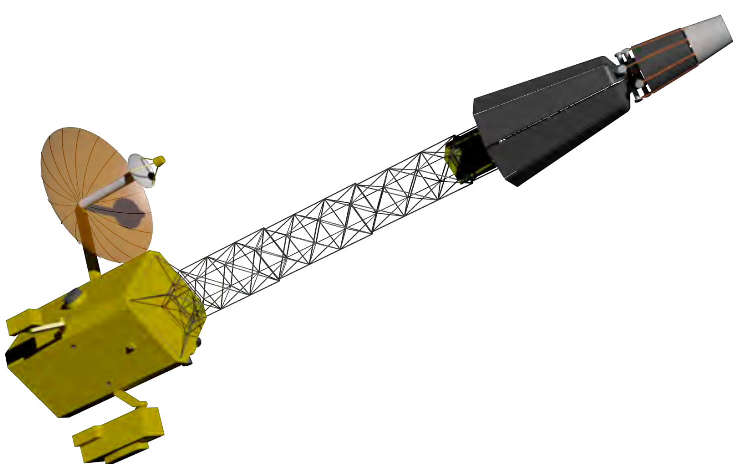
Orpheus

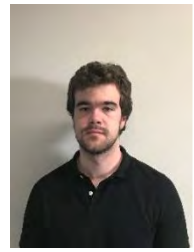
AIAA Pluto Orbiter Design Competition

AOE 4166: Senior Design
Instructor: Dr. Kevin Shinpaugh
Pluto Orbiter Group 1



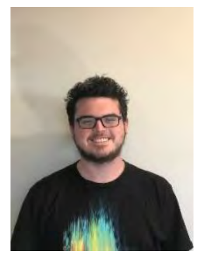
Submitted by: Rob Engebretson, Timothy Hughes, Daniel McGrath,
Jon-Paul Query, Clifton Reese
May 10, 2018

Daniel McGrath
Team Lead
Communications, Command and Data Handling/Attitude, Trajectory, and Orbits
AIAA Member Number: 921354

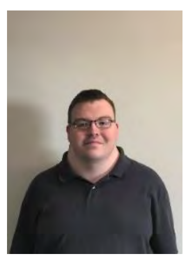


Rob Engebretson
Propulsion/Structures and Launch Vehicles
AIAA Member Number: 594706

Timothy Hughes
Attitude, Trajectory, and Orbits/Power, Thermal, and Environment
AIAA Member Number: 919746



Jon-Paul Query
Power, Thermal, and Environment/Structures and Launch Vehicles
AIAA Member Number: 919779



Clifton Reese
Attitude, Trajectory, and Orbits/Propulsion
AIAA Member Number: 921655

Kevin Shinpaugh
Faculty Advisor

Executive Summary

New Horizons, launched by NASA in 2006, performed a flyby of Pluto nine years later and gathered large amounts of data from the system. Orpheus, the successor to New Horizons, is designed to enter orbit around the Pluto system and send back larger amounts of data back to scientists on Earth. The name Orpheus was chosen for its mythological resonance with Pluto. Orpheus traveled to the underworld in order to bring back his wife, Eurydice, however at the end of the journey Orpheus left the underworld leaving behind his wife.

The primary goal of this project is to gather scientific data from the Pluto system while orbiting the system for at least one year. Orpheus will need to topographically and optically map Pluto's surface while inside the system. Instruments onboard the spacecraft will also thermally map and retrieve high resolution images of Pluto, Charon and its smaller moons. Since the spacecraft will spend at least one year on orbit, the spacecraft can study the dynamic evolution of Pluto's atmosphere before ejection from the system. Geological features that New Horizons missed during its flyby can be studied due to the prolonged presence around Pluto and close orbit.

The project requires the mission to conclude after 25 years from launch with a minimum of one year in the Pluto system. Due to the current position of Pluto and the outer planets, as well as booster availability, a Jupiter flyby trajectory has been chosen that requires a launch date of December 2028. This trajectory requires approximately 9.18 km/s of ΔV of onboard propulsion to achieve Pluto orbit, and will take roughly 17.5 years to arrive at Pluto. With this trajectory a total of 7.5 years can be spent inside the system exploring Pluto and its moons.

The high ΔV required to successfully capture the spacecraft implies that a high specific-impulse propulsion system is needed. After comparing several thrusters, the NEXT thruster system was selected because of its efficiency and long life expectancy. A set of two NEXT

thrusters will be used so that the risk of engine failure can be minimized.

