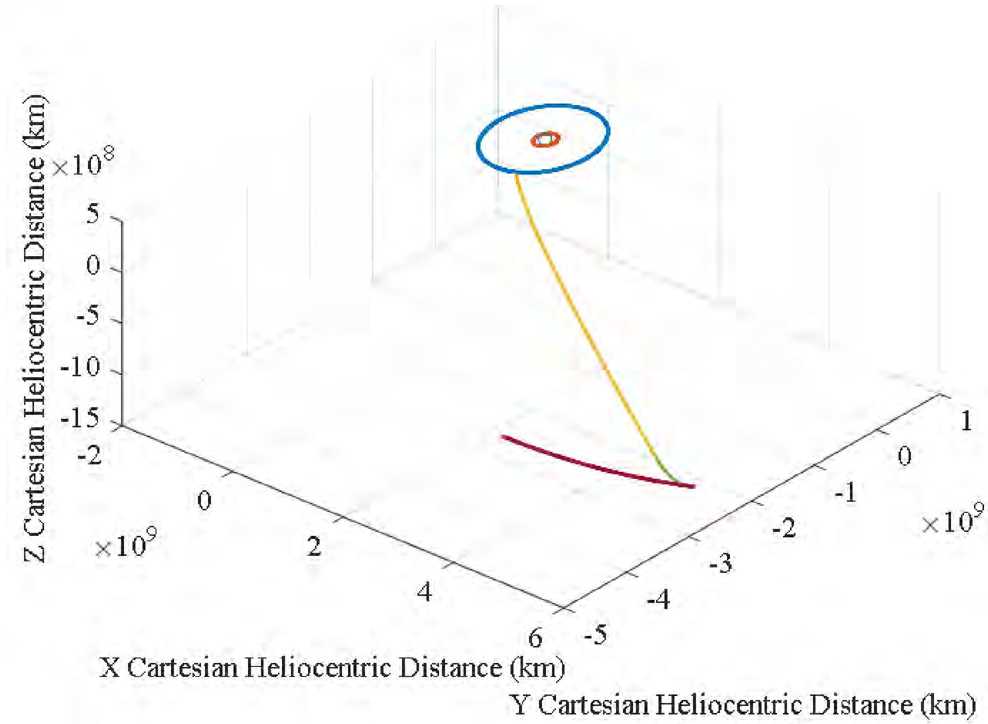


Figure 15: Off-Axis View of Trajectory from Jupiter to Pluto (Sun = Light Blue, Earth Orbit = Orange, Jupiter Orbit = Dark Blue, Pluto Orbit = Red, Cruise = Yellow, Burn = Green)

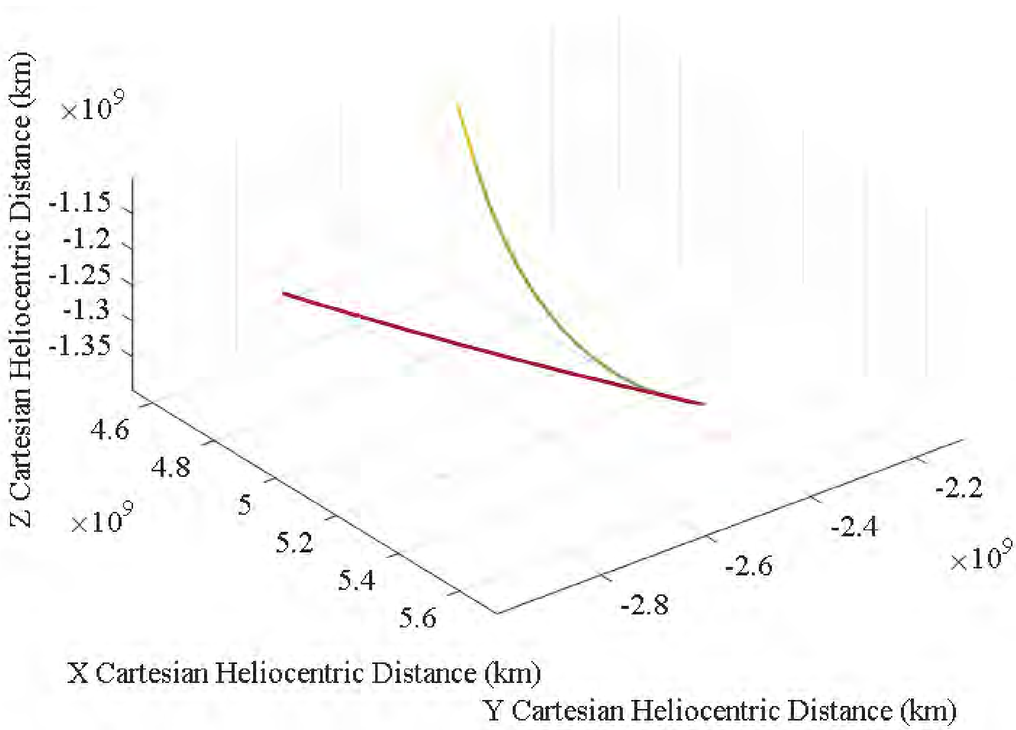


Figure 16: Enlarged Off-Axis View of Burn Phase (Pluto Orbit = Red, Cruise = Yellow, Burn = Green)

5.4 Mission Operations

Within this section, the mission operations for PROSERPINE will be discussed.

5.4.1 Circularization

After deceleration and rendezvous with the Pluto-Charon system, PROSERPINE must achieve orbital insertion. To accomplish this, PROSERPINE will burn retrograde when reaching closest approach to the system barycenter. The spacecraft will wait until it is nearing perihadion to burn to exploit the Oberth Effect. Due to the low mass of the Pluto-Charon system, the necessary ΔV can be achieved in the course of a single passage.

5.4.2 Mapping

Following initial capture, PROSERPINE will continue the initial burn until the orbital period has been reduced to 60 days. This allows the spacecraft to begin science operations immediately, without spending an excessive interval on a highly-eccentric orbit carrying it far away from the research targets of the system. It will then cruise to apohadion to perform a plane change into a near-polar mapping orbit. Assuming a 90° plane change is the maximally-conservative case, acknowledging the large uncertainties in the arrival parameters. Once again, the low mass of the Pluto-Charon system allows this to be completed in the course of a single orbit.

Once the spacecraft is in a polar orbit, it will conduct a series of short burns near perihadion to reduce its orbital eccentricity and semi-major axis. Eventually, the orbit will be near-circular, allowing for extended observation of Pluto at close range. A 3000-km orbit allows for reasonably high-resolution imaging while maintaining a reasonable propellant budget in Pluto orbit.

5.4.3 Moon Rendezvous

After planetary scientists are satisfied with the mapping resolution of Pluto, the spacecraft will begin a progressive prograde burn to raise its orbit out to the moons. Performing another plane change at sufficiently low altitudes to orbit the moons would be prohibitively expensive; therefore a series of flybys is preferable. PROSERPINE will enter resonant orbits with the various moons, allowing for repeated flybys of each target.

The spacecraft will have already observed Charon at reasonably-high detail during the mission operations phase focused on Pluto. When not in close proximity to the moon, the spacecraft will turns its instruments back towards Pluto and continue observations at-a-distance. This pattern will repeat after the transfers to Styx, Nix, Kerberos, and

Hydra. As PROSERPINE ventures further away from the primary body, the science focus will shift more to the orbital dynamics of the system as a whole.

5.4.4 End-of-Life

Once the mission goals around Pluto have been completed, the spacecraft operations will enter a legacy mode, continuing scientific observations until the powerplant or attitude control fuel reserves fall below critical levels. At this point, PROSPERINE will be placed on a collision trajectory with Pluto or Charon. This will require a trivial quantity of propellant given the high final observation orbit. Immediately prior to impact, the spacecraft will broadcast high-resolution images and readings of the body surface back to Earth, providing a final batch of valuable scientific data to end the mission.

5.4.5 Simulation and Results

The path of PROSERPINE through the Pluto-Charon System was calculated using simple numerical integration using the two-body equations of motion, with an additional acceleration term added from thrust. The acceleration on was calculated based on the thrust of the engine and the average spacecraft mass over the time step, as calculated from the NEXT mass flow rate. The acceleration term only applies during propulsive periods. Figure 17 shows the spacecraft path during mission operations. The ΔV, propellant consumption, and burn times for PROSERPINE mission operations at Pluto are summarized in Table VII.

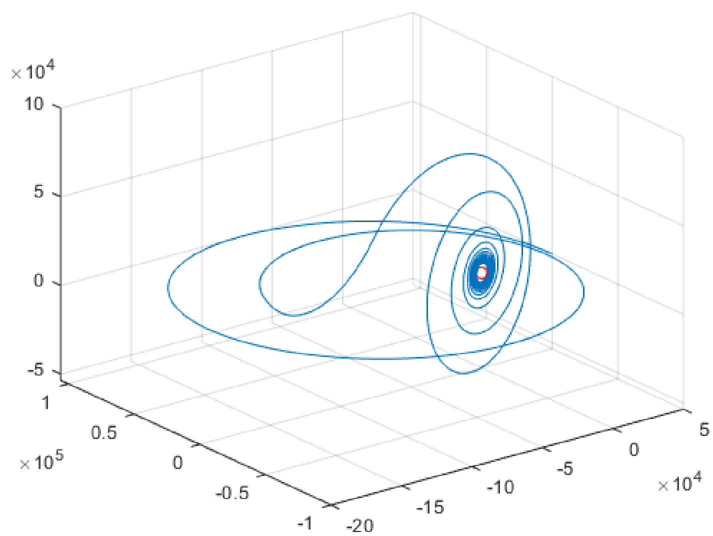


Figure 17: Mission Operations Simulation

Table VII: Mission Operations Requirements

Operations Phase	ΔV	Propellant Mass	Time
Period Reduction	30.5 m/s	2.1 kg	4 days
Plane Change	107 m/s	7.3 kg	15 days
Spiral Down	331 m/s	22.5 kg	35 days
Moon Rendezvous	360 m/s	21.1 kg	45 days
Total	829 m/s	54 kg	97 days

6. PROPULSION SYSTEM

Within this chapter, the propulsion system for PROSERPINE will be discussed.

6.1 Limiting Factors

The propulsion system for PROSERPINE will control the trajectory en route to Pluto, perform deceleration into orbit of the Pluto-Charon system, and enable mission operations after orbital insertion. To reach Pluto within the time-span allotted by the RFP, PROSERPINE needs to take a high-energy trajectory, arriving with considerable excess velocity [1].

The RFP explicitly limits the in-space propulsion system to Technology Readiness Level 6 or above [1]. Additionally, the propulsion system should function reliably after many years in deep space, and keep the overall spacecraft wet mass within the payload capability of current and near-future launch vehicles.

6.2 Possibilities

A number of possible propulsion systems were considered, including chemical, nuclear, and electric propulsion. This section discusses the relevant properties and merits of these systems before the downselection in the following section.

6.2.1 Chemical Propulsion

Chemical propulsion has a long history of successful flights on both manned and unmanned spacecraft, making it the natural choice for risk-conscious mission designers. However, chemical rockets have relatively low performance limits compared to the other system considered for PROSERPINE [49].

Solid rocket motors are a popular choice for in-space propulsion due to their general reliability and ease-of-manufacture. However, the expected specific impulse is relatively low, potentially leading to a high launch mass. They are commonly used on Earth-orbiting and some interplanetary spacecraft, though the effects of prolonged exposure to deep space has not been conclusively determined [49].

Liquid-fueled rockets can realize higher specific impulses, but this requires propellants which do not store readily for extended periods. Space-friendly liquid propellants have specific impulses comparable to solid-fueled motors [49]. However, researchers at NASA Glenn Research Center are developing reduced and zero boil off tankage designs for cryogenic fuels, which provides some insight into the mass which a liquid-fueled rocket would necessitate [52].

6.2.2 Nuclear Thermal Propulsion

Nuclear thermal propulsion (NTP) potentially allows for a dramatic increase in specific impulse without sacrificing spacecraft thrust. However, nuclear rockets require exhaust gases with low molecular mass to achieve these exhaust velocities within realistic temperature constraints, with molecular hydrogen offering the highest value [53].

Storing hydrogen in space for extended periods would again require the zero boil off tankage discussed in the previous subsection. This would contribute to the propulsion system mass, however, the mass contribution from the drive itself would be more significant than for chemical rocket engines.

There are two heat sources for nuclear thermal propulsion: nuclear reactors, and radioisotope thermal generators. Nuclear reactors have been the classic approach, developed in depth by NASA and the Atomic Energy Commission in the 1950s and 1960s. These allow high temperatures and thus high specific impulses, but also require greater mass [54].

Radioisotope thermal propulsion (RTP) represent a less complex approach to developing thrust from a nuclear power source. Instead of relying of the fission of uranium or plutonium, radioisotope rockets exploit the natural decay of plutonium to generate heat. The same heat source could be used as a radioisotope thermoelectric generator (RTG) to provide spacecraft power. The downside of using a radioisotope rocket is depressed specific impulse [55].

6.2.3 Electric Propulsion

The final category of propulsion option to consider is electric propulsion systems. Electric propulsion uses electromagnetic energy to accelerate particles to extremely high velocities. Such systems realize specific impulses in the thousands of seconds, but require a great deal of power to run continuously. The electric thrusters which have been qualified for the expected PROSERPINE burn duration require multiple kilowatts of power.

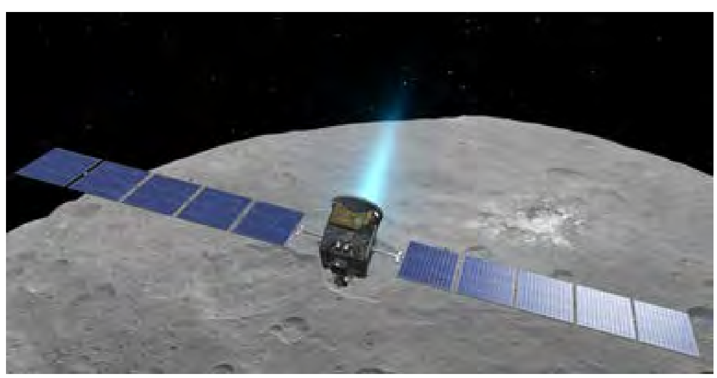


Figure 18: Artist's Rendering of the Dawn Spacecraft [56]

A number of interplanetary spacecraft have already used electric propulsion, including the Dawn mission exploring the asteroid belt (Figure 18). These spacecraft have used solar power sources, which would be challenging to use in the outer solar system. The state-of-the-art are the panels used on the Juno spacecraft currently orbiting Jupiter. (Figure 19). These panels provide an estimate of the panel mass necessary to use electric propulsion in the vicinity of Pluto.

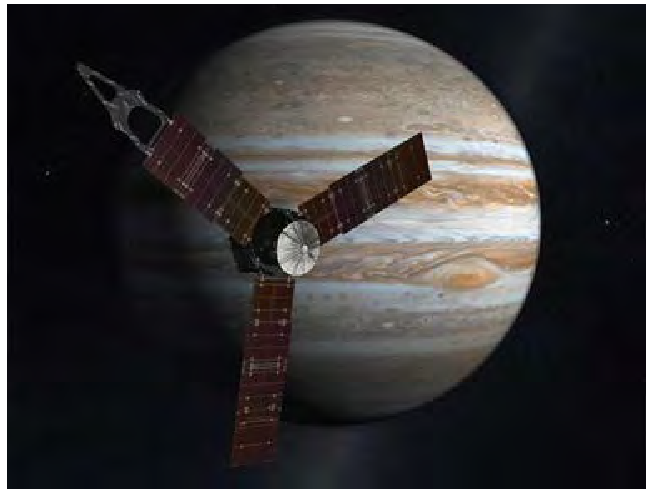


Figure 19: Artist's Rendering of the Juno Spacecraft [57]

Aside from Juno, all spacecraft to visit the outer planets have been powered by radioisotope thermoelectric generators. RTGs generate electricity through the thermoelectric effect, with heat provided by the decay of radioactive elements, usually plutonium [49]. Powering an electric thruster using RTGs has been proposed, but presents certain challenges. Chief among these is low output. All flight units to date have had an electric output in the low hundreds of Watts [61].

Analyses of radioisotope electric propulsion (REP) generally assume a much lower spacecraft mass than that of PROSERPINE [64]. These analyses include outer planetary orbiters, but the instrument load is considerably reduced compared to prototypical orbiters [65]. The RFP specifies that the instrument baseline from New Horizons should be maintained if at all possible [1].

Figure 20: Preliminary Kilopower Reactor Concepts [70]

A final option for powering the electric propulsion system is nuclear reactors. While not currently in use, nuclear reactors have been used as power sources on both American and Soviet spacecraft [66]. In the United States, space reactor development ended in 1973 [54]. The only reactor tested under the

Space Nuclear Auxiliary Power (SNAP) program was SNAP-10A, which orbited successfully in 1965 until suffering a non-nuclear failure [67].

Nuclear electric propulsion (NEP) has been proposed for outer planetary probes, including Pluto orbiters [68]. The primary challenge for an NEP system would be reactor selection. Fortunately, NASA is currently developing a new series of reactors in a joint venture with the Department of Energy for use on lunar and Mars missions, both manned and unmanned [69]. These reactors have an expected output in between 1 and 10 kWe, earning the moniker "Kilopower". Preliminary designs of the Kilopower reactors are shown in Figure 20. Their relevant criteria will be discussed in the following sections.

6.3 Downselection

Within this section, the downselection of the propulsion system for PROSERPINE will be discussed.

6.3.1 Criteria

There are five major merit criteria which will govern the downselection of primary propulsion system of PROSERPINE. These are: mass, technology readiness level, safety, technical risk, and cost. Each of these will be discuss futher.

Wet mass for the spacecraft will play a major role in choosing a launch vehicle. While the RFP sets no boundaries upon the launch vehicle besides a maximum of one, using a current or near-future launch vehicle is strongly preferable [1]. The propulsion system provides three major components of the spacecraft wet mass: the mass of the propulsion system itself, the mass of the propellant, and tankage mass. Hardware masses are based on existing flight units or theoretical analyses, while propellant mass was based on the rocket equation. A preliminary estimate of the non-propulsive bus mass can be made from instrument mass based on an algorithm in Ref. 49. The nominal instrument load of 87 kg yields a bus mass of 650 kg, used in downselection estimates.

Additionally, the RFP specifies that the in-space propulsion system must be at TRL 6 or higher at launch date, rather than at the date which the RFP was issued, as that would limit the spacecraft to at most electric propulsion systems which have already been deployed [1]. As a general rule, chemical systems have a higher technology readiness than electric propulsion systems, which have a higher technology readiness than nuclear systems in turn. Within each category, secondary considerations such as tankage and power systems may override.

Concerns for persons on the ground has been a major consideration affecting the adoption of advanced technologies. Missions in the past have faced protests over relatively innocuous power and propulsion systems; estimating the risks accurately is essential for public acceptance. These risks include dangers to personnel handling spacecraft equipment and to the general public in the event of a launch failure.

While the former category of risk can be mitigated through adequate use safety procedures and with sufficient materiel, the latter is largely a function of launch vehicle reliability and the quality of the launch vehicle adaptor. Should the spacecraft fail to attain an interplanetary escape trajectory, it will eventually re-enter the atmosphere and impact the surface of Earth. Even if it falls far away from human populations, environmental effects from e.g. toxic propellants are worth considering.

Safety ratings are estimated preliminarily based on the perceived dangers of the system in question to handlers and in the event of launch failure. These ratings are necessarily first-order estimates; for a real space system, subject-matter experts should consult experienced reliability and environmental engineers to accurately assess the dangers.

Beyond the raw TRL and safety values, there is a good deal of technological risk involved in several of the propulsion options presented above. This risk may constitute research, development, and testing costs, as well as reducing the odds of success at any given point during the mission. This category compares the not only the TRL but also the cost and time-frame for developing the technology to a reasonable level. Given that this represents one of the largest unknowns in the spacecraft design, perceived technical risk will be a strong factor in the downselection process.

No price limits are imposed on the spacecraft [1]. However, as a general rule, less expensive spacecraft are preferable, as they maximize the overall scientific return of the space program. Therefore, we wish to minimize the cost of PROSERPINE. However, increased spending can, if properly directed, reduce the safety and mass concerns while increasing the readiness of various technologies. As there is no upper limit on cost, we will assign it the lowest weighting in the process of downselecting the propulsion system.