To power the electric thrusters, Orpheus will employ an 8 kW reactor and a MMRTG to produce a total of 8.1 kW for the spacecraft. Only nuclear power systems were deemed viable because of the distance from the sun which makes solar power useless. Fuel cells were briefly discussed but became unnecessary once the reactor became viable.

Thermal control is accomplished by passive and active system which keep the spacecraft within its operational ranges. MLI will be used to thermally insulate the spacecraft and falls under the passive thermal system. Four sets of louvers are used to radiate excess heat near the inner solar system and a set of heaters will be used in the outer solar system.

Orpheus is designed to maximize performance for a reasonable cost, estimated to be between \$1,570 and \$2,027 million, while mitigating risk for the mission. Redundant systems have been used where necessary to ensure that Orpheus is able to complete its mission, including completely redundant computers and control hardware and a spare NEXT thruster. We believe that Orpheus will meet overall mission objectives and reveal valuable new information on both Pluto and dwarf planets in general.

Contents

List of Symbols and Abbreviations

$^\circ$	degrees
μm	Micrometers
AU	Astronomical Unit
dB	Decibel
Bps	Bits per second
BPSK	Binary Phase Shift Keying
C3	Characteristic Energy
CALORI	Close And Long range Optical Reconnaissance Imager
DEAP	Deep Space Network Aperture Enhancement Project
ΔV	Change in velocity
DNE	Do Not Exceed
DSN	Deep Space Network
Eb/No	ratio of received energy-per-bit to noise-density
GPHS-RTG	General-Purpose heat source RTG
Gs	Gravitaion force
HIRISE	High Resolution Imaging Science Experiment
IMU	Inertial Measurement Unit
Isp	Specific Impulse
keV	kiloelectronvolt
Kg	kilogram
Km	kilometer
$\frac{Km^2}{s^2}$	Kilometers squared per second squared
kW	KiloWatts
LA	Laser Altimeter
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
m	meter
m^2	square meters
mN	millinewton
$\frac{m}{s}$	meters per second
MLI	Multi-layer Insulation
MMRTG	Multi-Mission RTG
NASA	National Aeronautics and Space Administration
NEXT	NASA Evolutionary Xenon Thruster
NSTAR	NASA Solar Technology Application Readiness
PEPPSI	Pluto Energetic Particle Spectrometer Science Investigation
PPU	Power Processing Unit
RCS	Reaction Control System
REX	Radio Science Experiment
RS	Reed-Solomon decoding
RTG	radioisotope thermoelectric generator

SLS	Space Launch System
SPICE	Spacecraft Planet Instrument C-matrix Events
SWAP	Solar Wind at Pluto
SDC	Student Dust Counter
THEMIS	Thermal Emission Imaging System
TIS	Thermal Imaging System
US	United States
W	Watts
$\frac{W}{m^2}$	Watts per meter squared

List of Figures

List of Tables

1 Introduction

The goal of Orpheus is to follow New Horizons and gather more scientific data about Pluto and its moons by entering an orbit within the system. Questions about Pluto remain unanswered by the flyby performed by New Horizons and this project sets out to answer them. Using an instrument payload similar to that of New Horizons, Orpheus will gather topographical and thermal data about Pluto's dark side. Using a modified camera, Orpheus will also take higher resolution images of Pluto's smaller moons - Styx, Nix, Kerberos, and Hydra. While within a close orbit of Pluto, smaller features, such as potential cryovolcanoes, will be resolved.

1.1 Background

In February of 1930, Clyde Tombaugh discovered a ninth planet in the solar system by observing movement of an object in various photographs. [1] Later named Pluto, after the mythological god of the underworld, it was known as the ninth planet in our solar system for more than 70 years. In 2006, Pluto was reclassified as a dwarf planet. [2]

In the same year, NASA launched its New Horizons probe to study the dwarf planet Pluto in a flyby maneuver. Its goal was to study the surface of Pluto and its relatively large moon, Charon. After completing a gravity assist around Jupiter in 2007, New Horizons arrived in the Pluto system in July of 2015. The flyby was successful in obtaining far more detailed images of Pluto and Charon's surfaces and low-resolution images of Pluto's other, much smaller, moons. It is now exploring the Kuiper Belt and will eventually study other bodies in this system. [3]

Table 1: Needs, alterables, and constraints for Orpheus.

Category	Element
Needs	Launch a probe from Earth to Pluto orbit
	Send scientific data back to Earth
Alterables	Trajectory
	Propulsion Method(s)
	Launch Vehicle
	Launch Date
	Structure
	All Subsystems
	Orbit
Constraints	Maximum mission time of 25 years
	Orbit around Pluto system for at least one year
	One launch vehicle

1.2 Requirements

The primary mission requirements come from the AIAA RFP. The main requirement is gathering scientific data while in orbit within the Pluto system. The spacecraft must remain in orbit for at least one year, with a maximum mission duration of 25 years. This mission is limited to a single launch, and the selected propulsion system must be of TRL 6 or higher. The needs, alterables, and constraints for this project are listed in Table 1.

As described in Figure 1, the most important objective for the project is the overall design. This requires two second level objectives maximizing overall performance and minimizing total cost. These objectives are then divided into smaller factors which are measured by their effectiveness. On-orbit time describes how long the spacecraft is in orbit around Pluto. Power available deals with how much power can be allocated to the different systems, such as the cameras, antennas, or engines at any given time. This is measured in Watts and will change over the course of the mission. Transfer ΔV describes the speed needed to enter orbit around Pluto and is measured in km/s. This quantity for New Horizons was too high to be able to enter an orbit. Pointing error and position error deal with the orientation of the spacecraft and are measured in degrees and centimeters respectively. The total mass of the

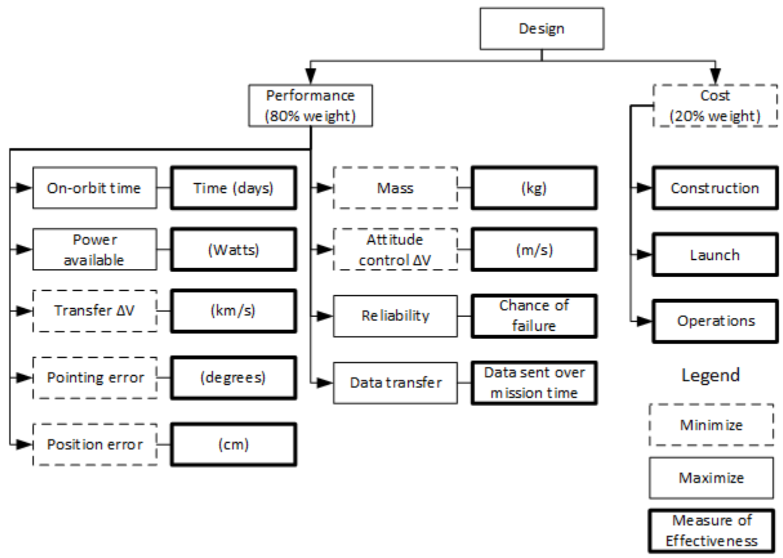


Figure 1: Orpheus objective hierarchy.

spacecraft is important when selecting a launch vehicle and amount of fuel needed to make it to Pluto. Reliability of the spacecraft is very important in trying to reduce the chance the spacecraft will fail. Data transfer is important in that it measures how efficient data can be sent back to Earth. Attitude-control ΔV describes the need for changes in speed based on the precession of the orbit around Pluto, and is measured in m/s.

Although not a direct constraint, or as important as performance, cost is a driving factor in the design. This is split among construction or assembly cost, launch cost, and operational cost. Cost of construction includes the individual subsystems and instruments. Launch cost includes the desired launch vehicle, which depends on the mass of the spacecraft. Operational cost only includes ground station cost, such as personnel, as it would be impossible to perform structural repairs after it is launched.

2 Scientific Objectives

To make this mission worthwhile, there have to be specific scientific goals to be completed. Based on the data received from New Horizons, there are many areas where there is still a significant lack of understanding. There are five main objectives that Orpheus is to complete.

The first objective will be to topographically and optically map Pluto's surface. While New Horizons brought back a resolution of a few hundred meters per pixel of the entire daylight side, with higher resolution in limited areas, an orbiter should provide the opportunity to return resolutions of 100 times greater over the entire surface. While New Horizons used stereo imaging to determine height, it only resulted in limited data, so a better technique needs to be used in order to provide a topographical map of the entire surface.

The second objective will be to thermally map Pluto and its moons. A thermal imaging system will allow the night side of Pluto to be mapped since an optical system won't work. Thermal maps will also allow spectroscopic determination of the materials on Pluto's surface that can't be identified with other instruments.