6.3.2 Results

The propulsion system options can be compared using a downselection matrix. Based on preliminary calculations and estimates, each option is assigned a number on a scale from 9 to –9, with increments of three. These values are assigned to the system mass, system TRL, safety, technical risk, and cost. The estimation process is described below, and the system with the highest score selected.

Mass estimates were performed using the rocket equation. Tankage and drive system masses were assumed optimistically from existing flight units or theoretical analyses when sizing equations were not available. The results of these calculations and the parameters used are given in Table VIII.

Table VIII: Parameters for Various Propulsion System Options

Propulsion System	I_{sp}	Tank Mass	Drive Mass	Fuel Mass	Wet Mass
Solid Chemical	290 s	200 kg	50 kg	122,500 kg	123,400 kg
Liquid Chemical	380 s	983 kg	100 kg	65,800 kg	67,500 kg
RTG Thermal	650 s	211 kg	500 kg	9200 kg	10,600 kg
Nuclear Thermal	850 s	214 kg	2000 kg	11,800 kg	14,700 kg
Solar Electric	4190 s	7100 kg	203,000 kg	126,000 kg	436,000 kg
RTG Electric	4190 s	75 kg	1740 kg	1000 kg	3470 kg
Nuclear Electric	4190 s	50 kg	1400 kg	852 kg	2950 kg

The technology readiness level of each system was taken from the 2015 NASA Technology Roadmap for in-space propulsion technologies, unless more recent sources were available [74]. These figures were extrapolated based on the expected technology development timetables presented in the roadmap, with some modification based on known test plans. Certain systems such as advanced ion thrusters are currently TRL 5 but will have been used for space missions launching well before PROSERPINE and thus will be at least TRL 6 by launch date [60].

Safety rankings are primarily based on concerns about toxicity or danger when handling the spacecraft before launch, and the risks in event of launch failure. This was primarily based on the risk of nuclear systems, notably the plutonium used in RTGs. Smaller units have survived launch failures in the past, but it is uncertain whether that would hold for a spacecraft the size of PROSERPINE [61]. The uranium used in nuclear reactors is relatively safe, and will not activate until the spacecraft is on an interplanetary trajectory to avoid the production of dangerous fission products [54]. Additionally, consideration was made for the propellants in question. More notably, solid propellants are relatively stable, while cryogenic propellants such as liquid hydrogen and liquid oxygen are highly flammable [32]. Xenon is chemically inert but is an asphyxiant, which may become significant given the massive quantity required by the solar electric propulsion concept.

Additional technical risks beyond the system TRL also come into play. For chemical propulsion systems, this is largely a matter of tankage technology validation. Electric propulsion risks are mainly located in the attendant power system. For example, the deployment mechanisms for solar electric propulsion would be operating on an unprecedented scale, leading to concerns about failure. Similarly, the supply of plutonium and scalability of RTGs is cause for hesitation. Nuclear thermal propulsion, furthermore, has not been developed in-depth since the 1970s, and complications may appear which the NASA TRL estimates cannot predict [54].

Modelling the cost of various systems is largely a function of mass, which may appear redundant in the context of this downselection. Here, however, it also includes the complexity of the system, and the amount of development which is necessary for its safe inclusion in PROSERPINE. Manufacturing difficulties are also included.

The various categories may now be weighted and the final system selected. System wet mass is weighted at 30% of the total, technology readiness and safety each at 15%, technical risk at 30%, and cost at 10%. Based on these values, Table IX shows that nuclear electric propulsion is the strongest candidate for primary propulsion system of PROSERPINE.

Table IX: Propulsion System Downselection Matrix

Metrics	Weight	Solid	Liquid	RTP	NTP	SEP	REP	NEP
Wet Mass	30%	−6	−6	0	−3	−9	6	9
System TRL	15%	9	3	0	0	6	3	3
Safety	15%	6	3	−6	−6	9	−6	−3
Technical Risk	30%	0	−3	−3	−3	−9	−3	0
Cost	10%	−3	−6	−6	−6	−9	−3	−3

6.4 Configuration

Within this section, the configuration of the propulsion system for PROSERPINE will be discussed.

6.4.1 Xenon Storage

Spacecraft xenon storage is largely a solved problem. The Dawn spacecraft successfully operated using a lightweight xenon tank, designed by Cobham. This tank is shown in Figure 21. It contains 450 kg of xenon while itself only massing in at 22 kg [72]. PROSERPINE will use three such tanks for a total of 1350 kg of xenon propellant and 66 kg of tankage mass. They will be mounted within the main spacecraft bus and cross-fed through the propellant management system to reduce the risk of a single-point failure to the overall mission. At current best estimates, the deceleration phase of the mission will require 1050 kg while the mission operations phase will require 56 kg. Given the total propellant reserve of 1350 kg, this leaves a 18% margin for attitude control purposes and other corrections.

Figure 21: Cobham Xenon Tank [72]

6.4.2 Ion Thruster

A number of ion thrusters are available on the market today; however, the vast majority of these are intended for attitude-control purposes on Earth-orbiting spacecraft. The total impulses for which such thrusters have been validated are considerably lower than 3 years expected for PROSERPINE.

For purposes of risk mitigation, only thrusters manufactured in the United States are candidates. Combined with the thrust time requirement, this effectively limits the options to the NASA Solar Technology Application Readiness (NSTAR) thruster used on Dawn, and the NASA Evolutionary Xenon Thruster (NEXT) currently in development [78].

Of the two, NEXT has a higher thrust, higher specific impulse, and is more likely to be available near the PROSERPINE launch date. NSTAR requires less power, but lower thrust implies longer burn times and may exceed the validation range of the thruster. NEXT has already been successfully evaluated over long periods in a vacuum environment, increasing confidence that it can perform successfully on PROSERPINE [78]. It is expected to be used on the Double Asteroid Redirection Test (DART) mission launching in 2021 [60].

6.4.3 Power System

The only space reactor system currently in development—and within the output range PROSERPINE requires—is the Kilopower Project of NASA [70]. Rather than developing a new reactor from scratch, this is the natural choice for a first-generation planetary spacecraft.

Kilopower reactors are intended to be scalable, which enables estimating the parameters which a reactor of a given output will have [70]. From existing model data, a reactor with an 8.5 kW_e output will mass approximately 1390 kg. NEXT requires 7.23 kW_e during operations, with the remainder being reserved for subsystem uses.

6.4.4 System Layout

In addition to the thruster itself, NEXT includes a gimbal, power processing unit (PMU), and propellant management system (PMS). Electric propulsion systems typically use multiple units for redundancy, and this represents a non-trivial mass increase [79].

All of these components are necessary. Gimbals allow NEXT to align the thrust vector through the spacecraft center of gravity, minimizing the thrust torque due to off-axis masses. The gimbal range is 19° by 17°, enabling significant thrust vectoring [79]. The gimbal step size is 0.003°, enabling highly accurate thrust pointing. The gimbal mass itself is a mere 6 kg [80].

The power processing unit converts electrical power from the nuclear reactor to a form which is useable by NEXT. The PPU has a mass of 34.5 kg, and multiple units are common on electrically propelled spacecraft [80].

The propellant management system has a combined mass of 5.0 kg and consists of a high-pressure and low-pressure assembly. It requires an additional 20.2 W_e during spacecraft operations, which will be included in the subsystem power total. Once again, multiple units are commonly used [80].

Finally, the thruster itself has a mass of 12.7 kg [80]. Dawn used three NSTAR units, cycling between the three to minimize thruster decay. Rotating between multiple units mitigates performance loss [78].

The trade-off between redundancy and system mass is not easy to make. Based on existing spacecraft designs, PROSERPINE will use three NEXT units with their appropriate gimbals, two PPUs, and two PMSs. The resultant masses for the propulsion subsystem are summarized in Table X, including the three xenon propellant tanks.

Table X: Propulsion System Component Summary

Component	Quantity	Unit Mass
NEXT	3	12.7 kg
Gimbal	3	6.0 kg
PPU	2	34.5 kg
PMS	2	5.0 kg
Xenon Tank	3	22 kg
Total		201.1 kg

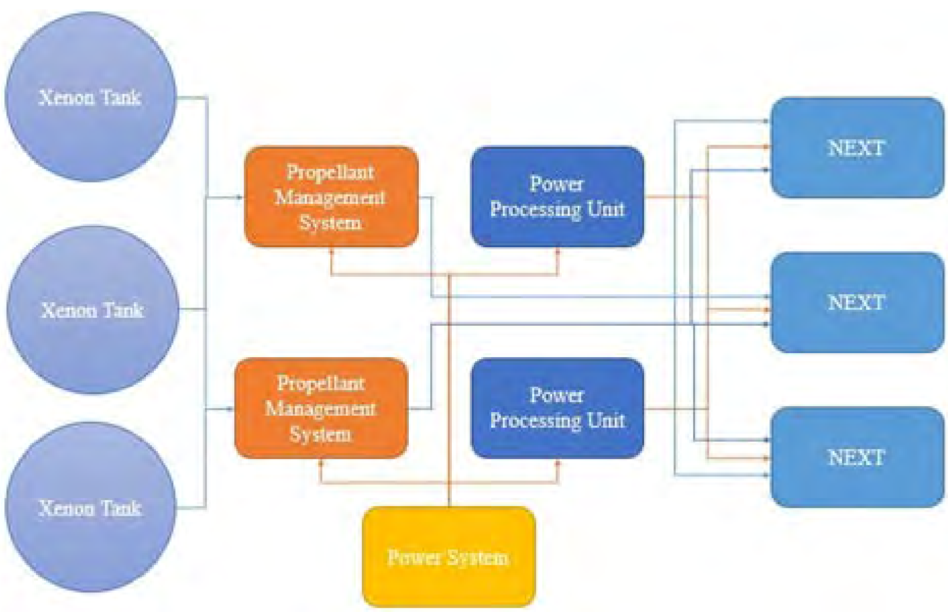


Figure 22: Propulsion System Power Diagram

7. POWER SYSTEM

There are many different possible design concepts for the power source of a spacecraft. Historically, the majority of spacecraft have been powered by solar panels and batteries. The solar panels of a spacecraft operate during time in the sunlight and the batteries of a spacecraft operate during time in the shade. However, some spacecraft travel too far away from the sun for solar panels to be the primary power supply, such as New Horizons. New Horizons traveled too far from the sun to make use of the typical solar power design, and utilized a Radioisotope Thermoelectric Generator (RTG) to power the spacecraft systems [81].

Typically, the process for designing a spacecraft power system begins with determining the power requirements of each subsystem needed to complete the mission. Then a power source is designed to provide an output that equals or exceeds what is required. For New Horizons, an available RTG power source was chosen as the power source and all the other systems were designed around an expected power limit from the RTG at Pluto [81]. The main consideration to be made when using RTGs is the thermocouple decay. The core material of an RTG experiences an exponential decay, based on the half-life. As the core material of an RTG decays, the power output of the RTG also experiences an exponential decay. New Horizons' power system was designed around an RTG with a plutonium-238 core, and a 240 W output at Earth, before departure. Plutonium-238 has a half-life of 87.7 years, so after the 9 ½ year trek to Pluto and the Kuiper Belt, the RTG was expected to only produce a 200 W output [81]. This made it so the instruments and subsystems of New Horizons had to only use 200 W or less.

7.1 Power Budget

A substantial amount of power is required to fully power all the systems of PROSERPINE. The total power is routed from two sources to eight subsystems including: thermal, attitude determination and control, power, command and data handling, telecommunications, propulsion, mechanisms, and the scientific instruments. The two key components of the power system for this design are the in - space propulsion system used to slow down and fall into orbit in the Pluto - Charon system, and the telecommunications system. The NEXT Ion thrusters being used to perform the deceleration requires a significant amount of power to operate. And from discussions with current New Horizons team, the data rate of New Horizons was extremely slow [42]. An increased data rate will allow for the information collected to be delivered back to Earth faster. To increase the data rate, more power needs to be put into the telecommunications system.

To determine the power requirements for PROSERPINE, the flight was broken down into five different mission stages: Earth Departure, Jupiter Flyby, Jupiter – Pluto Transit, Slowdown for Arrival at the Pluto - Charon System, and orbiting within the Pluto - Charon System. The power required during each stage of the mission was determined using the procedures outlined in Ref. 49 and Ref. 82. An interface spreadsheet was created and populated by subsystem designers, to determine the total power used by each of the spacecraft subsystems. Using the power requirements for each subsystem, a total power estimation was found, for PROSERPINE.

The power requirements for the scientific instruments were found based on research done on each specific instrument selected for the mission. The power requirement of each instrument was then added to the power interface spreadsheet, to find the total power requirement for PROSERINE. Lastly, a MS of 10% was added to the total power requirement, derived from the procedure laid out in Ref. 49. Table XI shows the subsystem power breakdown during each flight stage.

Table XI: Subsystem and Flight Stage Power Breakdown

Subsystem	Earth Departure	Jupiter Flyby	Jupiter to Pluto	Deceleration	Pluto Orbit
Propulsion	0 W	0 W	0 W	7250 W	0 W
Power	36 W	36 W	36 W	36 W	36 W
Thermal	0 W	0 W	0 W	0 W	0 W
C&DHS	25.5 W	63.5 W	25.5 W	25.5 W	63.5 W
Telecommunications	700 W	6000 W	700 W	700 W	6000 W
AD&CS	133.5 W	133.5 W	133.5 W	133.5 W	133.5 W
Payload	0 W	141 W	0 W	0 W	141 W
Mechanism	0 W	10 W	0 W	0 W	0 W
Subtotals	795 W	6384 W	795 W	8045 W	6374 W
MS (10%)	79.5 W	638.4 W	79.5 W	442 W	637.4 W
Total	984.5 W	7022.4 W	984.5 W	8487 W	7011.4 W

7.2 Downselection

Figure 23 shows where the PROSERPINE power lies on a popular power selection graph [49]. Although a mission length of 25 years is not represented on this chart, it still gives a reasonable view of what kind of power system to use based on the power requirements. The red marker shows a rough estimation of where the mission will operate. The marker on Figure 23 suggests that the power system should either be a solar array or a nuclear reactor. Following the procedure Ref. [49], solar panels were sized for the power requirements of PROSERPINE at the Pluto – Charon system, and a nuclear reactor was also sized. To use solar power at Pluto would require a solar panel with a mass of 435,000 kg, which could not be launched in a single launch vehicle. This is because as the spacecraft moves further from the sun, solar panels will become less effective, so the panels have to be even larger to produce the required power. A nuclear reactor power source will be able to provide the required power for PROSERPINE and still be able to be launched in an existing launch vehicle. So, a nuclear reactor design was selected over a solar array design.

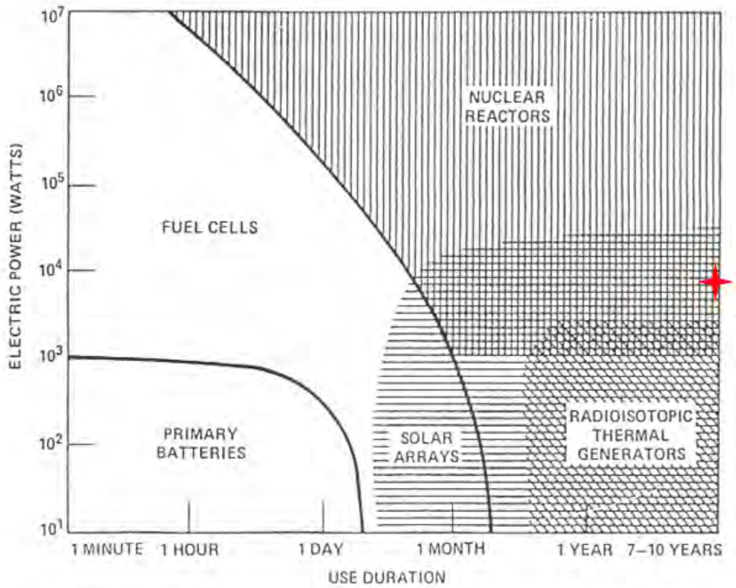


Figure 23: Power System Design Selection [49]

7.3 Kilopower Reactor Using Stirling Technology (KRUSTY)

While New Horizons, and many other missions, used RTGs, there are few that have flown with nuclear reactors. A total of 41 missions with nuclear reactors have been launched, and the United States has not launched a nuclear powered spacecraft in over 40 years. However, a new system to power deep space travel, using nuclear technology is in development at the Los Alamos National Laboratory. The Kilopower Reactor Project, which can be seen in Figure 24, is aimed at developing a kilowatt class nuclear reactor for space travel [83]. Currently, the Kilopower project is at TRL 5, as NASA reported on May 2018, and work is underway to achieve TRL 6 by 2020 [83, 84]. The Kilopower reactor utilizes a Stirling engine to generate electrical power. It uses a nuclear reactor, with a Highly

Enriched Uranium-235 core, as the heat source and the cold of space to drive the Stirling engine [69]. Then the Stirling engine provides power to the spacecraft systems. Also, control rods are used to regulate the power output of the reactor [70]. The rods will control the system, so the reactor will operate at a constant temperature, so the system will not overheat. Then the Kilopower reactor uses its radiators to bleed off any excess heat and power. And if the system starts to reach critical levels the rod is automatically inserted into the reactor by a redundant computer system, to stop the reaction and power down the system.

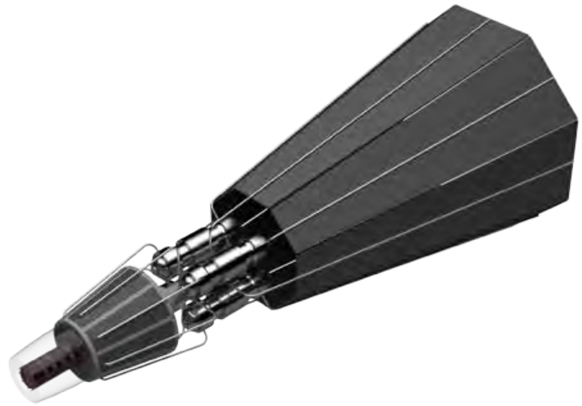


Figure 24: KRUSTY

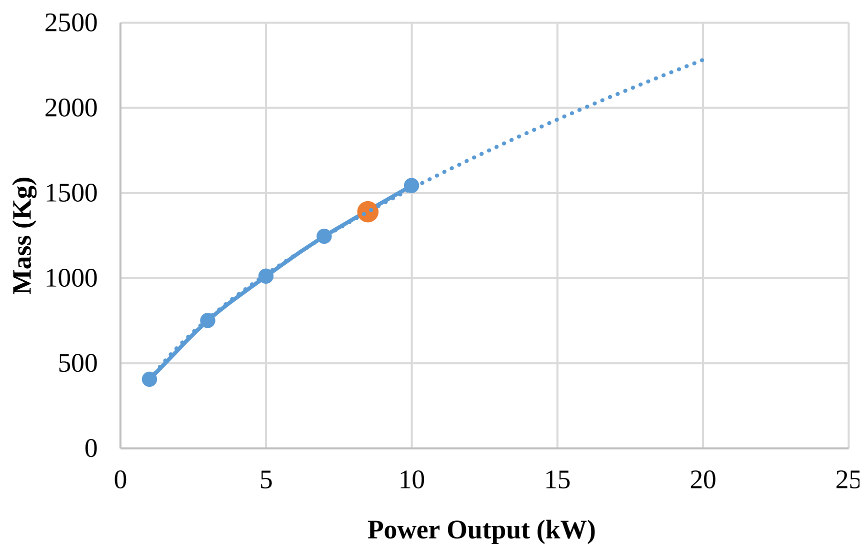


Figure 25: KRUSTY Mass Determination

For this mission, a single reactor sized to power all subsystems of PROSERPINE, outputting 8.5 kW has been selected. An 8.5 kW reactor was selected based on the power requirements mentioned previously. The sizing of the nuclear reactor was based on a joint report from the Los Alamos National Laboratory and NASA Glenn Research Center, where theoretical sizes for a 1, 3, 5, 7, or 10 kW reactor were determined [70]. To size an 8.5 kW reactor, the design parameters for the five existing theoretical designs were plotted against each other and a best fit curve was plotted over the existing design points. Then, the design characteristics for an 8.5 kW reactor were pulled from the plots. The mass relationship can be seen in

Table XII: KRUSTY Characteristics

Characteristic	Quantity
Mass	1390 kg
Stowed Diameter	1.45 m
Length	3.19 m
Radiator Area	16.66 m^2
Volume	1.75 m^3
Specific Power	6.15 W/kg

Figure 25. The equations were found as a relationship of power to find each of the design characteristics. This was seen to be a reasonable way of sizing the Kilopower reactor, because the report from the Los Alamos Research Laboratory said the Kilopower design should be scalable for a 1-10 kW output [70]. The characteristics of the 8.5 kW Kilopower nuclear reactor are defined in Table XII.

7.4 Batteries

Since the spacecraft will be carrying a nuclear reactor, the limitations of the Outer Space Treaty of 1967 need to be considered. The Treaty does not deny the use of nuclear reactors as a power source, it only denies any nation from having nuclear weapons in space [85]. Still, to avoid any international conflict, the nuclear reactor will not be turned on until after PROSERPINE has departed from the Earth. During this time, the main power will come from the installed batteries. A trade study was done to determine the optimal battery to use on PROSERPINE. The breakdown can be seen in Table XIII. Of note, solid state Li-Ion batteries are in the early stages of design, but they are expected to have a high safety rating.

Table XIII: Primary Downselection for 1.5 kW Load

Battery Type	Capacity (W-h)	Cells / Battery	Mass (kg)	Safety Index	TRL	Percent Total Mass of Spacecraft (%)
Ni-Cd	12000	1	500	High	9	11.3
Ni-H		24	218	High	9	4.93
Li-Ion		1	73	Low	9	1.64
Li-Polymer		9	45	Med	8	1.02
Solid State Li-Ion		9	16.3	High	3	0.04

A battery load of 1.5 kW was determined from the Earth departure phase of the mission, where the power requirement is 984 W, with a M.S. of 10%. During later mission stages, the power provided by theses batteries will be used as a backup power supply to run the minimum systems during an emergency. An eight hour operating time was determined based on the amount of time it will take to send an emergency signal to Earth, and then receive any information back to the spacecraft. A 1.5 kW load operating for eight hours means the battery needs to have a capacity of 12,000 W-h, as shown in Table 4. The battery power system will consist of three battery banks. Each bank will hold 500 W of power, or a capacity of 4,000 W-h. The battery system is designed in a way, so PROSERPINE will utilize all three banks to provide power to the systems, but in the event one bank loses power or stops working the other two banks can still provide enough power to the spacecraft. This redundancy was implemented to reduce the risk associated with failure common to batteries used over a long period of time. Although the batteries are sized for an eight hour operating time, during launch and earth departure they will only power PROSERPINE for half an hour. After half an hour, PROSERPINE will be far enough away from Earth to start up the nuclear reactor, and not cause any international problems or Treaty violations.

Based on Table XIII, solid state Li-Ion batteries will be installed on PROSERPINE. The solid state Li-Ion batteries were chosen because solid state Li-Ion batteries provide a significant weight savings from the traditional Ni-Cd batteries, and also has a higher safety rating than Li-Ion or Li-Polymer batteries [86, 87]. Unlike other lithium batteries, solid state Li-Ion batteries do not have an electrolyte solution, which removes any hazards of having a solution, such as dendrite growth. Dendrites can reduce the efficiency of the battery, or cause the battery to start fires [86].

Table XIV: Power Requirement for System Failure

Subsystem	Power Allotment
Propulsion	0 W
Power	0 W
Thermal Control	0 W
CD&HS	37.5 W
Telecommunications	400 W
AD&CS	133.5 W
Payload	0 W
Mechanisms	0 W
Total	628 W

The specific energy of solid state Li-ion batteries, is stated to be 735 W-h/kg [88]. Having a higher specific energy means the battery can have a higher capacity without a significant increase in mass. Solid state Li-Ion batteries also have a highly extended cycle life of nearly 1,500 cycles. In the event of a shutdown, the solid state Li-Ion batteries for PROSERPINE will be used as a backup power supply. When being used as a backup power supply, the solid state batteries may have to operate multiple times throughout the mission lifetime, so a battery that can be cycled extensively would be necessary [88]. Current solid state lithium ion batteries have a TRL of 3, and an investment will be made to achieve a higher rating by the 2031 launch date.

During later stages of the mission, the solid state Li-Ion batteries will be used as a backup power supply for the spacecraft systems, in the event of an emergency where the Kilopower reactor is shut down. The power requirement for an emergency event is 628 W. In an emergency situation, only the necessary systems would be operating. These necessary systems include: telecommunications, attitude determination and control, and command and data handling. Table XIV shows what the power requirements would be for an emergency.

7.5 Block Diagram

A block diagram of the power system is detailed below. The final block diagram (Figure 26) for the power system of PROSERPINE is based on the New Horizons power system block diagram. The diagram is split up into four sections. The first is the power sources, including the nuclear reactor, and batteries. The Kilopower reactor take commands from the shunt regulator unit (SRU) to turn on, remove, or insert the control rods to adjust the power output of the reactor. If there is excess thermal power it is routed to the dump which then routes it to the radiators. The radiators dump the excess power as heat. If the Kilopower reactor is designed to work at a constant temperature and the SRU tells the Kilopower reactor to move the control rods, to maintain the temperature of the system. Then the SRU routes the power to the bus. From here, the diagram runs into the power distribution unit. This system collects all the power and routes it towards the power bus, which sends it to the power distribution unit. In the next section, the required power is distributed to each of the subsystems. The subsystems then run to the computer system. The computer system accurately measures the power draw of each system and reports back to the SRU system. The power distribution unit then adjusts the power it sends to the subsystems as needed. The computer system also constantly monitors the status of the nuclear reactor. If there is ever an emergency, the computer can shut down the reactor and relay the information to the power management system, at which point the spacecraft will run off the power from the solid state Li-Ion batteries. The SRU, PDU, and the computer system all have stacked duplex backups that sit idle until the main systems cannot complete its task. If the computer system detects any emergency and shuts down the reactor, the battery power sources will then provide enough power for the Telecommunications system to report an emergency to earth, so programmers can try to fix the problem before complete failure.

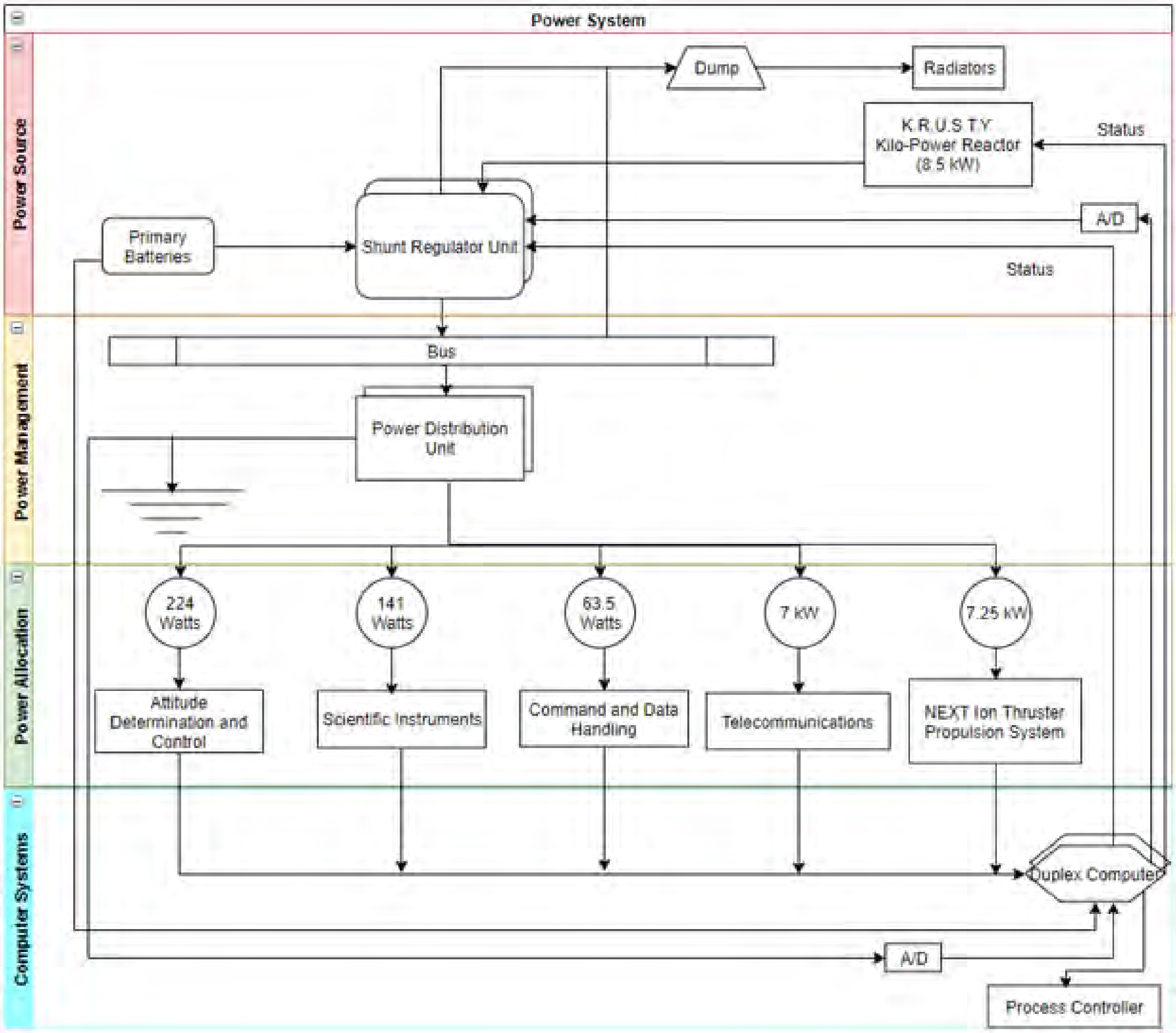


Figure 26: Power System Block Diagram

8. THERMAL CONTROL SYSTEM

The thermal control system of PROSERPINE has the purpose of controlling the onboard temperature within the operational and survivable temperature ranges for the payload and various subsystems. PROSERPINE will utilize passive thermal methods to both dissipate and contain heat generated throughout the mission. Passive methods involve thermal control on unmanned spacecraft that tend to use less electrical power, are more cost effective, and do not use up as much of the mass budget as active control subsystems. Passive methods will be utilized aboard PROSERPINE. The operational temperatures are the conditions required by the components to operate properly. The survivable and operational temperatures must be able to be maintainable at all scenarios that the mission will encounter both on the way to Pluto and during its final orbit. The block diagram detailing the thermal control system can be viewed below in Figure 27. The thermal control system components necessary to maintain the required spacecraft temperature are as follow:

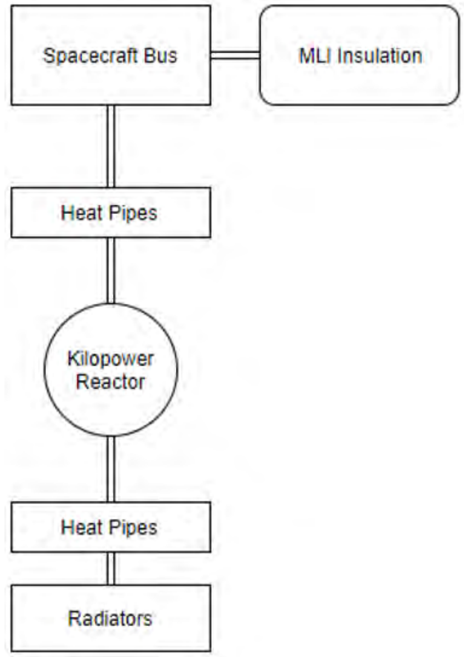


Figure 27: Thermal Control System Block Diagram

- Carbon Fiber Radiators
- Haynes 230 Alloy/Sodium Heat Pipes
- Kapton-Mylar Multilayer Insulation

8.1 Thermal Ranges

For the thermal control system, the following temperature ranges were taken into consideration for sizing of the components. These temperature ranges are the basic approximations for the operational temperatures of the payload and components of PROSERPINE. Only the operational temperature ranges were taken into consideration as these margins are more precise and if the spacecraft is able to maintain these temperatures, it will be able to maintain the survivable temperature ranges. The temperatures that drove the sizing are set by the batteries temperature range as the highest low temperature is 0 degrees Celsius and the lowest high temperature is 15 degrees Celsius. If the spacecraft is able to maintain temperatures between the range of 0 to 15 degrees Celsius, all of the components of PROSERPINE will be able to function correctly. These values are 273.15 K and 288.15 K respectively.

Table XV: Operational Temperature Ranges

Component	Range (°C)
Batteries	0 to 15
Power Box Baseplates	-10 to 50
Reaction Wheels	-10 to 40
Gyros/IMUS	0 to 40
Star Trackers	0 to 30
C&DH Box Baseplates	-20 to 60
Antennas	-100 to 100
EIS	-10 to 40
REASON	-10 to 40
SWAS	0 to 45
FGM	-100 to 175
V/SHM	-100 to 175
SWIA	0 to 45
MASPEX	0 to 45
ALICE	0 to 45
E-THEMIS	-10 to 40

8.2 Worst-Case Surface Temperature Scenarios

The maximum and minimum temperatures that the spacecraft would encounter in its journey from the Earth to Jupiter and finally to Pluto were determined to size the components of the thermal control system. For calculation simplicity, the spacecraft was approximated as a sphere. The spacecraft payload bus and the nuclear reactor were considered with two separate point mass approximations. The maximum temperature was calculated using a function of the solar constant, absorptivity of the spacecraft, planetary body IR emission, emissivity of the spacecraft, view factor of a flat plate to the body, the albedo of the body, the internal power dissipation, the diameter of the spacecraft, and the Stefan-Boltzmann constant. The minimum temperature is a simplification of the maximum temperature equation but with the solar constant being negligible. The view factor, which took the payload section of the spacecraft a modeled it as a flat plate when viewed from either Earth, Jupiter, or Pluto was calculated as a function of altitude and radius above the planetary body.

Six different scenarios were examined during the mission for the temperature analysis; these being Earth at a day and night pass, Jupiter at a day and night pass, immediately prior to deceleration to Pluto, and Pluto. The differences between day and night passes for Pluto were negligible. For these three bodies, three different internal power dissipations were taken into consideration for the nuclear reactor. A maximum power dissipation which entails a 36.8 kW thermal wattage generated by the 8.5 kW_e Kilopower reactor [70]. The second power level was a cruising speed power dissipation which entails a 4.3 kW thermal wattage generated with a 1 kW output from the Kilopower reactor. For the payload bus of the spacecraft, the thermal energy generated was taken from the electronics running dissipation. This thermal energy was 180 W.

The temperature calculations at each significant scenario the nuclear reactor of PROSERPINE will encounter can be seen Table XVI. The temperature calculations at each significant scenario the nuclear reactor of PROSERPINE will encounter can be seen Table XVII. The maximum temperature that PROSERPINE will endure will be at Jupiter during a day pass under full power expenditure of the Kilopower reactor with a temperature of 455.7 K. The minimum temperature occurs at 6.13 billion km from the sun right before the beginning of the slowdown to arrive at Pluto. This scenario incorporates the cruising power dissipation from the Kilopower reactor and results in a temperature of 267.4 K. These temperatures are what are used for sizing the components of the thermal control system. The thermal wattage was found using data provided for the NASA Kilopower reactor [70].

Table XVI: Surface Temperatures during Mission (Nuclear Reactor)

Scenario	Power Level (kW)	Q Internal (kW)	Temperature (K)
Earth (Day Pass)	1000	4.3	312.3
Earth (Night Pass)	1000	4.3	281.5
Jupiter (Day Pass)	8500	36.8	455.7
Jupiter (Night Pass)	8500	36.8	455.4
Beginning of Deceleration for Pluto Arrival	1000	4.3	267.4
Pluto (Day Pass)	8500	36.8	455.4
Pluto (Night Pass)	8500	36.8	455.4

Table XVII: Surface Temperatures during Mission (Payload)

Scenario	Q Internal (W)	Temperature (K)
Earth (Day Pass)	200	262.2
Earth (Night Pass)	200	196.39
Jupiter (Day Pass)	200	133.85
Jupiter (Night Pass)	200	123.67
Beginning of Deceleration for Pluto Arrival	200	133.84
Pluto (Day Pass)	200	123.98
Pluto (Night Pass)	200	123.76

8.3 Heat Radiation

To size the radiator, the worst case hot temperature that PROSERPINE can encounter was used. This temperature found using function of the Stefan-Boltzmann constant, area of the radiator, and temperature of the spacecraft. The area of the radiator could be solved for using this Stefan-Boltzmann analysis.

Carbon fiber radiators were chosen to be used for PROSERPINE as this is what is currently used for the NASA Kilopower reactor. Carbon fiber radiators performance increases as the temperature that the radiators are encountering increases. Therefore, at the maximum temperature that the nuclear reactor of PROSERPINE will be encountering, these Carbon fiber radiators emissivity will be 0.76 [89]. The area sized for the radiators of PROSERPINE is 16.66 m^2.

8.4 Heating

To size the amount of heat needed for the mission, the worst case cold temperature that the payload bus of PROSERPINE will encounter was used. To mitigate heat loss, Kapton-Mylar multilayer insulation is to be used on PROSERPINE. 30 layers will be used due to the low effective emittance of 0.00031 W/(m-K), and conductivity of 0.005 W/m^2 [49].

Due to the mass budget, a heater will not be used. The nuclear reactor will be able to provide sufficient waste energy depending on the power level that it is outputting. The thermal energy required to be transferred to the payload bus was 4584 W. This was determined as a function of the required inner temperature, the outside surface temperature, the thermal energy being radiated, and the thermal energy being produced from the inside of the payload bus. The thermal energy required to enter the spacecraft bus is to be equal to the thermal energy being radiated from the spacecraft bus. This is to ensure thermal equilibrium and that the temperature within the spacecraft payload bus remains constant.

The waste heat from the nuclear reactor will be transported to the main bus using Haynes 230 alloy/Sodium heat pipes. Haynes 230 alloy heat pipes were chosen due to their high creep resistance and service temperature strength. Heat pipes are a way to transfer heat throughout the spacecraft through capillary action without the use of pumps. This allows the TCS to remain completely passive. Using a working fluid of Sodium, the heat generated causes the liquid to vaporize and then travel through the heat pipes to transfer heat throughout the spacecraft as it condenses [70]. The weights for components were calculated and tabulated in Table XVIII.

Table XVIII: Mass Budget

Component	Weight (kg)
Radiator	38
Heat Pipes	8.4
MLI	14.6
Total	61

9. COMMAND AND DATA HANDLING SYSTEM

Table XIX: C&DH Mass and Power Breakdown

Component	Mass	Power
Solid State Recorders (SSRs)	4.34 kg	17.4 W
Vehicle Management Computers (VMCs)	4 kg	20 W
Data Links, Misc.	34.5 kg	26 W
Total	42.84 kg	63.4 W

The CD&HS of PROSERPINE is responsible for managing engineering and science data, data compression and processing, commanding subsystems, and interfacing with the telecommunications system to receive and send data. The C&DH system is comprised of one primary IEM and one redundant IEM, as well as redundant interfaces between instruments and subsystems with the IEMs. Each IEM includes multiple processors, solid state recorders (SSRs), and an ultra-oscillator (USO). This is modeled after the C&DH architecture used in the New Horizons mission [90]. Additionally, flight computers will be able to be reprogrammed during flight. This functionality saved the New Horizons mission after an onboard failure. The block diagram detailing the C&DH architecture is shown in Figure 28. Of note, this block diagram shows the interconnections with one IEM. A redundant IEM is connected in a similar manner. The mass and power values for the C&DH system is shown in Table XIX.