The third objective is to obtain high resolution images of the outer moons of Pluto and precisely determine their mass. The similarities of the Pluto system to the Earth and Mars systems could provide insight into their formation. Their masses and therefore densities could be determined through a close flyby of the small moons and measuring the perturbation.

The fourth objective would be to study the dynamic evolution of the atmosphere throughout the mission. Since an orbiter would watch Pluto closely for a long time, it provides a significant advantage over a flyby for this purpose. Scientists believe that Pluto's atmosphere may change or even freeze as Pluto recedes from the Sun, and this mission will allow a chance to test this theory.

The fifth objective would be to search for geological features that were missed in New Horizon's flyby. Specifically cryovolcanoes, or any other features that would allow insight into the geological history of Pluto. Cryovolcanoes are present on the geologically similar Triton, a moon of Neptune, leading to speculation that Pluto may have them too.

To achieve these objectives, several scientific instruments will be on board the spacecraft. This mission was designed to use the scientific instruments from New Horizons unless there are valid reasons to remove a specific instrument or replace it. Ralph was New Horizon's main camera, consisting of a visible light camera with four color imagers and three panchromatic imagers. Ralph was able to achieve a 1 km per pixel and search for cloud formations inside Pluto's atmosphere. The Long Range Reconnaissance Imager (LORRI) is a high-magnification visible light camera. LORRI was used to capture high resolution detail pictures of Pluto's geology and image Pluto's smaller moons during the flyby. The Student Dust Counter (SDC) is an instrument designed to measure microscopic dust grains in the Pluto system. It can measure the mass and speed of particles hitting the detector. The SDC was also the first dust counter to operate beyond 18 Au and was able to continue working after New Horizon's flyby. One of the instruments removed was the student dust counter for the reason that the one on New Horizons did not detect higher levels of dust particles than the rest of the solar system, so it was reasoned that including it would bring no significant scientific value. The LORRI and RALPH camera were replaced by a single camera that could exceed the capabilities of both, by employing a large sensor with higher resolution and a larger aperture.

Alice is an ultraviolet spectrometer intended to study the composition of Pluto and Charon's atmospheres. Alice can measure light between 500 and 1800 Angstroms and can be used in airglow or occultation mode. In airglow mode, Alice measures emissions from the atmosphere, while in occultation mode it measures the sun or a star shining through the atmosphere. Alice has a mass of 4.5 kilograms and uses an average power of 4.4 watts.

The Radio Science Experiment (REX) is simply a circuit board for signal-processing connected to the communications system. REX requires Pluto to pass between the spacecraft and Earth in order to run its experiment. Radio signals will be beamed from Earth, through the atmosphere and finally interact with the spacecraft's dish. REX will be able to determine the atmosphere's composition by how the radio waves interact in Pluto's atmosphere. Through successive passes the average molecular weight and temperature of the atmosphere can be determined. REX can also record radio emissions from Pluto and other bodies. Each REX board has a mass of 100 grams, and uses an average power of 2.1 watts.

Solar Wind at Pluto (SWAP) is an instrument used to measure interactions between Pluto's atmosphere and the solar wind. When gas escapes Pluto's atmosphere, it becomes energized by the solar wind and SWAP can measure these charged particles and compare them to solar wind measurements in interplanetary space. Data from SWAP will be able to determine the rate of gas escaping Pluto's atmosphere. SWAP can measure particles up to 6.5 keV and only has a mass of 3.3 kilograms while using an average power of 2.3 watts.

Similar to SWAP, the Pluto Energetic Particle Spectrometer Science Investigation (PEPSSI) is a directional energetic particle spectrometer. It will record charged particles caused by gas escaping Pluto's atmosphere and becoming charged by the solar wind. PEPSSI can measure up to 1000 keV which is much more energetic particles than SWAP. PEPSSI has a mass of 1.5 kilograms and uses an average power of 2.5 watts.

Three new instruments will supplement the four that have been inherited from New Horizons. They are designed to replace the instruments that were removed from New Horizon's instrument loadout or provide new functionality. Each new instrument will have to be developed from scratch, but there are similar instruments on previous missions to different solar system missions that they will be based on.

2.1 Close And Long Range Optical Imager (CALORI)

Instead of carrying two telescopes, Orpheus will carry one large telescope that will improve on the capabilities of both of the originals. The new instrument will be based on the

HIRISE telescope that was present on the Mars Reconnaissance Orbiter. [4] CALORI will be slightly smaller, with a 0.4 meter aperture instead of 0.5 meters. Like HIRISE, it will be developed with high resolution in mind, however due to the severely lower light conditions present at Pluto, a significant portion of the resolution available on HIRISE will have to be sacrificed. Instead of a resolution of 800 megapixels, it will likely have downgraded to as low as 40 megapixels, with the rest of the low light being compensated through longer exposures. This may likely limit the distance that Orpheus can approach before the images become blurred due to orbital motion, but this can be partially compensated through techniques such as tracking the surface features with the camera. This still allows a vastly superior resolution over LORRI, which only had 1 megapixel resolution. With this resolution, it would be possible to obtain resolutions as high as 3 meters per pixel, although stereo imaging techniques similar to the ones employed on HIRISE could reduce it down to 1 meter/pixel. Unlike LORRI, CALORI will allow imaging in visible light, with the sensor composed of red, green, and blue photoreceptors in equal numbers. In addition, CALORI will be the main tool for studying the other moons of Pluto as well as limited mapping of Charon from a distance. The key limitation of CALORI will be that it is unable to image the side of the dwarf planet surrounding the anti-solar pole, which could leave up to 40% of the surface unmapped. Thus its capabilities will have to be supplemented by other instruments.

2.2 Thermal Imaging Spectrometer

A thermal imaging spectrometer will be included to help alleviate some of the shortcomings of CALORI. The thermal imaging spectrometer is not limited to the ambient visible light conditions, so it can be used to map the morphology of the dark side of Pluto. This instrument will be based on the THEMIS instrument that was present on the Mars Odyssey spacecraft, but instead designed to match the thermal background temperature of Pluto. This will make the instrument most sensitive to infrared light around the wavelength of 66

μm. The resolution will be rather low compared to CALORI, only 0.038 megapixels due to the large individual photoreceptors that will be required to capture light at this high of a wavelength. In addition to mapping, the spectrometer will allow the analysis of the surface compositions of Pluto as well as search for cryovolcanoes, by seeking their greater heat signatures than the background of Pluto's surface.

2.3 Laser Altimeter

The final instrument that will be added to complete the study of Pluto will be a laser altimeter, which will allow full topographical mapping of Pluto's entire surface. The systems will consists of a pulsed 20 W laser that will fire a beam of light in a very narrow wavelength. Light from that beam will hit the surface of Pluto and scatter, but a very small portion will be directed to a dedicated 0.15 meter telescope. The difference in time between firing the beam and when the receiver detected the reflected photons will allow a precise distance to the surface to be determined. This technique, combined with a polar orbit, will allow all of Pluto to be mapped down to a vertical resolution of no more than a few meters. This system will be developed from knowledge on many different instruments with an identical purpose from past Mars and Mercury missions. [5]

While New Horizons did not put any of its instruments on a boom and instead mounted them directly on the spacecraft, this necessitates rotating the entire spacecraft in order to aim the instruments. To save fuel, the instruments that have to be pointed in a specific direction will be on booms that allow them to aim without the use of fuel. The booms would be able to retract close to the spacecraft and lock in place in order to survive the stresses of launch.

3 Concept of Operations

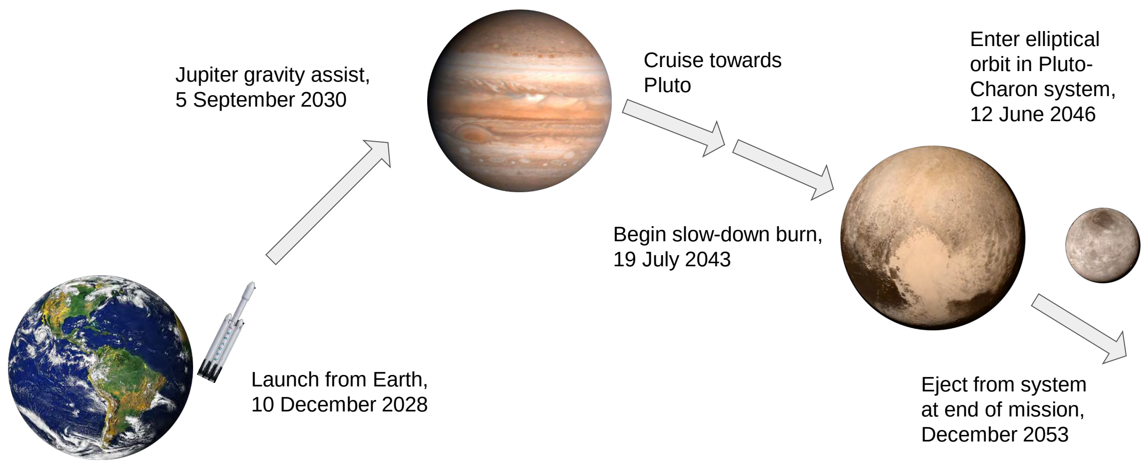


Figure 2: Concept of operations for Orpheus.