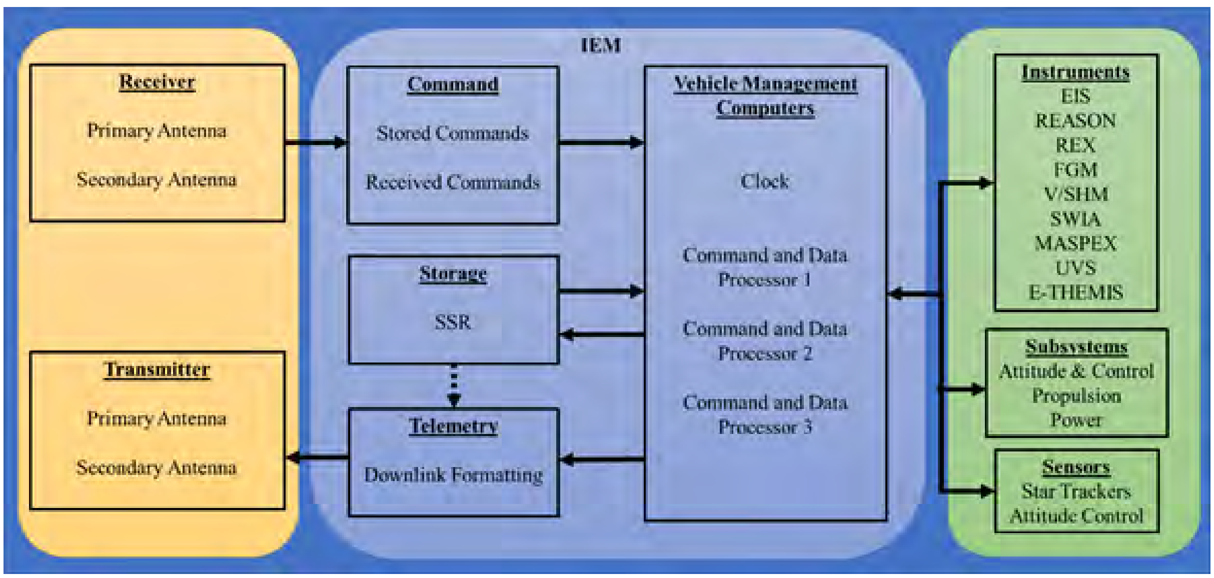


Figure 28: C&DH Block Diagram

9.1 Vehicle Management Computers

Three processors will be used in each IEM. A majority based voting structure will be implemented to ensure command accuracy and reliability. Boeing Chiplet processors will be used for this mission. The Boeing Chiplet is under development as part of the High Performance Spaceflight Computing (HPSC) NASA program [91], the goal of which is to "develop a next-generation flight computing system addressing the computational performance, energy management, and fault tolerance needs of NASA missions through 2030 and beyond." The Boeing Chiplet processor is a dual quad-core processor with functionality to adjust performance and power consumption as well as parallel computing tasks [92]. Processing memory requirements are presented in Table XX. The processors will also decode received information and encode information for transmission.

Table XX: C&DH Data Breakdown

Subsystem	Task	LOC (ADA)	Memory, Code Words	Memory, Data Words	Code Size (kb)	Data Stored (kbits)
C&DHS	Telemetry Processing	1350	13500	3375	1080	270
	Command Processing	2400	24000	6000	1920	480
	Polling / Multiplexing	1200	12000	3000	960	240
	Formatting	600	6000	1500	480	120
	Configuration Table	975	9750	2438	780	195
Telecommunications	Uplink Processing	900	9000	2250	720	180
	Downlink Processing	600	6000	1500	480	120
AD&CS	Attitude Determination	1500	15000	3750	1200	300
	Attitude Control	2400	24000	6000	1920	480
	Ephemeris Processing	975	9750	2438	780	195
Articulation	High Gain Antenna	1200	12000	3000	960	240
Fault Protection	Safing	1500	15000	3750	1200	300
	CDS Fault Protection	1800	18000	4500	1440	360
	ACS Fault Protection	11500	115000	28750	9200	2300
Operating System	Operating System	1000	10000	2500	800	200
Utilities	Utilities	2200	22000	5500	1760	440
	Totals	32100	321000	80250	25680	6420
	Total RAM Required (Mb)	12.84				

9.2 Solid State Recorders

With the antenna configuration, the maximum downlink rate is 12 Mbps. Following, the maximum downlink data sent over a one year period is ~21 TB, assuming 8 hours each day on the Deep Space Network (DSN). The maximum calculated instrument data rate was 5.26 Mbps. Details on instrument data rates are presented in Table XXI.

Following, the maximum calculated instrument data collected over a one year orbit was ~21 TB. The instrument data rate was matched to the maximum downlink rate by specifying how many frames the EIS captures per day. Because these images are 4k resolution, they contribute the most to instrument data rate. The maximum data rate of 5.26 Mbps assumes one EIS frame per minute. Thus, each IEM will have SSR storage capacity of 21 TB. Image data will be compressed using CCSDS 120.0-G-3 lossless data compression, with a typical compression ratio of 2:1 for imaging data [93]. It is reasonable to assume that after data compression, all instrument data collected can be transmitted.

Southwest Research Institute (SRI) has off-the-shelf SSRs up to 12 TB which are radiation hardened, adaptable for multiple input and output data interfaces, and have CCSDS formatting built in to directly downlink from the SSR to the transmitter. It is not unreasonable to assume that by the anticipated launch date of 2031 a single SSR will have 21 TB capacity. If this is not the case, multiple lower memory SSRs will be used in conjunction to meet the 21 TB requirement.

Table XXI: Science Instruments Data Rates

Instrument	Bit Rate (bps)	Hours Operating per Day	Total Data Collected over 1 Year (MB)
Europa Imaging System (EIS)	1.60E+06	24	6.31E+06
Ice Penetrating Radar (REASON)	3.20E+05	24	1.26E+06
Submillimeter Wave Array Spectrometer	2.33E+06	24	9.20E+06
Magnetometer (FGM)	2.41E+00	24	9.50E+00
Magnetometer (V/SHM)	2.41E+00	24	9.50E+00
Solar Wind Ion Analyzer (SWIA)	7.04E+02	24	2.78E+03
Mass Spectrometer for Planetary Exploration (MASPEX)	1.40E+04	24	5.52E+04
Ultraviolet Imaging Spectrometer/Spectrograph (UVS)	2.80E+00	24	1.10E+01
Europa THermal Emission Imaging System (E-THEMIS)	1.00E+06	24	3.94E+06
Total	5.26E+06		2.08E+07
Total (Alternate Units)	5.26 Mbps		20.77 TB

10. TELECOMMUNICATIONS SYSTEM

A reliable and functional telecommunications system is the backbone of a spacecraft design. During the design of the PROSERPINE telecommunications system, many factors were taken into consideration in order to determine the optimal communications system. The following requirements were imposed on the communications design. PROSERPINE must be able to communicate with Earth at 50 AU maximum distance and operate on no more than 6,000 W. It must maintain communications with Earth with 700W during slowdown phase of mission. It must achieve an uplink rate of 144 kbps or greater and achieve a downlink rate of 12 Mbps or greater at Pluto. Finally PROSERPINE must use the DSN. The Radio Frequency (RF) communications system for PROSERPINE was then designed to meet or exceed the requirements.

10.1 Configuration

The telecommunications system for PROSERPINE will use both X-band and S-band frequency. The frequencies selected will ensure compatibility with the DSN 70-m or 34-m antenna. However, due to age and maintenance cost, all 70-m DSN antennae are to be decommissioned as part of the Deep Space Network Aperture Enhancement Project (DAEP) [94]. The DAEP will replace all 70-m antennas with arrays of four 34-m beam waveguide (BWG) antennas by 2025 at all DSN locations [94]. Because the arrayed antennas are designed to perform equally to a single 70-m dish, the RF communications system is designed to be compatible with the future DAEP system. The costs associated with the DAEP are assumed to be comparable to the 70-m network.

The telecommunications system will use one primary antenna operating at either X-band or S-band frequencies. X-band will be used when high data transfer rates are needed whereas the S-band will be employed as a redundant backup system. Although Ka-band was considered, lack of space-worthy high-power transmitters lead to consideration of other frequency options. Should 3,000 W Ka-band Traveling Tube Wave Amplifiers (TWTA) or Solid-State Power Amplifiers (SSPA) be developed by the launch date, the design team recommends the use of Ka-band for primary communications with an X-band redundant system.

The design of the RF communications system drew on heritage from the New Horizon spacecraft [90]. The primary antenna is a 4.2-m parabolic antenna that incorporates both a cassegrain feed system for X-band transmissions and a front feed system for S-band communication. The 4.2-m solid antenna is the core of a deployable 12.6-m antenna. A low gain S-band horn located on the front of the S-band feed is used for near Earth command and data handling up to 1 AU. To comply with the Space Frequency Coordination Group (SFCG) and the CCSDS recommendations for, high data rate, bandwidth-efficient modulation, PROSERPINE will use Gaussian Minimum Shift Keying with a time-bandwidth product of 0.5, (GMSK BTs=0.5), with a precoding modulation scheme [95]. A GMSK BTs=0.5 modulation scheme allows for efficient bandwidth utilization and low E_b/N_0 requirements at a bit error rate of 10-6 dB [95].

While at Pluto, the communications system will operate at 6000W for maximum downlink data transfer while the thrusters are not in use. The maximum downlink data rate needed for the transmission of science, HD photos and HD video data is 12,000 kbps. During transit to Pluto or when the ion thrusters are active, data will be transmitted with 700W and at a minimum rate of 198 kbps using the X-band system. In the event of a failure in the nuclear reactor, the telecommunications system of PROSERPINE will be able to operate and communicate with Earth at a data rate of 8 kbps using 435W of power for the S-band system on the 4.2-m antenna or 29W using X-band frequencies. If the antenna has deployed to 12.6-m, the power required for emergency data transmissions is on X-band communications is 4W. Figure 29 shows uplink data rates using DSN 34-m arrayed antenna. Data rates for different mission profiles are shown in Figure 30 and Figure 31. The data transfer rates at arrival at Pluto and at apoapsis of Pluto shown in Figure 32. The data shows that the maximum and minimum data transmission rates of PROSERPINE meet or exceed all requirements for the system even in the worst-case scenario at the apoapsis of Pluto. Table XXII show the link budget for each antenna.

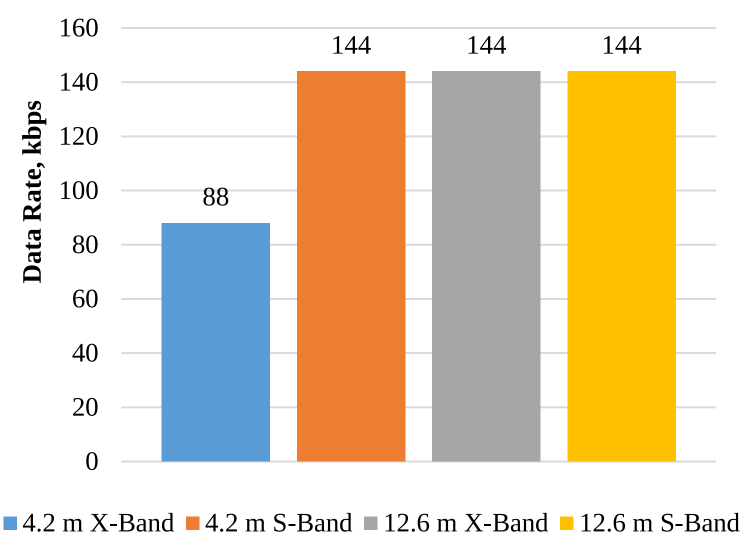


Figure 29: Command and Control Uplink Capabilities with DAEP 34-m Antenna Arrays

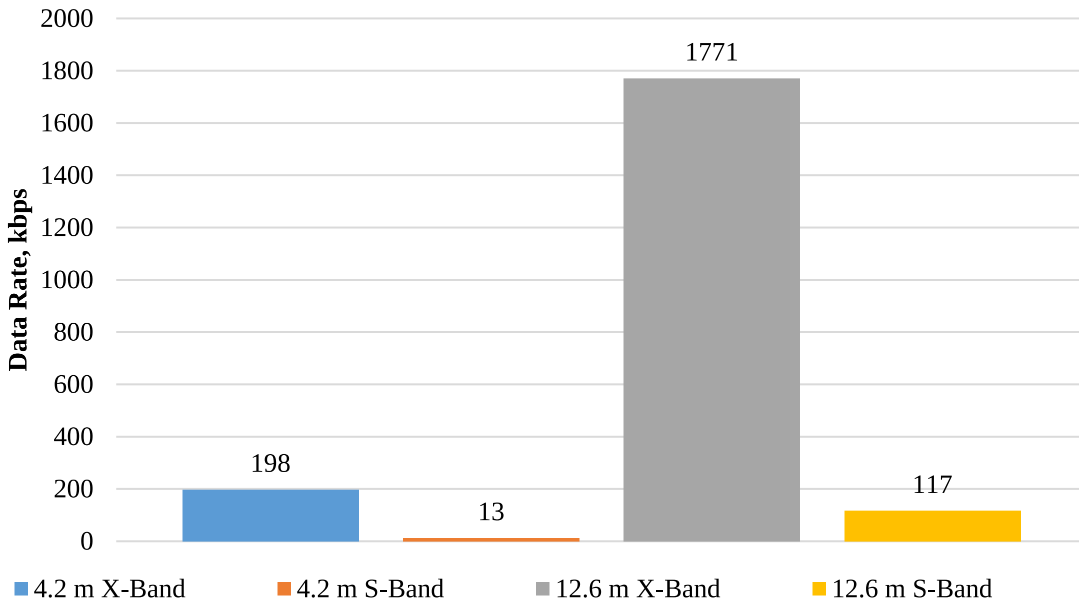


Figure 30: Transit Science and Engineering Downlink Capabilities with 700 W Transmission to DAEP 34-m Antenna Arrays

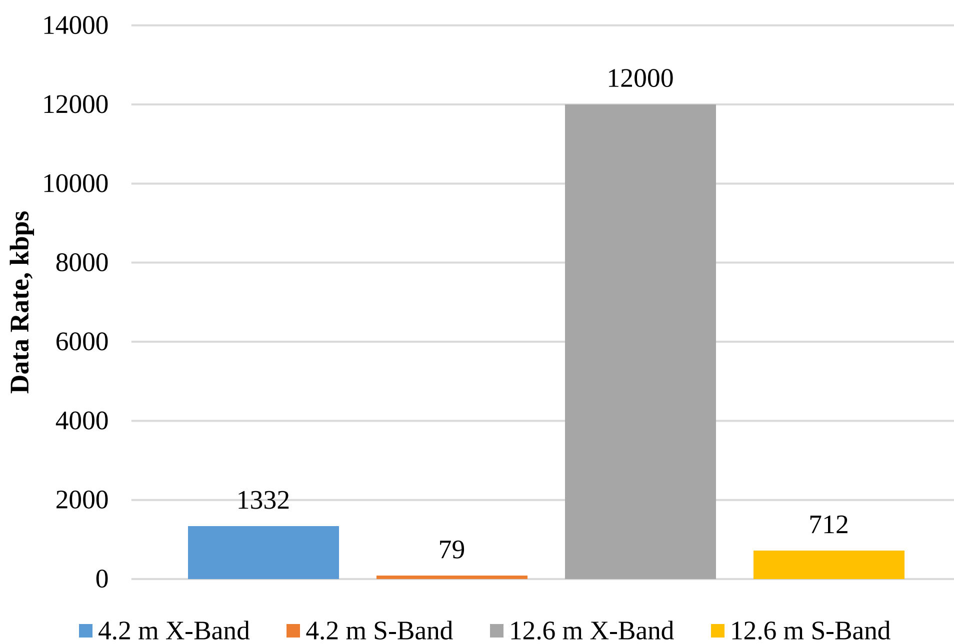


Figure 31: Orbiting Science and Engineering Downlink Capabilities with 6000 W Transmission at 50 AU to DAEP 34-m Antenna Arrays

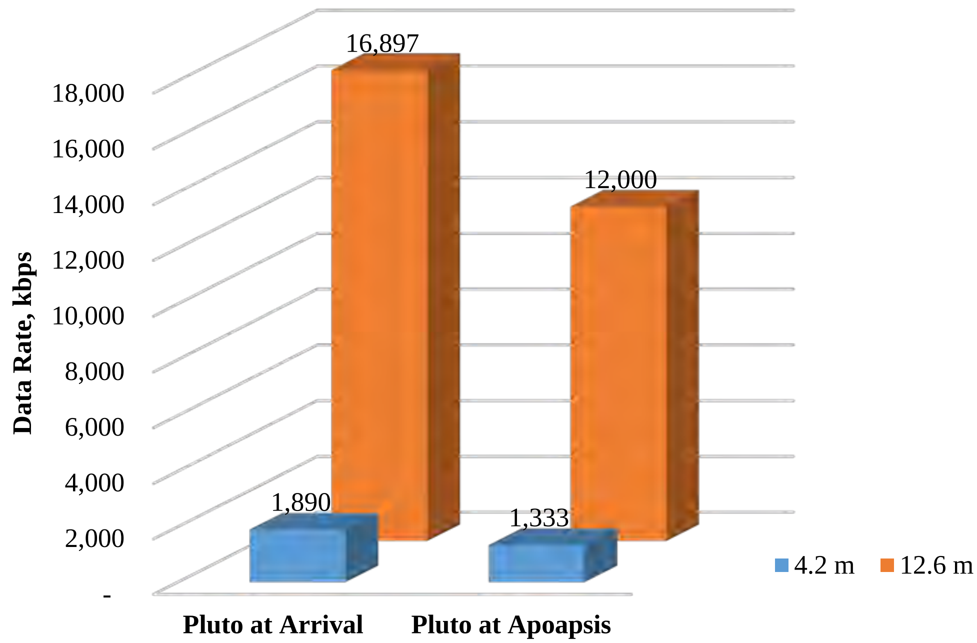


Figure 32: X-Band Transmitting Data Rates for Each Antenna Diameter at Arrival at Pluto and at Apoapsis of Pluto, 6000W at 3 dB Margin

Table XXII: 6000 W Link Budget

Item	Units	4.2-m Diameter		12.6-m Diameter		0.56-m Diameter
		HGAX	HGAS	HGAX	HGAS	LGH
Frequency	GHz	8.425	2.295	8.425	2.295	2.295
Transmitter Power	W	6000	6000	6000	6000	6000
Transmitting Antenna Gain (Net)	dBi	49.8	38.5	59.3	48.1	19.6
Equivalent Isotropic Radiated Power	dBW	87.5	76.3	97.1	85.8	57.4
Space Loss	dB	-308.4	-297.1	-308.4	-297.1	-274.5
Receiving Antenna Gain	dBi	74.3	63	74.3	63	63
Data Rate	kbps	1333	79	12000	712	144
E_b/N_0	dB	7.7	7.7	7.7	7.7	7.7
Required E_b/N_0	dB	2.6				
Implementation Losses	dB	-2.1				
Margin	dB	3	3	3	3	3

10.1.1 High Gain Horn

For Earth communication at Pluto, a primary antenna (HGAX) will transmit and receive X-band frequencies. The HGAX system contains a cassegrain feed system, dichroic subreflector, a 4.2-m spun aluminum parabolic reflector core with a deployable outer section. Once deployed, the antenna will have a diameter of 12.6-m. The front subreflector will extend normal to the parabola so that it is positioned at the focus of the antenna. The furlable section of the antenna will store out of the way of the inner 4.2-m reflector so that in the event of a deployment malfunction, the inner core reflector can act as the primary antenna. PROSERPINE is to launch with the antenna retracted and deploy the antenna once it has traveled 1 AU. Because the spacecraft is spin stabilized during transit, the antenna is mounted in line with the spinning axis. A two-axis gimbal will mount at the base of the reflector to enable 0.01 degree pointing accuracy.

For redundancy, the high gain antenna system (HGAS) will also transmit and receive S-band communications. S-band communications will only be used in the event of X-band system failure or gain degradation due to rain attenuation on Earth. The HGAS will utilize the same deployable reflector as HGA1 but will feature a front feed system. When the antenna is deployed, the front feed system will extend normal to the antenna so that is positioned in the focus of the parabola. The dichroic subreflector of HGAS is only reflective in X-band frequencies allowing the S-band signal to pass through the subreflector. With this setup, the antenna can act as both a cassegrain and front feed system at the same time [96]. The configuration for HGAX is shown in Figure 33 below. Figure 34 shows the layout for HGAS.

10.1.2 Low Gain Horn

During launch and up to 1 AU from Earth, all communications will be completed with the low gain horn, (LGH). Using S-band frequency, the LGH is capable of receiving and transmitting 144 kbps. The LGH will be mounted facing outward on the front feed system of HGAS and operate on the S-band frequency. Two, single-pole double-throw (SPDT) relays will attach to the output of each diplexer to allow the use of the LGH in the event of a malfunction of any transmitter. The LGH configuration is shown in Figure 34.