Orpheus's mission timeline is described in Figure 2. Using Space X's newly designed Falcon Heavy rocket, Orpheus begins it journey with a launch in December of 2028. After about a year and a half, Orpheus will perform a gravity assist around Jupiter in September of 2030, giving the spacecraft the needed plane change to reach Pluto. After cruising for more than ten years, Orpheus will begin its slow down burn in July of 2043. This will slow Orpheus down enough so that it can enter Pluto's weak gravity well. After burning for nearly three years, Orpheus will enter an elliptical orbit within the Pluto system in June of 2046. At the mission's conclusion in December of 2053, Orpheus will eject from the system to avoid crashing into or contaminating Pluto or one of its moons.

4 Trajectory and Orbits

Getting from Earth to Pluto in 24 years or less is a non-trivial problem, made worse by the current position of Pluto and the outer planets. A naive Hohmann trajectory would take 44.6 years, far more than the required mission timeline allows. Bi-elliptic transfers are worse. On top of that, Pluto is currently below the plane of the ecliptic, and getting further away each year. This requires that any trajectory handle a large plane change in addition to reaching the outer edge of the Solar system.

The General Mission Analysis Tool (GMAT) was used for all trajectory calculations to ensure that they accurately reflected the current and future state of the solar system. GMAT performs numerical integration of the equations of motion using SPICE (Spacecraft Planet Instrument C-matrix Events) planetary ephemeris data, published by the Navigation and Ancillary Information Facility, for the positions of astronomical bodies. [6] The default Solar system ephemeris data (DE421) does not contain full information on the Pluto system (or any moons other than Luna) so the PLU055 dataset, based on data from New Horizons, was added to ensure an accurate simulation. Additionally, the mission timeline extends beyond the DE421 data set's end date of January 1, 2050, requiring the use of the extended and updated DE432 data set, which contains ephemeris data for the same objects, extrapolated to January 25, 2650.

4.1 Trajectory

Initially, a direct general elliptic transfer to Pluto was considered. While such a trajectory could be adjusted to meet the mission timeline requirement, GMAT simulation showed that it would require a C3 launch energy of 200 $\frac{km^2}{s^2}$. This is beyond the capabilities of any current or near-future booster. The only remaining alternative is gravitational assists.

The excessive launch energy comes primarily from the plane change required to match

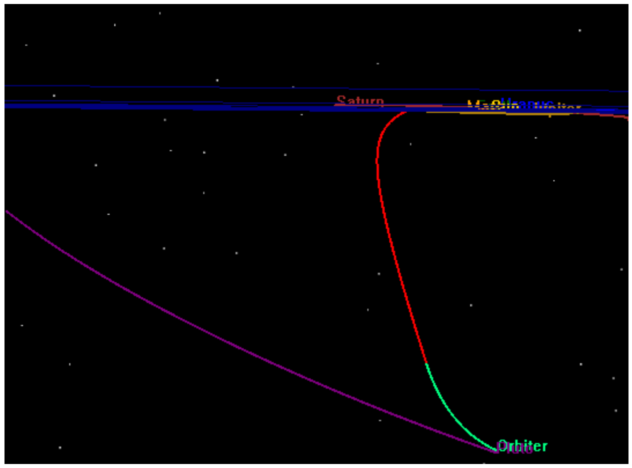


Figure 3: View of Orpheus trajectory from edge of ecliptic plane. Note plane change at Jupiter to match Pluto's 17.16° inclination. Rendered in GMAT.

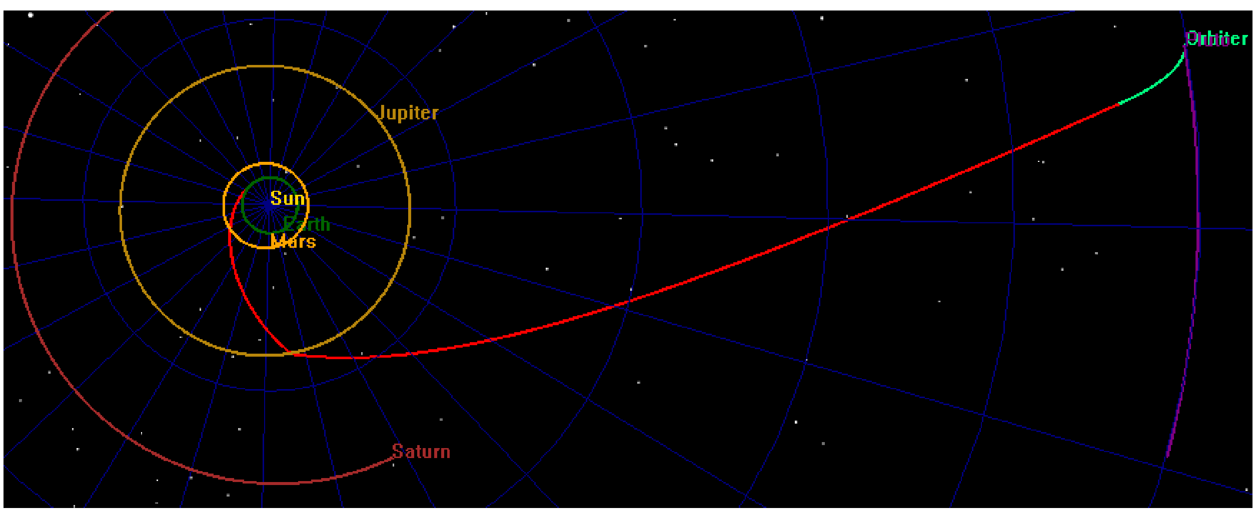


Figure 4: View of Orpheus trajectory from above ecliptic plane. Trajectory simulated for launch date of December 10, 2028. Jupiter flyby shown occurs September 5, 2030. Pluto insertion burn begins July 19 2043, and ends June 12, 2046, with the relevant section of the trajectory colored green. Rendered in GMAT.

Pluto's 17.16° inclination, as shown in Figure 3. A gravitational assist from any of the outer planets can be used to accomplish that plane change, but only Jupiter is going to be in position to be useful in the next thirty years. The inner planets could theoretically be used to reduce the required launch velocity, but at the expense of increased transit time. For this reason, Orpheus will be using a single Jupiter gravitational assist to both increase the spacecraft velocity and accomplish the plane change required to reach Pluto.

From the relative positions of Earth and Jupiter, the earliest launch window is in December of 2028, and is approximately one month wide. After many hours of refining the trajectory, using a launch date of December 10, 2028, a trajectory was found that reaches Pluto by June 12, 2046, with a low enough C3 energy of 98 $\frac{km^2}{s^2}$ to launch on current boosters, and a low enough ΔV of 9.12 $\frac{km}{s}$ to be practical to enter Pluto orbit. The full trajectory is shown in Figure 4. The initial trajectory was targeted assuming impulsive propulsion for the Pluto insertion burn, then refined to simulate electric propulsion when the engine parameters were known. With current spacecraft mass, the trip takes 17 years, 6 months, (6393 days) and allows for seven and a half years of science operations at Pluto. If the spacecraft mass were to increase this could be reduced to six years before reaching the current Do Not Exceed (DNE) mass.

4.2 Science Orbit

Orpheus's initial science orbit will be among Pluto's outer moons Styx, Nix, Kerberos, and Hydra, with a radius of 100,000 km relative to the Pluto system barycenter, and in-plane with Pluto and Charon's orbits. Orpheus will then spiral inward, making close passes with each of the outer moons for imaging and mass calculation. Finally, Orpheus will achieve an 800 km altitude close orbit around Pluto to map its surface, as seen in Figure 5. At the end of the mission, Orpheus will leave the Pluto system entirely to avoid contaminating any of its resident objects.