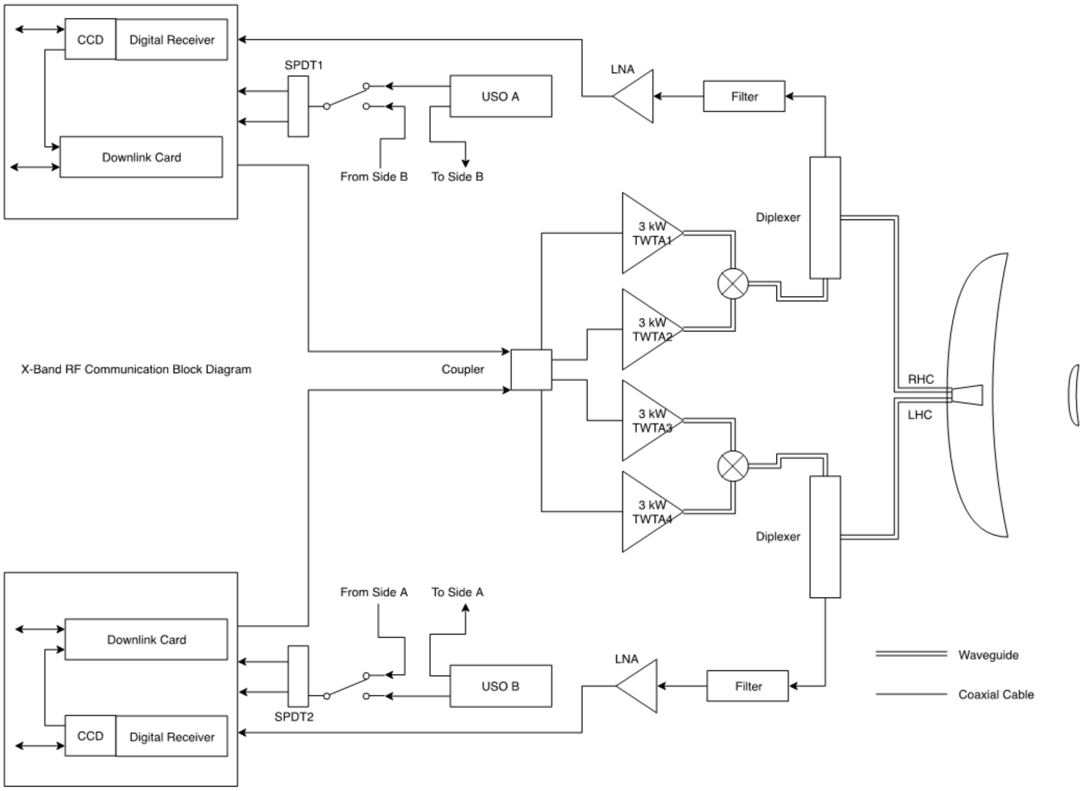


Figure 33: HGAX Block Diagram

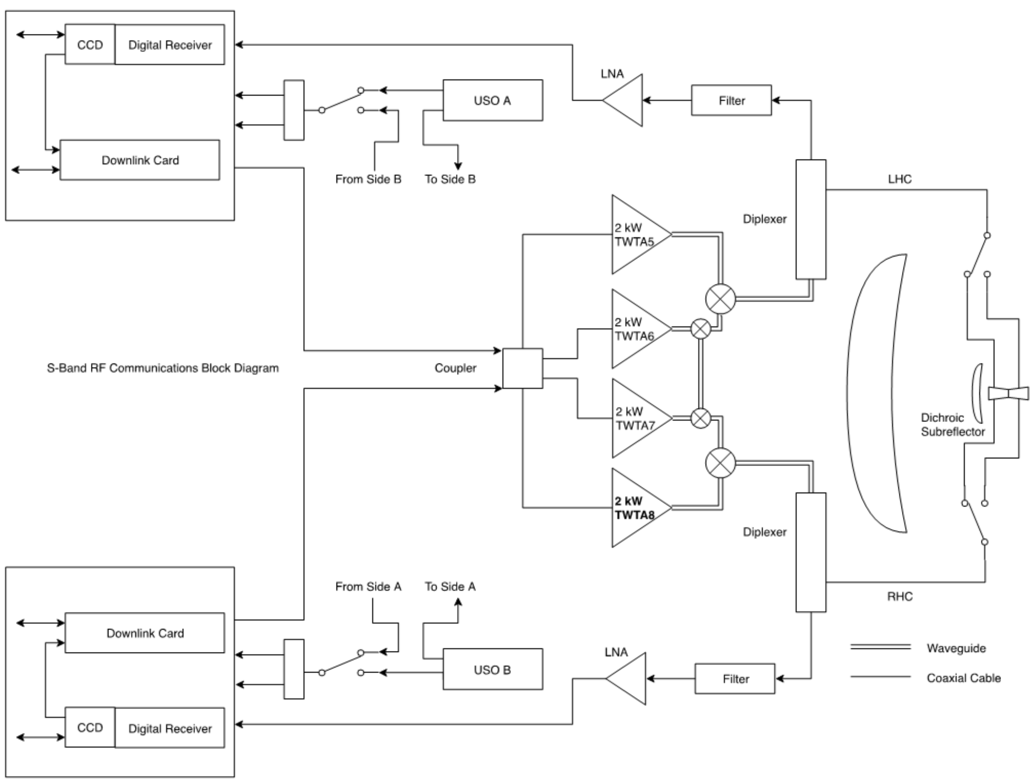


Figure 34: HGAS and LGH Block Diagram

10.1.3 RF Amplifier Enclosures

All amplifiers and transmitters for the HGAX, HGAS and LGH system are mounted directly to the back of the high gain parabolic reflector. The HGAX system will be equipped with four 3kW Teledyne MTI 3048Q pulse TWTA units. Only two TWTAs will operate at any one time allowing for either 6kW left hand circular (LHC) polarization, right hand circular polarization (RHC) or both LHC and RHC simultaneously. The HGAS system is equipped with four 2kW Teledyne MTG 3041L2 TWTAs with a maximum of three operating simultaneously. Both the HGAX and HGAS use waveguide outputs from the TWTAs and are connected to either the LHC or RHC diplexer. The diplexers then receive the TWTAs and passively execute frequency-domain multiplexing across the signal so that the same antenna can send and receive signals simultaneously [97]. For the HGAX system, the diplexers output into waveguide channels that feed to the cassegrain horn. The HGAS and LGH system diplexers output to coaxial cables that lead to a switch to select either the HGAS or LGH.

10.1.4 Transponders

PROSERPINE is to have one primary transponder and one redundant transponder with only one being powered at a time. To comply with SFCG recommendation 7-1R5, the transponder will use a turnaround ratio of 749/880 and 221/240 [98]. The transponders main functions are X-band and S-band receiver, X-band and S-band excitation, command and data handling, differential one-way ranging, telemetry modulation, and command detector. The transponders are each connected to one separate ultra-stable oscillator (USO) for time keeping and to ensure an accurate reference frequency signal.

11. ATTITUDE DETERMINATION AND CONTROL SYSTEM

Within this chapter, the AC&DS for PROSERPINE will be discussed.

11.1 Limiting Factors

The RFP does not explicitly outline limiting factors for the AC&DS [1]. However, there were a few derived limiting factors from the propulsion system, communications system, and trajectory. As with other systems, there are additional risk mitigation tactics implemented. The communications system and scientific instruments depend on the pointing accuracy of the spacecraft. To reach the data rate desired for the spacecraft, it is necessary for the AD&CS system to provide a pointing accuracy below 0.1°. The AD&CS dictates the orientation of the spacecraft in an inertial frame. This is necessary for all maneuvers and counteracting external torques perturbing the spacecraft.

11.2 Weight and Balance

The AD&CS is closely linked with the configuration of PROSERPINE. Location of center of mass (CM) and the various moments of inertia dictate the level of handling of the spacecraft. To calculate these values, the location of the various components were determined. Figure 35, Figure 36, Figure 37 show the location of CM for the components for the 3-body axes of the spacecraft. Table XXVI details the identification numbers for the components. For further contextual information, an isometric view of PROSERPINE is presented in Figure 38. A summary of the CM location and moments of inertia is presented in Table XXV.

11.3 Attitude Determination

The level of data sent back to Earth, or the lack there of, is directly linked to the precision antennas are able to point towards Earth. The final configuration utilizes gimbals for the two antennas. This drastically increase the accuracy of the pointing angle. Additionally, the scientific instruments rely on the orientation of the spacecraft with respect to the intended target.

11.4 Attitude Control

The AD&CS was sized for required thrust, torque, and momentum saturation. These parameters were evaluated for external disturbances, spacecraft generated torques, and intended slew maneuvers, thrusting, and orbit transfers.

11.4.1 External Disturbances

The greatest external disturbances expected for PROSERPINE are:

- Gravitational-Gradient
- Magnetic Torque
- Gravitational Torque
- Aerodynamic Drag

Further, the greatest amount of experienced torque is expected at our key mission events. In essence, the control system is sized by the Earth departure, Jupiter Fly-by, and Pluto-Charon orbit. Table XXIII summarizes these results.

Table XXIII: Total Torque Values

	Earth	Jupiter	Pluto
T_s, Solar Torque Earth (N-m)	3.21E-04	1.19E-05	2.10E-07
T_m, Magnetic Torque (N-m)	6.20E-05	3.21E-08	5.66E-11
T_g, Gravitational Torque (N-m)	3.18E-02	1.51E-10	9.19E-06
T_{DF}, Drag Force Torque (N-m)	4.49E-04	1.25E-03	0.00E+00
T, Total Torque During Condition (N-m)	3.26E-02	1.26E-03	9.40E-06

These calculated torque values were applied to the time between momentum dumping to get maximum angular momentum saturation, presented in Table XXIV.

Table XXIV: Maximum Saturated Angular Momentum

Angular Momentum Jupiter	Nms	153.7
Angular Momentum Pluto	Nms	3.62

11.4.2 Internal Disturbances

PROSERPINE will have to counteract disturbances created internally of the spacecraft. These include CM offsets and fuel slosh. Fortunately, the way PROSERPINE is configured these values are an order of magnitude lower than external disturbances. Therefore, these torques were not used to size the system.

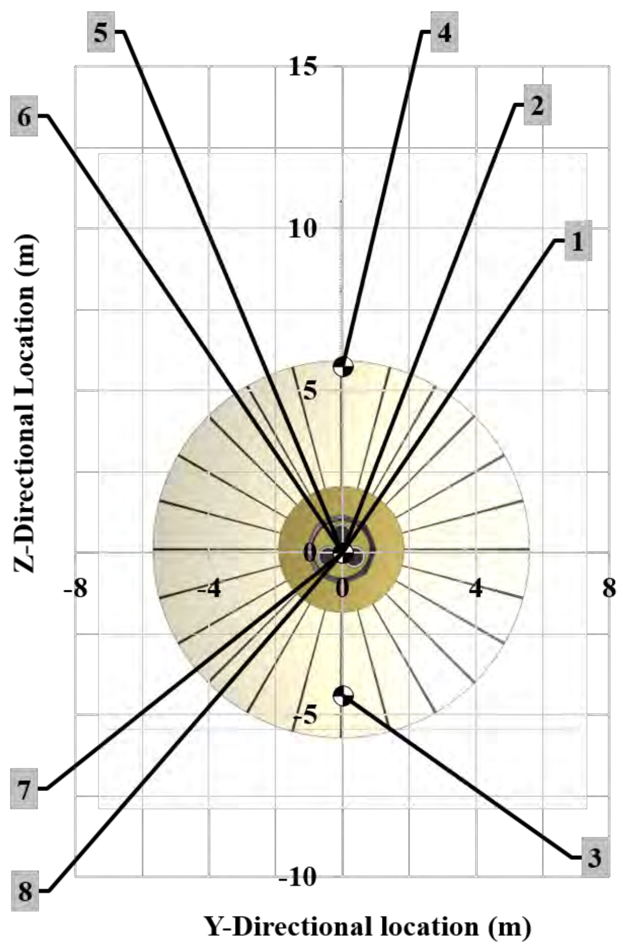


Figure 37: YZ View of CM Locations of PROSERPINE (Scale 1:250)

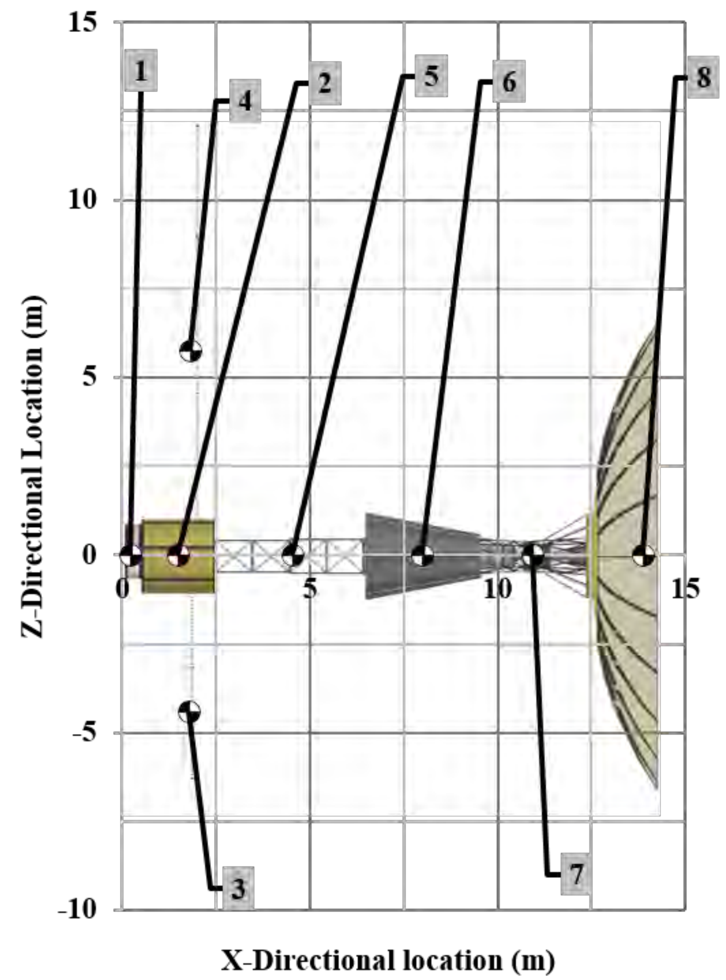


Figure 36: XZ View of CM Locations of PROSERPINE (Scale 1:200)

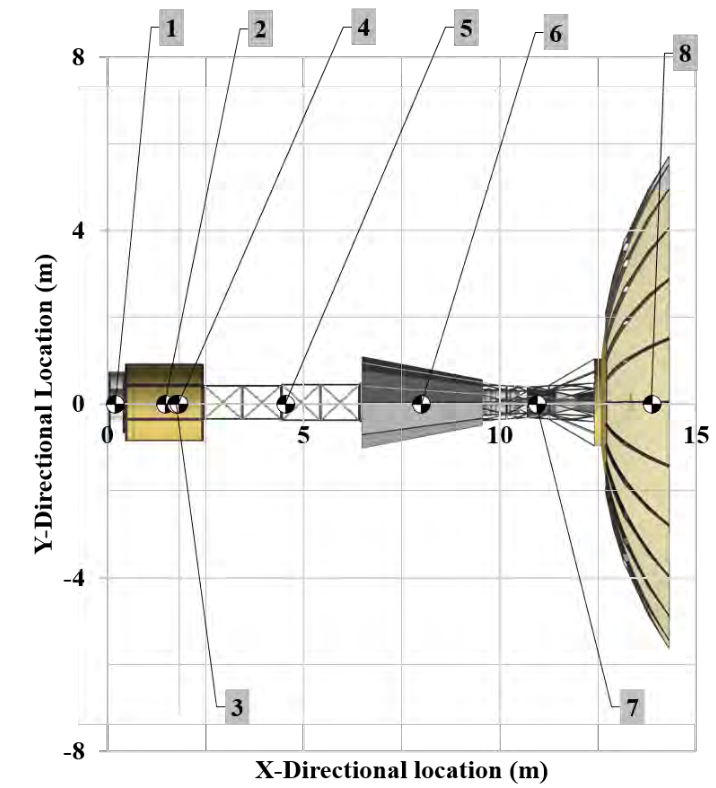


Figure 35: XY View of CM Locations of PROSERPINE (Scale 1:200)

Table XXVI: PROSERPINE Component Identifications

Identification	Group/Unit
1	NEXT-C
2	Main Bus
3	REASON Complex
4	Magnetometer Complex
5	Truss Complex
6	Radiator
7	Reactor
8	Antenna System

Table XXV: Center of Mass and Moments of Inertia

Center of Mass Position (m)	X	9.88E+00
	Y	0.00E+00
	Z	8.16E-03
Second Moment of Inertia ($kg\text{-}m^2$)	I_{xx}	5.87E+03
	I_{yy}	1.07E+05
	I_{zz}	1.05E+05

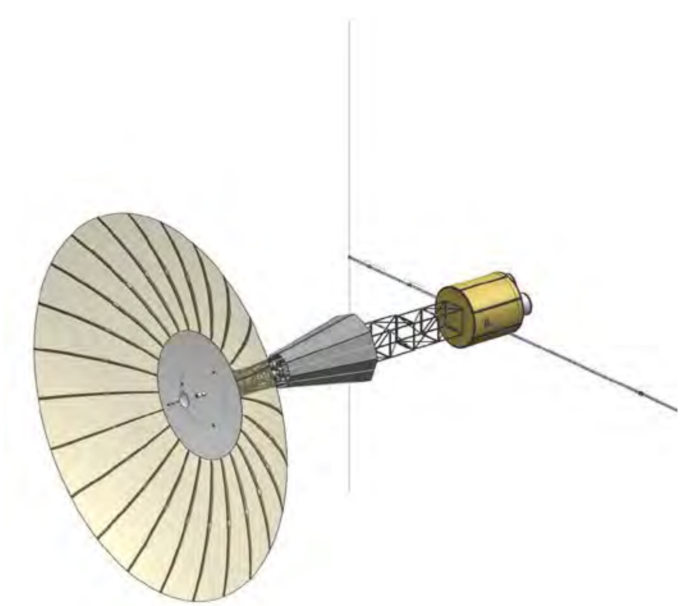


Figure 38: Isometric View of PROSERPINE

11.4.3 Slew Maneuvers, Thrusting, and Orbit Transfers

The AD&CS system will have to reorient the spacecraft periodically through various slew maneuvers. Examples of this are when the spacecraft needs to position itself to get readings for the various instruments on board. Another example would be orienting its antennas to Earth. The time this takes affects both the measurement reading and the data transferred back to Earth. This was an important consideration when selecting the control method. Additionally, PROSERPINE must maintain control during thrusting through propulsive burns. An inability to do so can lead to trajectory misalignment and ultimately mission failure.

With primary propulsion in line with the body axis of PROSERPINE and the moment of inertia about this axis being relatively low, the AD&CS was not sized by control required of thrusting. Control induced torque, which is linearly proportional to the angular acceleration of the spacecraft, was stressed when selecting hardware for the AD&CS.

11.5 Hardware

11.5.1 Downselection

A trade study was conducted of spacecraft with mission profiles similar to the trajectory planned for PROSERPINE [99-103]. Configurations were weighted more favorably for spacecraft with later launch years. This was done to ensure that the technologies aboard PROSERPINE were up to date for the projected 2031 launch. Additionally, the closer the studied mission profiles resembled the operations of PROSERPINE, the more credence were put on their specific configurations.

For attitude determination, the trade study, supplemented with system guidelines, generated the following list of hardware items for attitude determination and navigation [49, 82]:

- Two Star Trackers
- Three Sun Trackers
- Two IMUs[104]

PROSERPINE only necessitates one operational unit from each category. However, as this system is mission critical, the number of units were increased. The varying levels of redundancy were chosen with respect to the results of the trade study. The star tracker captures ten images of the universe every second. These images are compared to a catalogue of roughly 3,000 stars to determine the orientation and position in an inertial frame [103].

For attitude control, the trade study, supplemented with system guidelines, generated the following list of hardware items for attitude control [49, 82]:

- Four Reaction Wheels
- Six Tile 5000 Xenon Ion Thrusters [105]
- Gimbaled NEXT Ion Engines used for Primary Propulsion

The angular acceleration for the 3-body axes were calculated and are presented in Table XXVII.

Table XXVII: Angular Acceleration Rates of PROSERPINE

Maximum Angular Acceleration due to Reaction Wheel Assembly about:		
X-Axis, α_x (rad/s)	Y-Axis, α_y (rad/s)	Z-Axis, α_z (rad/s)
3.41E-04	1.86E-05	1.90E-05

11.5.2 Budget

Table XXVIII presents and totals the mass and power for the AD&CS of PROSERPINE.

Table XXVIII: Mass and Power Budget for AD&CS

Unit	Quantity	Unit Mass (kg)	Total Mass (kg)	Power Intake (W)
Star Trackers	2	5.5	11.0	7.5
LN-2000S: Inertial Measurement Unit	2	0.75	1.5	12.0
Sun Trackers	3	2.0	6.0	1.5
Reaction Wheel Assembly	4	5.0	20.0	60.0
TILE 5000: Xenon Ion Thruster	6	1.1	24.0	90.0
		Total	62.5	171.0

11.5.3 Integration

The AD&CS is integrated to PROSERPINE in a way that communication between units happens both upstream and downstream. Specifically, all of the sensors on board will send and receive data from the vehicle management computer. This in turn will send and receive information from propulsion and control system, power system, and command and data. This was done with reference to how other spacecraft with similar mission profiles were configured [106]. This is visually presented in the block diagram seen in Figure 39.

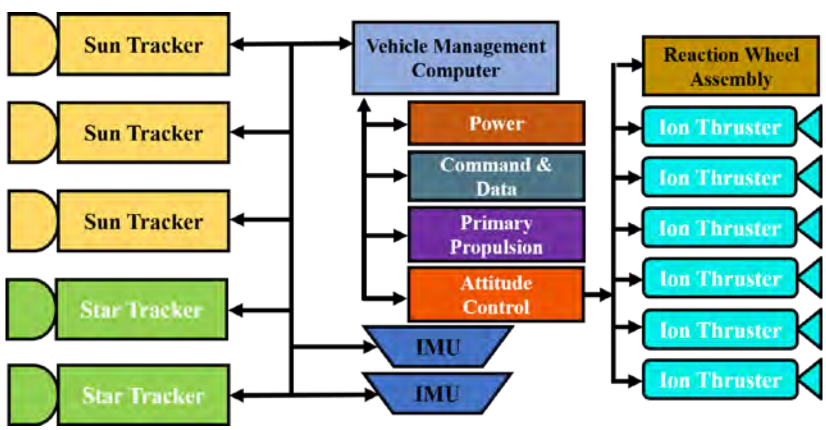


Figure 39: Block Diagram for AD&CS Integration

12. COST ANALYSIS

Within this chapter, the cost analysis for PROSERPINE will be discussed. The fundamentals of cost analysis include the development of preliminary cost driving elements followed by a Work Breakdown Structure (WBS), an organized table used to categorize and normalize costs.

12.1 Assumptions

Parametric estimating uses a series of mathematical relationships that correlate cost to physical parameters that are historically known to influence cost. These mathematical relationships are known as Cost Estimating Relationships (CER). The advantage of a parametric estimating model lies in the top-down approach. System requirements and design specifications are all that are required to complete the cost estimation. The use of parametric models implies the following assumptions [82]:

- Future costs will reflect those of historical trends to some degree.
- Program costs are variables that cannot be predicted with 100% accuracy.
- Influence of all other variables other than the cost drivers is estimating error.
- CERs are simplifications of the relationship they are emulating.
- Constant year dollars should be used for consistency and then converted for then-year dollars by using an inflation factor.

12.2 Inflation Estimation

A table of inflation factors relative to the year 2000 based on projections by the office of the Secretary of Defense was given in Ref. 82 from 1980 to 2020. To account for the projected date of mission operation, the inflation factors were plotted with a trend line to produce an equation to estimate future inflation factors (Figure 40).

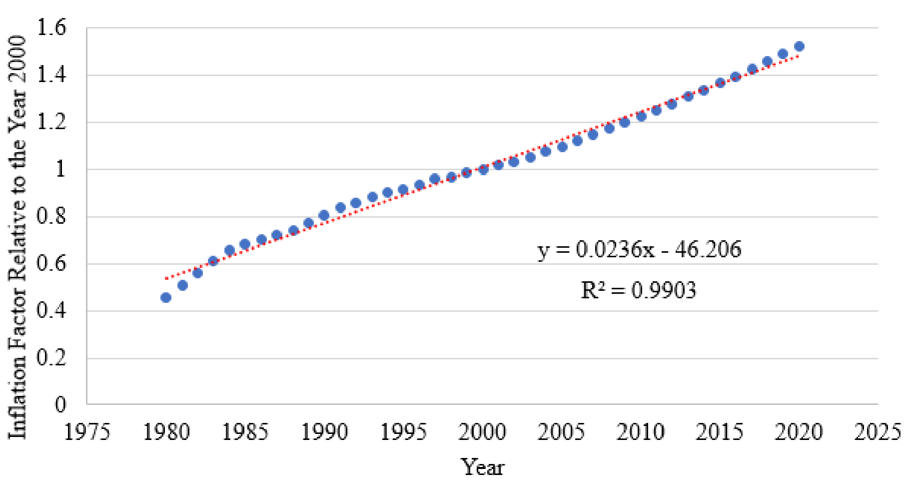


Figure 40: Projected Inflation Factors

12.3 Work Breakdown Structure

The WBS used for PROSERPINE was categorized into four structures including Research, Development, Test, and Evaluation (RDT&E), Theoretical First Unit (TFU), Communication Operations (CommOps), and Mission Operations (MOps) Team. Items listed from Ref. 82 were obtained by using CERs. CERs for the RDT&E and TFU WBSs are found in Tables 20-4 and 20-5, respectively, from Ref. 82. The CommOps and MOps WBSs were created independent of CERs and their respective references are indicated.

The items listed in black font are those included in the example WBSs from Ref. 82 while those listed in red are unique to PROSERPINE. All dollar amounts were inputted in the WBS using Fiscal Year (FY) 2000 costs and then converted to the proposed launch year of 2031 via an inflation factor.

12.3.1 Research, Development, Test, and Evaluation

Table XXIX shows the WBS for RDT&E with the specific propulsion that will be included on PROSERPINE. Costs estimated via CERs are from Ref. 82. Costs of the ion propulsion system components are estimated from the journal article Electric Propulsion System Selection Processes for Interplanetary Missions [107].

Table XXIX: RDT&E WBS Cost Estimate

Research, Development, Test, and Evaluation (RDT&E) WBS								
Cost Component	**Parameter**	**Input**	**CER (FY00$K)**	**Admin Fees (30% Overhead) FY00$K**	**Total (Subsystem + Overhead) FY00$K**	**FY31$K Inflation Factor**	**Then -Year Dollars (FY31$K)**	**Ref.**
1. Spacecraft	spacecraft dry wt. (kg)	2736	$276,336	$ 82,901	$ 359,237	1.7256	$ 619,899	1
1.1 Structure	structure wt. (kg)	628	$ 32,977	$ 9,893	$ 42,870	1.7256	$ 73,976	1
1.2 Thermal	thermal wt. (kg)	417	$ 18,167	$ 5,450	$ 23,617	1.7256	$ 40,754	1
	spacecraft wt (kg) + paylad wt. (kg)	4173	$113,165	$ 33,950	$ 147,115	1.7256	$ 253,861	1
1.3 Electrical Power System (EPS)	EPS wt. (kg)	1430	$ 89,661	$ 26,898	$ 116,559	1.7256	$ 201,135	1
	BOL power (W)	8500	$291,274	$ 87,382	$ 378,656	1.7256	$ 653,408	1
1.4 Telemetry, Tracking & Control System (TT&C)	TT&C wt. (kg)	168	$ 26,913	$ 8,074	$ 34,987	1.7256	$ 60,373	1
1.5 Attitude Determination & Control System (AD&CS)	AD&CS wt. (kg)	41	$ 5,568	$ 1,671	$ 7,239	1.7256	$ 12,491	1
1.6 Ion Propulsion Motor								1
1.6.1 PPU	cost	NA	$ 3,466	$ 1,040	$ 4,506	1.7256	$ 7,776	2
1.6.2 Thruster	cost	NA	$ 10,783	$ 3,235	$ 14,018	1.7256	$ 24,189	2
1.6.3 Tank	cost	NA	$ 1,078	$ 323	$ 1,402	1.7256	$ 2,419	2
1.6.4 Propellant	cost	NA	$ 1,078	$ 323	$ 1,402	1.7256	$ 2,419	2
1.6.5 Spare System	cost	NA	$ 16,374	$ 4,912	$ 21,286	1.7256	$ 36,731	2
2. Integration, Assembly, & Test (IA&T)	spacecraft bus + payload total RDT &E cost	$ 1,344	$ 1,278	$ 383	$ 1,661	1.7256	$ 2,867	1
3. Program Level	spacecraft bus + payload total RDT &E cost	$ 1,344	$ 839	$ 252	$ 1,091	1.7256	$ 1,883	1
4. Ground Support Equipment (GSE)	spacecraft bus + payload total RDT &E cost	$ 1,344	$ 944	$ 283	$ 1,228	1.7256	$ 2,118	1
						Total	**$ 1,996,299**	

12.3.2 Theoretical First Unit

The TFU is what Ref. 82 recommends for a single use satellite that is not to be mass produced, such as PROSERPINE. The CERs and propulsion cost estimates from the previous WBS are carried over to the TFU WBS with new estimations for payload instruments. The payload instruments were estimated using the NASA Instrument Cost Model (NICM) Version VIIc [108]. An example of utilizing the NICM is as follows for the Europa Imaging System.

The instrument name was inputted and the cost output fiscal year was selected as FY$00K. Optical was selected as the instrument type from a list including Optical, Active, Passive, Particles, and Fields. The environment was selected as Planetary from a list that also included Earth Orbiting. Next, the total mass was inputted. Since the mass breakdown of the instrument is unknown, the total mass was distributed evenly. Lastly, the detector type was selected as CCD. Then, the cost model estimate was generated and the 50% probability cost for the total sensor was selected as an average [108]. This process (shown in Figure 41) was repeated for the remaining instruments. The TFU WBS also includes the cost estimation for the launch vehicle, the Atlas V 552 [43]. Table XXX shows the TFU WBS.

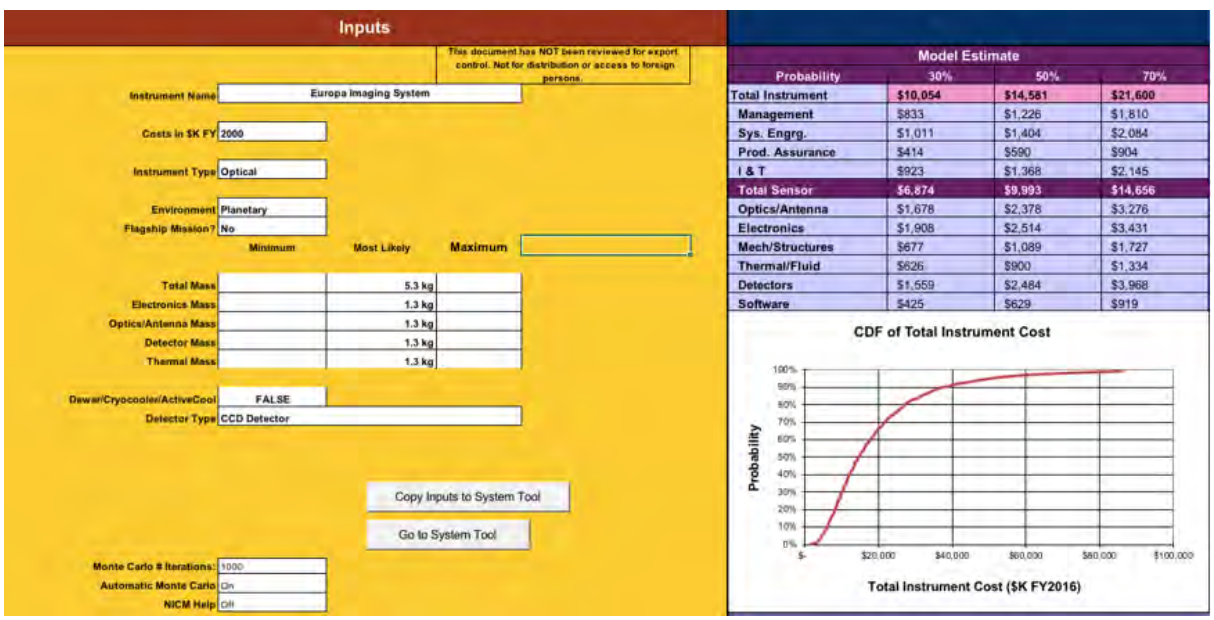


Figure 41: NICM Cost Modeling Estimation Example

Table XXX: TFU WBS Cost Estimate

Subsystem Theoretical First Unit (TFU) WBS								
Cost Component	Parameter	Input	CER (FY00$K)	Admin Fees (30% overhead)	Total (subsystem + Admin)	FY31$K Inflation Factor	Then -Year Dollars (FY31$K)	Ref.
1. Spacecraft	spacecraft dry wt. (kg)	2734	$ 117,562	$ 35,269	$ 152,831	1.7256	$ 263,724	1
1.1 Structure	structure wt. (kg)	628	$ 8,227	$ 2,468	$ 10,695	1.7256	$ 18,455	1
1.2 Thermal	thermal wt. (kg)	417	$ 3,602	$ 1,081	$ 4,683	1.7256	$ 8,081	1
	spacecraft wt (kg) + paylad wt.	4173	$ 113,165	$ 33,950	$ 147,115	1.7256	$ 253,861	1
1.3 Electrical Power System (EPS)	EPS wt. (kg)	1430	$ 28,625	$ 8,587	$ 37,212	1.7256	$ 64,213	1
	BOL power (W)	8500	$ 291,274	$ 87,382	$ 378,656	1.7256	$ 653,408	1
1.4 Telemetry, Tracking &	TT&C wt (kg)	168.0516	$ 11,663	$ 3,499	$ 15,162	1.7256	$ 26,164	
1.5 Attitude Determination & Control System (AD&CS)	AD&CS wt. (kg)	41	$ 5,297	$ 1,589	$ 6,887	1.7256	$ 11,883	1
1.6 Ion Propulsion Motor								
1.6.1 PPU	cost	NA	$ 3,466	$ 1,040	$ 4,506	1.7256	$ 7,776	2
1.6.2 Thruster	cost	NA	$ 10,783	$ 3,235	$ 14,018	1.7256	$ 24,189	2
1.6.3 Tank	cost	NA	$ 1,078	$ 323	$ 1,402	1.7256	$ 2,419	2
1.6.4 Propellant	cost	NA	$ 1,078	$ 323	$ 1,402	1.7256	$ 2,419	2
1.6.5 Spare System	cost	NA	$ 16,374	$ 4,912	$ 21,286	1.7256	$ 36,731	2
2. Integration, Assembly & Test (IA&T)	spacecraft bus wt. payload wt.	1404	$ 14,597	$ 4,379	$ 18,976	1.7256	$ 32,745	1
3. Program Level	spacecraft +payload total recurring cost (FY00$K)	$ 1,344	$ 458	$ 137	$ 596	1.7256	$ 1,028	1
4. Launch & Orbital Operations Support (LOOS)	spacecraft bus+payload wt. (kg)	1404	$ 6,877	$ 2,063	$ 8,940	1.7256	$ 15,428	1
4.1 Launch Vehicle	Atlas V 552 cost	NA	$ 111,032	$ 33,309	$ 144,341	1.7256	$ 249,075	4
5. Payload								
5.1 Europa Imaging System (EIS)	cost	NA	$ 9,993	$ 2,998	$ 12,991	1.7256	$ 22,417	3
5.2 Ice Penetrating Radar	cost	NA	$ 12,294	$ 3,688	$ 15,982	1.7256	$ 27,579	3
5.3 Submillimeter Wave Array Spectrometer (SWAS)	cost	NA	$ 13,852	$ 4,156	$ 18,008	1.7256	$ 31,074	3
5.4 Magnetometer (FGM)	cost	NA	$ 8,206	$ 2,462	$ 10,668	1.7256	$ 18,408	3
5.5 Magnetometer (V/SHM)	cost	NA	$ 8,437	$ 2,531	$ 10,968	1.7256	$ 18,927	3
5.6 Solar Wind Ion Analyzer	cost	NA	$ 4,426	$ 1,328	$ 5,754	1.7256	$ 9,929	3
5.7 Mass Spectrometer for Planetary Exploration (MASPEX)	cost	NA	$ 13,783	$ 4,135	$ 17,918	1.7256	$ 30,919	3
5.8 Ultraviolet Imaging Spectrometer/Spectrograph (UVS)	cost	NA	$ 10,281	$ 3,084	$ 13,365	1.7256	$ 23,063	3
5.9 Europa Thermal Emission Imaging System (E-THEMIS)	cost	NA	$ 13,084	$ 3,925	$ 17,009	1.7256	$ 29,351	3
						Total	**$ 1,883,266**	

12.3.3 Communication Operations

Table XXXI displays the cost breakdown of utilizing the DSN. For cost effectiveness, the orbiter will only transmit 0.5 hours, twice a week during the low power, 22-year transit. Once at Pluto, 8 hours of DSN time will be utilized a day for the 1-year mission. Costs for the utilization of the DSN 70 m antenna were estimated from Ref. 109.

12.3.4 Mission Operations

To estimate the team size needed and the respective costs for the Mission Operations Team (MOps), Fundamental Technologies, LLC was consulted. An interview with Fundamental Technologies owner and senior scientist, Dr. Jerry Manweiler, was conducted to explore the costs associated with processing data from a satellite post DSN. Based on his recommendations, a small team will monitor the orbiter as it cruises in low power, ready to transmit heartbeat mode. The team will monitor the craft during a typical 40-hour work week while continuously being on call in case of emergency. At Pluto arrival, the MOps team will include 5 team members at the Mission Operations Center (MOC). Additional team members will be at the Science Operations Center (SOC), including 2 engineers and 3 research scientists per instrument (45 total). Each instrument will also require unique interpretation software (9 total). A program manager was added for the duration of the mission as well [110]. Standard NASA salaries for research scientists and engineers were assumed [110]. One engineer per instrument (9 total) was assumed for pre-launch, satellite/vehicle integration. MOps is shown in Table XXXII.

Table XXXI: CommOps WBS Cost Estimate

Communication Operations (CommOps) WBS										
Cost Component	**Parameter**	**DSN Cost per Hour (FY00$K)**	**Hours per day**	**Length of Service (Days)**	**Subtotal Cost (FY00$K)**	**Admin Fees (30% overhead)**	**Total (subsystem + Admin)**	**FY31$K Inflation Factor**	**Then -Year Dollars (FY31$K)**	**Ref.**
1. Transit and Arrival Communication Operations										
1.1 Transit Deep Space Network (DSN) Time Rent	cost	$ 3.88	0.5	2288	$ 4,437	$ 5,321	$ 9,758	1.7256	$ 16,838	5
1.2 Arrival Deep Space Network (DSN) Time Rent	cost	$ 5.64	8	365	$ 16,475	$ 4,942	$ 21,417	1.7256	$ 36,957	5
								Total	**$ 53,796**	

Table XXXII: MOps WBS Cost Estimate

Mission Operations Team (MOps) WBS										
Cost Component	**Parameter**	**Cost FY00 $K**	**Team Size**	**# of Instruments**	**Leangth of Service (Years)**	**Admin Fees (30% overhead) FY00$K**	**Total (salary + Admin) FY00$K**	**FY31$ K Inflation Factor**	**Then - Year Dollars (FY31$K)**	**Ref.**
1 Program Manager	Senior Scientist Annual	$123	1	NA	23	Included	$ 2,840	1.7256	$ 4,900	6
2. Monitoring of Spacecraft/Launch Vehicle Integration	Annual Salary, Engineering, 1 Person per Instrument	$ 62	1	9	1	$ 166.67	$ 722.22	1.7256	$ 1,246	7
3. Team members during Transit	Graduate Student, 1/4 Full Time Equivalent	$ 41	2	NA	22	Included	$ 1,811	1.7256	$ 3,125	6
4. Team members at Arrival										6
4.1 Mission Operations Center (MOC)	Annual Salary, Mission Details Post DSN	$ 62	5	NA	1	$ 19	$ 327	1.7256	$ 565	7&6
4.2 Science Operations Center (SOC)	Annual Salary, Individual Payload per									
4.2.1 Engineering	Engineering Salary	$ 62	2	9	1	$ 333	$ 1,444	1.7256	$ 2,493	7&6
4.2.2 Science	Research Scientist	$ 55	3	9	1	$ 444	$ 1,926	1.7256	$ 3,323	7&6
5 Software	Instrument Specific	$ 69	NA	9	1	Included	$ 617	1.7256	$ 1,065	6
								Total	**$ 16,716**	

12.3.5 Summary

Table XXXIII summarizes the mission cost from the above WBSs. All of the WBSs were summed to provide the overall cost of PROSERPINE including research and development, integration to the launch vehicle, 22 years of transit, and one year of orbiting Pluto. The total cost of PROSERPINE is $3.3 Billion in 2018 Fiscal Year (FY) dollars. Figure 42 visually compares the four WBSs to display the change in cost within the mission timeline. The mission stage with the greatest cost is the RDT&E.

Table XXXIII: Mission Cost Estimate

RDT&E Component	Cost	FY	Cost	FY	CommOps. Component	Cost	FY	Cost	FY
1 Spacecraft	$1,989,431	FY31$K	$1,680,917	FY18$K	Transit & Arrival comm. Ops.	$ 53,796	FY31$K	$ 45,453	FY18$K
2 IA&T	$ 2,867	FY31$K	$ 2,422	FY18$K	**CommOps Total**	**$ 53,796**	**FY31$K**	**$ 45,453**	**FY18$K**
3 Program Level	$ 1,883	FY31$K	$ 1,591	FY18$K					
4 GSE	$ 2,118	FY31$K	$ 1,790	FY18$K					
RDT&E Total	**$1,996,299**	**FY31$K**	**$1,686,720**	**FY18$K**					

TFU Component	Cost	FY	Cost	FY	Mission Ops. Team Component	Cost	FY	Cost	FY
1 Spacecraft	$1,373,324	FY31$K	$1,160,354	FY18$K	Program Manager	$ 4,900	FY31$K	$ 4,140	FY18$K
2 IA &T	$ 32,745	FY31$K	$ 27,667	FY18$K	Spacecraft/Vehicle Integration	$ 1,246	FY31$K	$ 2,640	FY18$K
3 Program Level	$ 1,028	FY31$K	$ 869	FY18$K	Team Members During Transit	$ 3,125	FY31$K	$ 2,640	FY18$K
4 LOOS	$ 264,502	FY31$K	$ 223,484	FY18$K	Team Members at Arrival	$ 6,380	FY31$K	$ 5,391	FY18$K
5 Payload	$ 211,667	FY31$K	$ 178,842	FY18$K	Software	$ 1,065	FY31$K	$ 900	FY18$K
TFU Total	**$1,883,266**	**FY31$K**	**$1,591,216**	**FY18$K**	**MOps. Team Total**	**$ 16,716**	**FY31$K**	**$ 14,124**	**FY18$K**
					Mission Total	**$3,950,077**	**FY31$K**	**$3,337,513**	**FY18$K**

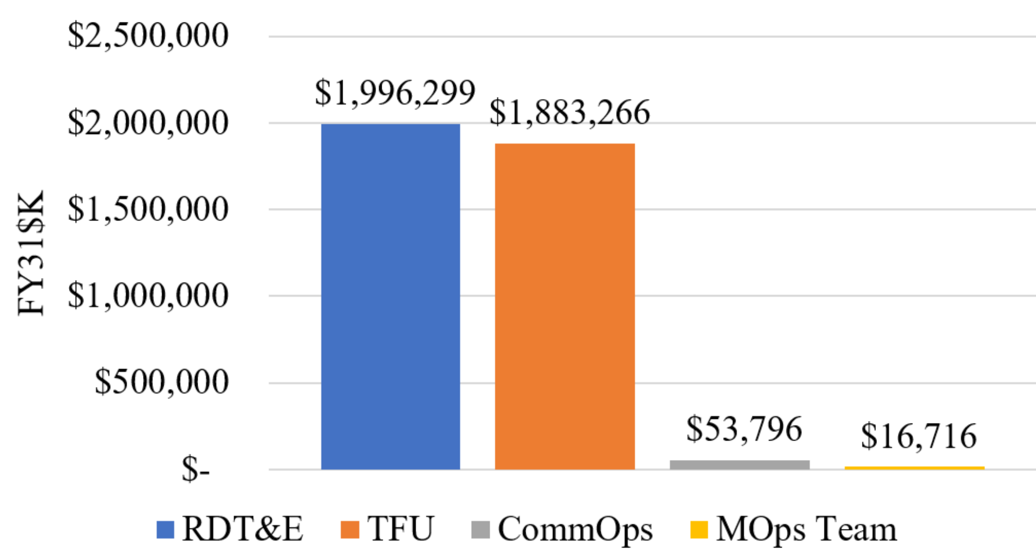


Figure 42: Mission Timeline Comparison

13. RISK ASSESSMENT AND MITIGATION

The Atlas V 552 satisfies the requirements for the 2031 mission. However, if that window is missed, both the Atlas V 552 and the SLS Block 2 do not satisfy the ΔV requirements of 13.4 km/s. Figure 43 shows the trade study conducted for the 2032 mission. If the Falcon Heavy does perform as it was calculated then it would provide enough ΔV, have a large enough fairing, and have a reasonable cost for the specific payload mass. However, by that time, the BFR should be ready for flight and is predicted to provide enough ΔV for the 2032 window.

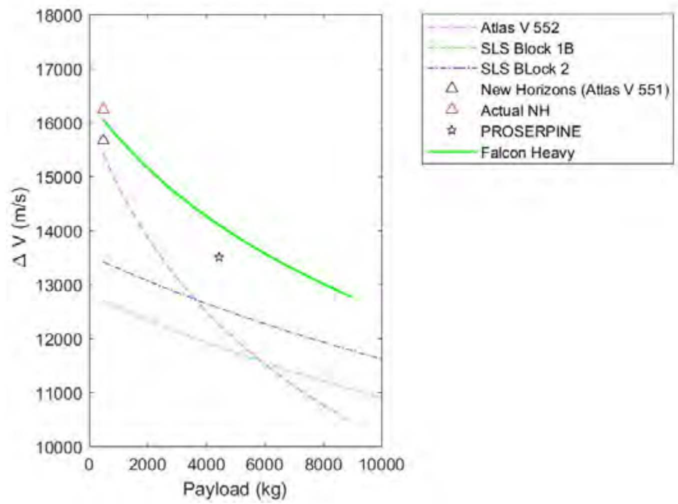


Figure 43: Trade Study for 2032

The inclusion of a fully redundant IEM is the foundation of the C&DHS risk mitigation strategy. In the event the primary IEM is silent for 180 seconds, the primary control functions will shift to the redundant IEM and the silent IEM will undergo a reboot process. Additionally, the processor voting structure within each IEM serves adds additional risk mitigation to the system.

The contingency mission for PROSERPINE is a flyby of the Pluto-Charon System. The minimum power was determined to be what PROSERPINE needed to still operate as a scientific mission in the Kuiper Belt, in the event the propulsion system fails to bring PROSERPINE into orbit in the Pluto-Charon system. At the minimum power, only the systems required to control the spacecraft and the instruments will be operating. These systems will include: telecommunications, power, attitude determination and control, command and data handling, and the scientific instruments. Table XXXIV shows the subsystem power allocation and the total power requirement with a 10% margin for a mission that would not be able to stop at the Pluto – Charon system.

Table XXXIV: Power Consumption for Pluto Flyby

Subsystem	Power
Propulsion	0 W
Power	36 W
Thermal	0 W
AD&C	43.5 W
C&DH	63.4 W
Telecommunications	6700 W
Payload	140.9 W
Mechanisms	0
Maximum:	8500 W

14. MARKETING AND PUBLIC RELATIONS

Given the significant cost of PROSERPINE, it is important to justify the mission to funding agencies and the general public. This strategy will focus on the scientific findings which we can expect from a new mission to the Pluto-Charon system, the technologies which PROSERPINE will demonstrate, and the opportunities for public science education which the mission will enable.

14.1 Scientific Objectives

The New Horizons mission revealed the Pluto-Charon system in unprecedented detail, greatly expanding human knowledge of the outer Solar System, yet simultaneously posing dozens of new questions which will go unanswered until another mission visits the distant worlds. PROSERPINE intends to answer as many of those questions as possible.

First, the spacecraft trajectory through the system is designed to allow broad mapping of surface, atmosphere, and magnetosphere of Pluto. New Horizons was only able to observe one side of Pluto and Charon in detail, which necessarily constrains scientists' ability to model the dynamical processes of the terrain. Furthermore, New Horizons did not carry a magnetometer, preventing any study of the magnetosphere. That spacecraft spent only a few days in close proximity to the system, precluding observations of the long-term processes and changes in the atmosphere and terrain. PROSERPINE is designed to remain in the Pluto-Charon system for at least eight years, enabling it to monitor these changes in detail.

Second, PROSPERINE carries instruments to investigate important surface and atmospheric phenomena, which will be used in conjunction with the general survey sensors to validate major models and test significant hypotheses about the structure of Pluto, Charon, and the broader system. Of particular interest is what Pluto can reveal about the early Solar System and planetary formation. Many suspect that Pluto will share features such as ice volcanoes with Triton, one of the moons of Neptune; confirming or disconfirming these hypotheses will improve scientific understanding of outer Solar System bodies.

Finally, it is unknown what new discoveries a new mission to the outer Solar System will produce. Only by performing such missions can humanity learn about its place in the universe.

14.2 Technology Development

PROSERPINE is explicitly intended to demonstrate new technologies which will enable other high-ΔV missions to the outer Solar System. Once an interplanetary spacecraft has successfully employed these technologies in the harshest deep space environments, mission planners will be free to incorporate such possibilities into their proposals.

The most important of these technologies is nuclear-electric propulsion. While the NASA Evolutionary Xenon Thruster will have been used in space well before PROSERPINE on the Double Asteroid Redirection Test, it is unlikely to have been powered by non-solar source. Successfully demonstrating nuclear power sources for spacecraft applications will expand the types of missions which space agencies can plan.

Similarly, NASA Kilopower Reactors are likely to have flown by 2031, but it is improbable that multiple units will have launched on spacecraft to the outer Solar System. Given the challenges of using solar power past Jupiter and the output limitations of the radioisotope thermoelectric generators, nuclear power offers the tantalizing possibility of large-scale missions to the outer planets. Without PROSERPINE, such missions will be political non-starters.

The other major advanced technology used on PROSERPINE is an extendable high-gain antenna. While such communications system cause trouble for the Galileo spacecraft, the configuration used on PROSERPINE follows models which have been successfully ground tested and are intended for near-term use in Earth orbit. This technology will be sufficiently mature for interplanetary flights by launch date and will allow a massive increase in data transmission rate while controlling the mass of the communications system. However, this improvement would be massively diminished without the high electrical power supplied by the nuclear reactor. These technologies operate synergistically and may be paired on future missions.

14.3 Public Outreach

PROSERPINE will launch almost a century after Clyde Tombaugh discovered Pluto in 1930. This astronomical coincidence provides an excellent opportunity to educate the public about the history of outer Solar System exploration, with an emphasis on changing views of Pluto. From Pioneer 10 through New Horizons, every new mission to the outer Solar System has updated the scientific consensus from a plethora of new evidence. PROSERPINE intends to continuous this tradition though extensive observations of Pluto and its moons.

To help the public connect with the mission, PROSPERINE will also follow several other spacecraft by carrying a microchip containing the names of space enthusiasts and other individuals interested in the flight. Recent probes carrying such chips include the Parker Solar Probe to study the photosphere of the Sun and the Mars InSight lander to study in the interior of the red planet.

Finally, new high-resolution images will be released as PROSERPINE reaches each body along its trajectory. These images and the other scientific findings from the mission will expand the cultural understanding of the Solar System, and impart a sense of wonder to future generations.

15. Future Possibilities

The development of an interplanetary spacecraft necessarily continues until mounted to the launch vehicle and carried to the launch pad. This evolution often occurs in ways which are difficult for mission planners to predict, but there are several areas of particular interest for PROSERPINE.

15.1 Upgrading the Science Suite

Scientific instruments may improve in two different ways before 2031. First, the mass, volume, or power requirement of a given instrument may be reduced. This would allow PROSERPINE to carry that same sensor, but reduces the impact of that sensor on the mission design. If enough instruments experience such improvements, it may become viable to include instruments which at the time of this writing had to be rejected.

Secondly, the scientific return of a particular instrument may be increased. Generally speaking, this is a tradeoff with the requirements for the instrument, though not in every case. If an instrument is improved in such a manner that it can be included with no additional cost to the spacecraft, than the improved version should be included.

Finally, new instruments may become available which are not useful in their current forms. In this event, the science team will review and decide whether it is worth substituting or adding further instruments. The development team of PROSERPINE should keep apprised of developments in the field of spacecraft sensors and adapt the science suite accordingly until the design fix date.

15.2 Impactor Probe

As currently configured, PROSERPINE is within the mass limit to employ an Atlas V 552 launch vehicle. Provided spacecraft wet mass does not grow dramatically before the launch date, this will enable further additions to the science payload. An exciting opportunity would be the inclusion of a surface impactor probe.

An impactor probe would enable two major types of scientific observations. First, it would return close-up images of the surface of Pluto or Charon prior to impact. Secondly, the impact itself would disturb the surface. The main spacecraft could then image the site on subsequent orbits, observing the resulting debris spray and determining what subsurface volatiles were released [42]. This would be the most direct way to study the subsurface composition without a dedicated lander.

The design and construction of the Compact, Low-Yield Dwarf Explorer (CLYDE) unit(s) will be outsourced to other space agencies or university teams as a mission-of-opportunity. Such teams will be responsible for the instrument selection, subsystem configuration, and fabrication of the CLYDE unit(s). The design will be constrained within tight mass and volume limits, most likely on the order of 100 kg. An impactor must be powered by primary batteries and transmit all of the data collected in real-time to PROSERPINE for relay to Earth. It must also contain the necessary systems to make observations, such as rudimentary attitude control, data handling, and propulsion. Preliminary calculations indicate that a cold gas thruster will be adequate for de-orbiting the impactor.

An RFP will be published closer to launch date if the spacecraft remains sufficiently under the Atlas V 552 mass limit. The CLYDE RFP will specify the mass, volume, power, data rate, and cost limitations necessary for inclusion on PROSERPINE. Winning design(s) will then be fabricated by the relevant space agency or university.

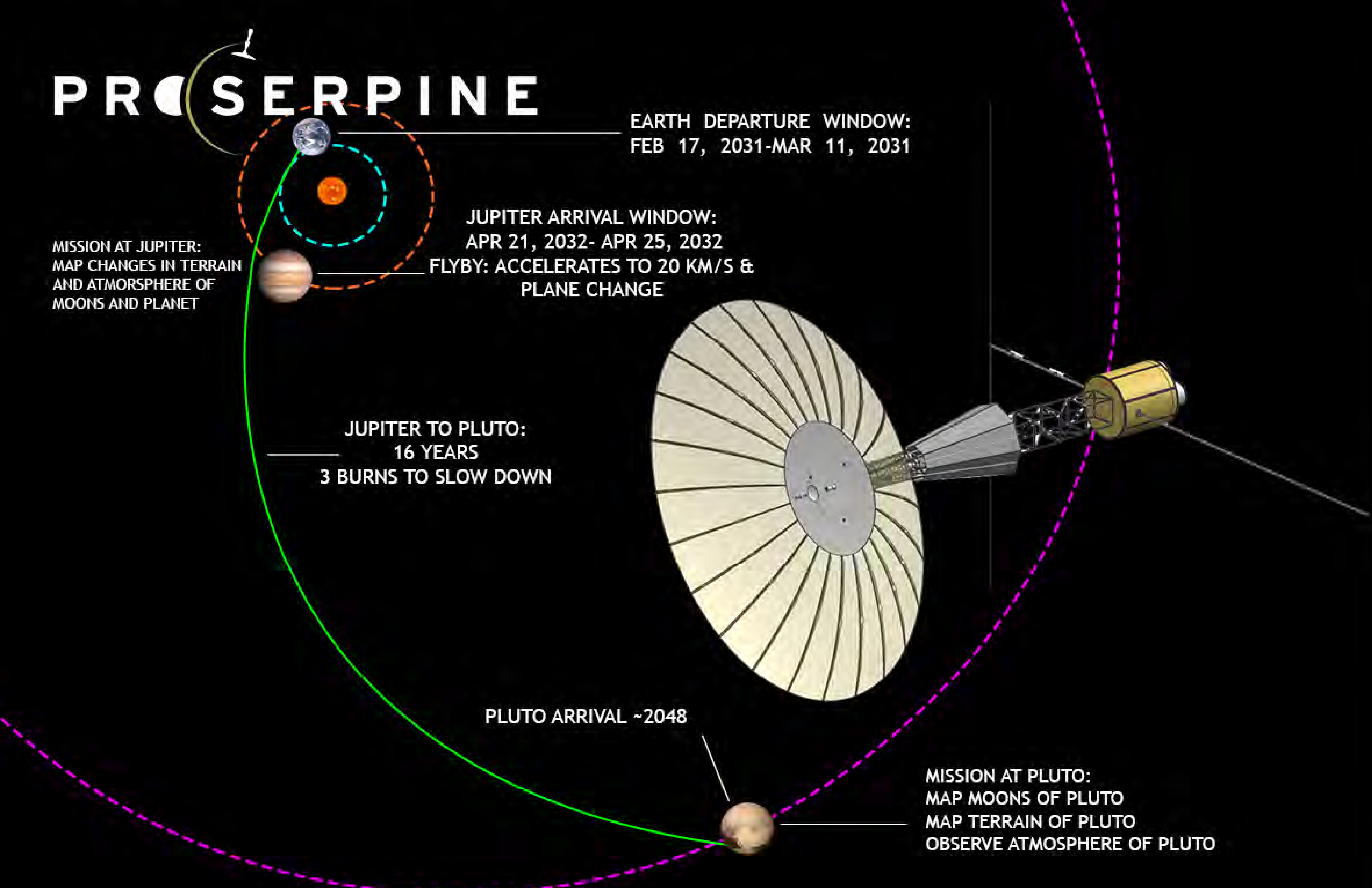
15.3 End-of-Life

After the initial mission operations complete, PROSERPINE will be eligible for extended missions to continue observing the Pluto-Charon system as it continues moving further from the Sun. The number and length of extensions will be subject to the availability of funding, the scientific objective to be satisfied, and the condition of the spacecraft. After several years in orbit, PROSERPINE will begin to run low on attitude control fuel. Once this reserve is exhausted, the spacecraft will be unable to point its instruments or align the high-gain antenna.

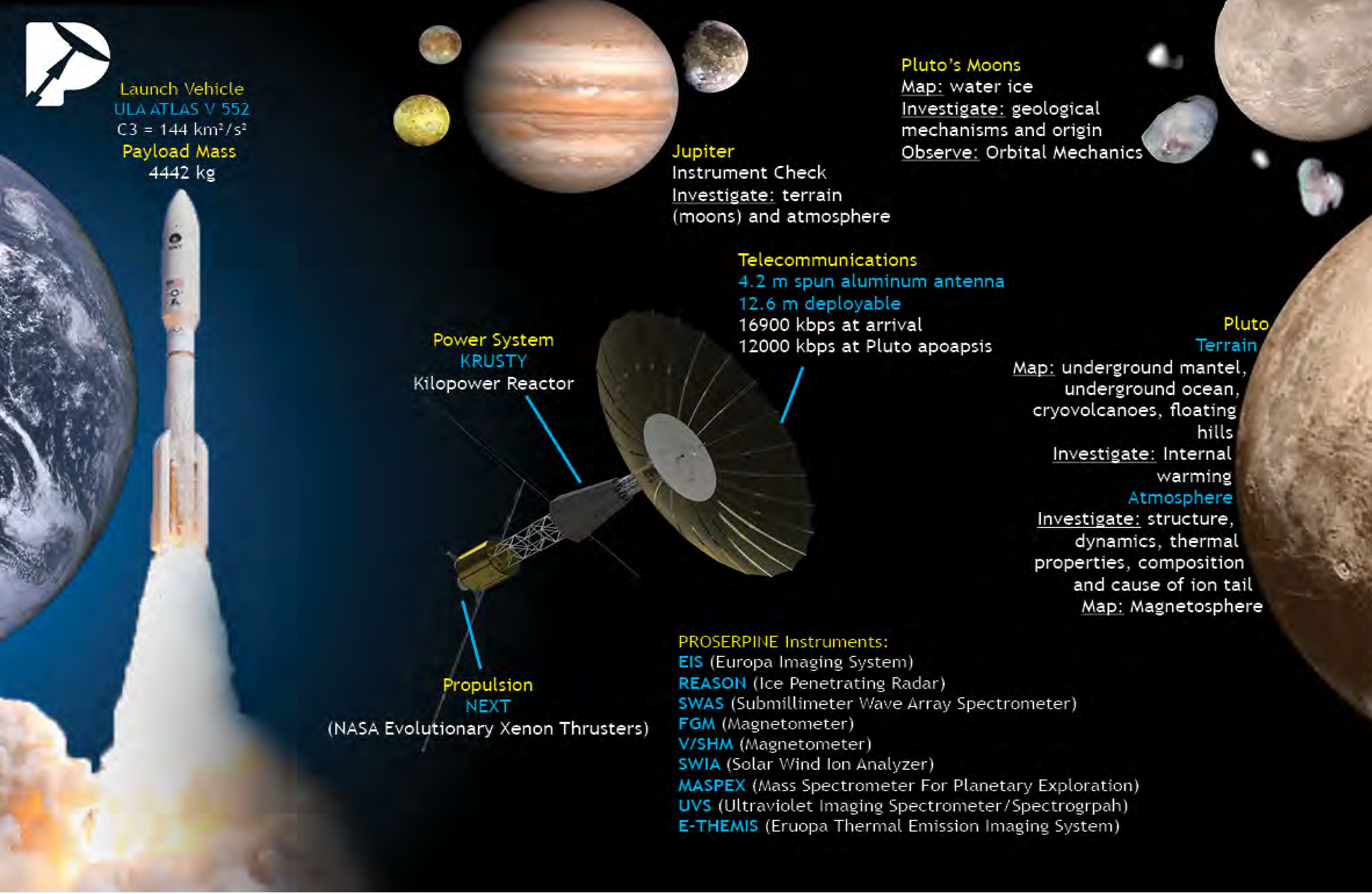
As this point approaches, ground controllers will consult planetary scientists about the optimal disposal technique based on the findings of the mission. Two possibilities are apparent at this time. First, the spacecraft can be placed into a disposal orbit. This would necessarily have to be a high orbit, well beyond those of the moons, to avoid the possibility of perturbations leading to a collision. Achieving this would require terminating close observation with plenty of fuel remaining to ensure that PROSERPINE will reach the disposal orbit.

The alternative would be to deliberately set up a collision with Pluto or Charon. This approach would allow mission controllers to conserve only a small quantity of xenon for final maneuvering, especially in the case of a Charon impact. It would also provide a final series of close surface observations, concluding the cornucopia of scientific findings from the distance worlds.

Though the Pluto-Charon system is distant and cold, planning an impact disposal implies certain planetary protection protocols would have to be followed. This may be unviable, or planetary scientists may find that impact is undesirable based on the mission findings. In either case, successful end-of-life will be completed intentionally, with all scientific data transmitted before turning the final page on the voyage of PROSERPINE to Pluto.



PROSERPINE
EARTH DEPARTURE WINDOW:
FEB 17, 2031-MAR 11, 2031
JUPITER ARRIVAL WINDOW:
APR 21, 2032- APR 25, 2032
FLYBY: ACCELERATES TO 20 KM/S &
PLANE CHANGE
MISSION AT JUPITER:
MAP CHANGES IN TERRAIN
AND ATMORSPHERE OF
MOONS AND PLANET
JUPITER TO PLUTO:
16 YEARS
3 BURNS TO SLOW DOWN
PLUTO ARRIVAL ~2048
MISSION AT PLUTO:
MAP MOONS OF PLUTO
MAP TERRAIN OF PLUTO
OBSERVE ATMOSPHERE OF PLUTO


Launch Vehicle
ULA ATLAS V 552
C3 = 144 km²/s²
Payload Mass
4442 kg
Jupiter
Instrument Check
Investigate: terrain (moons) and atmosphere
Pluto's Moons
Map: water ice
Investigate: geological mechanisms and origin
Observe: Orbital Mechanics
Telecommunications
4.2 m spun aluminum antenna
12.6 m deployable
16900 kbps at arrival
12000 kbps at Pluto apoapsis
Power System
KRUSTY
Kilopower Reactor
Pluto
Terrain
Map: underground mantel, underground ocean, cryovolcanoes, floating hills
Investigate: Internal warming
Atmosphere
Investigate: structure, dynamics, thermal properties, composition and cause of ion tail
Map: Magnetosphere
Propulsion
NEXT
(NASA Evolutionary Xenon Thrusters)
PROSERPINE Instruments:
EIS (Europa Imaging System)
REASON (Ice Penetrating Radar)
SWAS (Submillimeter Wave Array Spectrometer)
FGM (Magnetometer)
V/SHM (Magnetometer)
SWIA (Solar Wind Ion Analyzer)
MASPEX (Mass Spectrometer For Planetary Exploration)
UVS (Ultraviolet Imaging Spectrometer/Spectrogrpah)
E-THEMIS (Eruopa Thermal Emission Imaging System)

REFERENCES

1. "Request for Proposal: Pluto Orbiter", *American Institute of Aeronautics and Astronautics*, 2018.
2. "Payload", *The John Hopkins University Applied Physics Laboratory*, 2016.
3. "Ralph: A Visible/Infrared Imager for the New Horizons Pluto/Kuiper Belt Mission", *Space Science Reviews*.
4. Sutter, P., "The Mystery of Pluto's Beating Heart", *Purch*, 4 May 2017.
5. Talbert, T., "Pluto's Mysterious, Floating Hills", *National Aeronautical and Space Administration*, 6 Aug. 2017.
6. Chang, K., "What We've Learned About Pluto", *The New York Times Company*, 14 Jul. 2016.
7. Redd, N. T., "Ice Volcanoes May Erupt on Pluto", *Purch*, 9 Nov. 2015.
8. Talbert, T., "New Horizons", *National Aeronautical and Space Administration*, 3 Aug. 2017.
9. "New Horizons Reveals Pluto's Extended Atmosphere", *National Aeronautical and Space Administration*, 7 Aug. 2017.
10. "Pluto's Atmosphere", *The John Hopkins University Applied Physics Laboratory*, 2016.
11. Cofield, C., "Pandemonium! Motion of Pluto's Moons Perplexes Scientists", *Purch*, 10 Nov. 2015.
12. Algar, J., "Astronomers Struggle To Explain 'Pandemonium' Of Orbits of Pluto's Tiny Moons", *Tech Times*, 11 Nov. 2015.
13. Talbert, T., "Charon's Surprising, Youthful and Varied Terrain", *National Aeronautical and Space Administration*, 7 Aug. 2017.
14. "The Small Satellites of Pluto as Observed by New Horizons", *Science*, 18 Mar. 2016.
15. "Jupiter Encounter", *The John Hopkins University Applied Physics Laboratory*, 2016.
16. Scoles, A., "New Horizons Space Probe Gives First Look at Pluto", *Geek Insider*, 17 Jul. 2015.
17. Townsend, J., "Pluto-Bound New Horizons Sees Changes in Jupiter System", *National Aeronautical and Space Administration*, 9 Oct. 2007.
18. Salazar, D. E., "Jupiter's Great Red Spot Could Disappear Within 20 Years", *Purch*, 21 Feb. 2018.
19. "The Europa Imaging System (EIS), a Camera Suite to investigate Europa's Geology, Ice Shell, and Potential for Current Activity, *Europa Imaging System*, 16 Aug. 2017.
20. "Radar for Europa Assessment and Sounding: Ocean to Near-Surface (REASON)".
21. Gaier, J. R., "NASA Planetary Science Division's Instrument Development Programs, PICASSO and MatISSE, *National Aeronautical and Space Administration*.
22. Coleman, Jr., P. J., "Three-Axis Fluxgate Magnetometer", *National Aeronautical and Space Administration*, 21 Mar. 2017.
23. Fowler, C., "Cassini MAG Instrument".
24. "The Solar Wind Ion Analyzer for MAVEN", *Space Science Reviews*, 29 Nov. 2013.
25. "The Mass Spectrometer for Planetary Exploration (MASPEX)", *Institute of Electrical and Electronics Engineers*, 30 Jun. 2016.
26. "Seven SMD-Supported Instruments to Search for Evidence of Life on Europa", *Phys.org*, 20 Mar. 2017.
27. Waite, J. H., "MASPEX/Europa: Mass Spectrometer for Planetary Exploration", 24 Aug. 2015.
28. "ALICE: The Ultraviolet Imaging Spectrograph aboard the New Horizons Pluto-Kuiper Belt Mission), *Space Science Reviews*.
29. Christensen, P., "Europa Thermal Emission Imaging System (E-THERMIS)", 24 Aug. 2015.
30. Coskun, E. C., *Multistage Launch Vehicle Design with Thrust Profile and Trajectory Optimization*, Middle East Technical University, Sept. 2014.
31. Guo, Y. and Farquhar, R. W., *New Horizons Mission Design*, John Hopkins University.
32. Sutton, G.P. and Biblarz, O., *Rocket Propulsion Elements*, 9th ed., John Wiley & Sons, Inc., Hoboken, NJ, 2017.
33. Tewari, A., *Atmospheric and Space Flight Dynamics: Modeling and Simulation with MATLAB and Simulink*, Birkhauser, Boston, MA, 2007.

34. “Space Launch System (SLS): Mission Planner's Guide”, *National Aeronautical and Space Administration*, 12 Apr. 2017.
35. “Propulsion Products Catalog”, *Orbital ATK*, Oct. 2016.
36. “Atlas V Launch Services User’s Guide”, *United Launch Alliance*, Mar. 2010.
37. “Delta IV Launch Services User's Guide”, *United Launch Alliance*, Jun. 2013.
38. “Falcon 9 Launch Vehicle Payload User's Guide”, *SpaceX*, 21 Oct. 2015.
39. Kyle, E., “Vulcan Data Sheet”, *Space Launch Report*, 26 Apr. 2018.
40. Kyle, E., “SpaceX Falcon Heavy”, *Space Launch Report*, 6 Feb. 2018.
41. Kyle, E., “Blue Origin New Glenn”, *Space Launch Report*, 1 Apr. 2018.
42. Singer, K. and Finley, T., *Interview with New Horizons Team Members*, 12 Mar. 2018.
43. “Discover the Value of Launching on ULA’s Atlas V”, *United Launch Alliance*, 2018.
44. “FY 2018 Budget Estimates”, National Aeronautical and Space Administration, 2018.
45. Henry, C., “Blue Origin Enlarges New Glenn's Payload Fairing, Preparing to Debut Upgraded New Shepard”, *Space News*, 12 Sept. 2017.
46. Bruno, T., “Hey @Elonmusk, Congrats Again Your Heavy Launch. Clarification: Delta IV Heavy Goes for About $350M. That's Current and Future, after the Retirement of Both Delta IV Medium and Delta II. She Also Brings Unique Capabilities, At Least until We Bring Vulcan on Line”, *Twitter*, 12 Feb. 2018.
47. “Capabilities & Services”, *SpaceX*, 28 Nov. 2012.
48. Vallado, D. A., *Fundamentals of Astrodynamics and Applications*, 4th ed., Microcosm Press, Hawthorne, CA, 2013.
49. Brown, C. D., *Elements of Spacecraft Design*, American Institute of Aeronautics and Astronautics, Inc., Reston, VA, 2002.
50. “3D Solar System Simulator”, *The Sky Live*, 2018.
51. “Non-Impulsive Orbital Maneuvers, Constant Thrust”, *Small Satellites*, 2016.
52. Plachta, D. W., Christie, R. J., Jurns, J. M., and Kittel, P., “Passive ZBO Storage of Liquid Hydrogen and Liquid Oxygen Applied to Space Science Mission Concepts”, *Cryogenics*, Vol. 46, No. 2, 2006, pp. 89–97.
53. Clark, J. D., *Ignition! An Informal History of Liquid Rocket Propellants*, 1st ed., Rutgers University press, New Brunswick, NJ, 1972.
54. Dewar, J. A., *To the End of the Solar System: The Story of the Nuclear Rocket*, 1st ed., University of Kentucky Press, Lexington, KY, 2004.
55. Machado-Rodriquez, J. P., and Landis, G. A., “Analysis of a Radioisotope Thermal Rocket Engine”, *National Aeronautic and Space Administration*, Jan. 2017.
56. “Dawn Fires Its Engine above Ceres (Artist Concept)”, *Dawn Mission*, 17 Jul. 2015.
57. “Mission to Jupiter”, *National Aeronautical and Space Administration*, 20 Jan. 2010.
58. “The Juno Spacecraft Solar Panels”, *Space Math NASA*.
59. Dunbar, B., “Juno’s Solar Cells Ready to Light Up Jupiter Mission”, *Juno*, 15 Jul. 2011.
60. Shastry, R., Soulas, G., Aulisio, M., and Schmidt, G., “Current Status of NASA’s NEXT-C Propulsion System Development Project”, *National Aeronautical and Space Administration*, Sep. 2017.
61. Bennett, G. L., Lombardo, J. J., Hemler, R. J., Silverman, G., Amos, W. R., Johnson, E. W., Schock, A., Zocher, R. W., Keenan, T. K., Hagan, J. C., and Englehart, R. W., “Mission of Daring: The General-Purpose Heat Source Radioisotope Thermoelectric Generator”, 4th *International Energy Conversion Engineering Conference and Exhibit*, 2006.
62. Fortescue, P., and Stark, J., *Spacecraft Systems Engineering*, 2nd ed., Wiley, New York, 1995.
63. Foust, J., “Plutonium Supply for NASA Missions Faces Long-Term Challenges”, *Space News*, 20 Oct. 2017.
64. Bramanti, C., Izzo, D., Samaraee, T., Walker, R., and Fearn, D., “Very High Delta-V Missions to the Edge of the Solar System and Beyond Enabled by the Dual-Stage 4-Grid Ion Thruster Concept”, *Acta Astronautica*, Vol. 64, No. 7, 2009, pp. 735–744.

65. Oleson, S., Benson, S., Gefert, L., Patterson, M., and Schreiber, J., "Radioisotope Electric Propulsion for Fast Outer Planetary Orbiters", *National Aeronautical and Space Administration*, Sep. 2002.
66. Hagen, R., "Nuclear Powered Space Missions Past and Future (Technology)", *Space4Peace*, 8 Nov. 1998.
67. Voss, S., "SNAP Reactor Overview", *U.S. Air Force Weapons Laboratory*, Kirtland AFB, NM, Aug. 1984.
68. Woodcock, G., Kos, L., Johnson, L., Jones, J., Trausch, A., and Eberle, B., "Benefits of Nuclear Electric Propulsion for Outer Planet Exploration", American Institute of Aeronautics and Astronautics, Jan. 2002.
69. Gibson, M., Oleson, S. R., Poston, I., and McClure, P., "NASA's Kilopower Reactor Development and the Path to Higher Power Missions", *National Aeronautical and Space Administration*, Mar. 2017.
70. Gibson, M. A., Lee, M., Bowman, C., Poston, D. I., McClure, P. R., Creasy, J., and Robinson, C., "Development of NASA's Small Fission Power System for Science and Human Exploration", *National Aeronautical and Space Administration*, Jul. 2014.
71. Akin, D., "Akin's Laws of Spacecraft Design", *UMD Space Systems Laboratory*.
72. "Xenon Propellant Tank", *Cobham*.
73. Guzik, M. C., Plachta, D. W., and Feller, J. R., "Reduced Boil-Off System Sizing", Jun. 2015.
74. "In-Space Propulsion Technologies", *NASA Technology Roadmaps*, Jul. 2015.
75. Hall, L., "Powering Up NASA's Human Reach for the Red Planet", *National Aeronautic and Space Administration*, 13 Nov. 2017.
76. "Principles Relevant to the Use of Nuclear Power Sources in Outer Space", *United Nations*, 14 Dec. 1992.
77. Herman, D. A., and Unfried, K. G., "Xenon Acquisition Strategies for High-Power Electric Propulsion NASA Missions", Jun. 2015.
78. Van Noord, J., "Lifetime Assessment of the NEXT Ion Thruster", *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Joint Propulsion Conferences*, 2007.
79. Patterson, M. J., and Bensons, S. W., "NEXT Ion Propulsion System Development Status and Performance", *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Joint Propulsion Conferences*, 2007.
80. Hoskins, W. A., Aadland, R. S., Neckel, N. J., Talerico, L. A., and Monheiser, J. M., "NEXT Ion Propulsion System Production Readiness", *43rd AIAA/ASME/SAE/ASEE Joint Propulsion Conference & Exhibit, Joint Propulsion Conferences*, 2007.
81. "Spacecraft Systems and Components", *The John Hopkins University Applied Physics Laboratory*.
82. Larson, W. J and Wertz, J. R., *Space Mission Analysis and Design*, 3rd ed., Microcosm Press and Kluwer Academic Publishers, California, 1999.
83. "Nuclear Systems (NS) – Kilopower", *National Aeronautical and Space Administration*, 23 Jan. 2018
84. "Segall, G., "NASA Glenn Shares Successful Demonstration of Nuclear Power for Space", *Metro News*, 2 May 2018.
85. "Outer Space Treaty of 1967- Article IV", *National Aeronautical and Space Administration*, 6 Oct. 2006.
86. McKissock, B., Loyselle, P., and Vogel, E., "Guidelines on Lithium-Ion Battery Use in Space Applications", *National Aeronautical and Space Administration*, May 2009.
87. Smithrick, J. J. and O'Donnell, P. M., "Nickel Hydrogen Batteries - An Overview", *AIAA*, Jan. 1995.
88. Planetary Science Program Support Task, "Energy Storage Technologies for Future Planetary Science Missions", *NASA*, Dec. 2017.
89. Tomboulian, B.N., "Lightweight, High Temperature Radiator for In-Space Nuclear-Electric Power and Propulsion", Massachusetts, May 2014.
90. Fountain, G. H., Kusnierkiewicz, D. Y., Hersman, C. B., Herder, T. S., Coughlin, T. B., Gibson, W. C., Clancy, D. A., DeBoy, C. C., Hill, T. A., Kinnison, J. D., Mehoke, D. S., Ottman, G. K., Rogers, G. D., Stern, S. A., Stratton, J. M., Vernon, S. R., and Williams, S. P., "The New Horizons Spacecraft", *New Horizons*, 2009 ed., Springer.
91. "NASA Selects High Performance Spaceflight Computing (HPSC) Processor Contractor", *National Aeronautic and Space Administration*, 23 Mar. 2017.

92. “High Performance Spaceflight Computing (HPSC)”, *National Aeronautic and Space Administration*, 23 Jan. 2018.
93. The Consultative Committee for Space Data Systems, *Report Concerning Space Data System Standards: Lossless Data Compression*, CCSDS 120.0-G-3, Green Book, Apr. 2013.
94. **“**Deep Space Network Aperture Enhancement Project (DAEP)”, *National Aeronautic and Space Administration*, 7 Aug. 2017.
95. The Consultative Committee for Space Data Systems, *Report Concerning Space Data System Standards: Bandwidth-Efficient Modulations*, CCSDS 413.0-G-3-G-3, Green Book, Feb. 2018.
96. Blake, G., *Interview with RF Specialist*, 2018.
97. White, C. M., *Data Communications and Computer Networks*, Thomson Learning, Cambridge, MA, 2001.
98. Group, S. F. C., “Transponder Turnaround Frequency Ratios and Radio Frequency Channel Plans for Space Research, Category B”, Vol. SFCG 7-1R5, Space Frequency Coordination Group, 2007, pp. 1-5.
99. Massashi, U., Shirakawa, K., Hashimoto, T., Kubota, T., and Kawaguchi, J., “Attitude Control Challenges and Solutions for Hayabusa Spacecraft”, *Institute of Space and Astronautical Science (ISAS), JAXA.*
100. “Hayabusa-2 Asteroid Exploration Mission”, *Spaceflight101*, 2018.
101. Oh, D., Collins, S., Goebel, D., Hart, B., Latnoine, G., Snyder, S., and Whiffen, G., “Development of the Psyche Mission for NASA’s Discovery Program”, *35th International Electric Propulsion Conference Georgie Institute of Technology*, October, 2017.
102. “Dawn Spacecraft & Mission Overview”, *Spaceflight101*, 2018
103. “New Horizons: Spacecraft Overview”, *Spaceflight101*, 2018
104. “LN-200S: Inertial Measurement Unit (IMU)”, *Northrop Grumman*, 2013.
105. “Tile: Tiled Ionic Liquid Electrospray”, *Accion Systems*, 2018.
106. “2006-001A - New Horizons”, *Accion Systems*, 15 Oct. 2011.
107. Landau, D., Chase, J., Randolph, T., Timmerman, P., and Oh, D., "Electric Propulsion System Selection Process for Interplanetary Missions”, *Journal of Spacecraft and Rockets*, Vol. 48, No. 3, 2011, pp.467-476.
108. Habib-Agahi, H. "NASA Instrument Cost Model (NICM) Version VIIc", JPL, Pasadena, California, 2016.
109. “NASA’s Mission Operations and Communications Services”, *National Aeronautics and Space Administration*, January 2009.
110. Manwiler, J., *Interview with Fundamental Technologies, LLC*, 19 Apr. 2018.
111. “Average Salary for National Aeronautics and Space Administration (NASA) Employees”, *PayScale*, 2018.