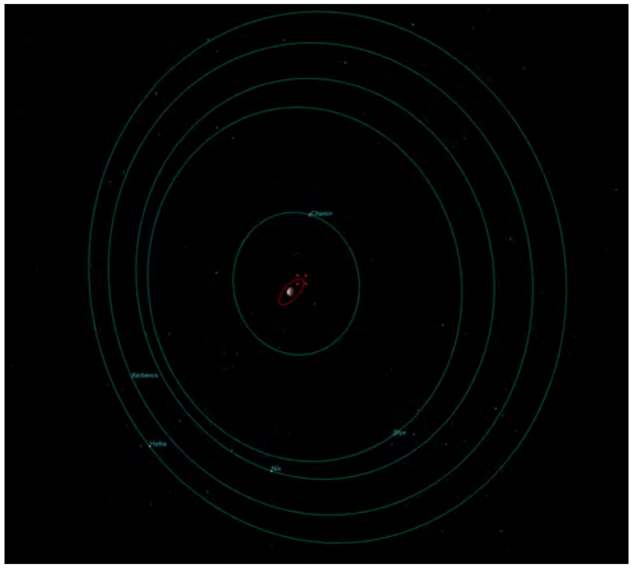


Figure 5: Orpheus's final mapping orbit, altitude of 800 km, shown in red. Rendered in Celestia, with orbit size exaggerated for clarity.

One complication in Orpheus's residence in the Pluto system is the fact that Pluto and Charon are similar enough in mass that the system barycenter is outside of Pluto. The perturbations from this make orbits between Styx and Charon very unstable without having perfect resonance with Charon. However, Charon's large mass does allow for gravitational assists to be used in-system for large orbit changes and for final ejection at the end of the mission timeline.

5 Propulsion

To achieve the required ΔV to get into orbit around Pluto, a propulsion system with a very high impulse was required. The propulsion system also needed to be small enough to be launched with enough C3 energy to get to Pluto from available launch systems. The mission does allow for long-term low-thrust burns.

5.1 Engine Types

There are two types of propulsion systems available to brake the craft: chemical rockets and electrical rockets. The chemical rockets allow a high thrust in exchange for a low specific impulse, and therefore require a lot more fuel. The highest specific impulse that can be found with a rocket using a fuel that is stable by itself over long periods of time is hydrazine, capable of a specific impulse of 330 s. [7] The highest specific impulse a chemical rocket can create is found in rockets burning liquid hydrogen and liquid oxygen, which is up to 470 s. [8] A LH2/LOX rocket would require heavy insulation to remain in liquid form for a couple decades, thereby reducing the structural mass ratio and overall effectiveness of the rocket despite the increases in specific impulse. The fuel to payload ratios required for 9,200 m/s were calculated to be 13 and 7, respectively. Without even considering the structural or engine mass, this easily puts the mass of the final stage at several tons.

Electrical propulsion was chosen as the only remaining option. The main advantage of electrical propulsion is having a much higher specific impulse, which reduces the payload mass ratio of the spacecraft. The main disadvantage of electric propulsion is low thrust, requiring long burn times. Another disadvantage of electrical propulsion is the power requirement. To achieve both high efficiency and I_{sp}, electric propulsion requires several kilowatts of power. Based on the analysis in Section 7, we will have eight kilowatts of power available.

Arcjets were the first form of electrical propulsion considered. However they have many disadvantages. Arcjets typically have less specific impulse than other types of electric propulsion. Additionally, they have lower overall efficiency, usually only around 30%, much less than the 60% typical of electrostatic thrusters. [9] Furthermore, arc jets are a lot less established than other forms of electrical propulsion. This leaves only Hall-effect thrusters and gridded electrostatic thrusters, both well-established forms of electric propulsion with efficiency of at least 55% at the power levels we would be operating at. Hall effect thrusters offer an advantage over the gridded electrostatic thrusters by having a greater thrust to power ratio, however this comes at the cost of specific impulse. Gridded electrostatic thrusters also offer a long lifespan than Hall Effect thrusters.

5.2 Engine Comparison

Both gridded electrostatic thrusters and Hall-effect thrusters were chosen for direct comparison due to the relatively few electric propulsion systems that were actually being developed for our high power levels. This included two Hall-effect thrusts, the XR-5 and the BHT-8000, and one electrostatic thruster, the NEXT.

The XR-5 was a thruster that was original developed by the company Busek, where it was known as the BHT-4000. [10] Development was taken over by Aerojet Rocketdyne, who renamed it the XR-5. Out of the engines chosen for comparison, the XR-5 is the only one that has flown to space, having been used on multiple Air Force satellites, where they performed with perfect reliability. However, due to their low operating power value of 4.5 kW maximum, at least two engines would have to be firing at the same time in order to make full use of the power available.

The next engine considered was the BHT-8000, from the same family of engines that the XR-5 was originally from [10]. Its nominal maximum power matches that of the reactor output, so only one would have to be firing at a time, and it would make perfect use of the power available. It has not been flown in space, but has been tested on the ground, achieving

Out[1434]=

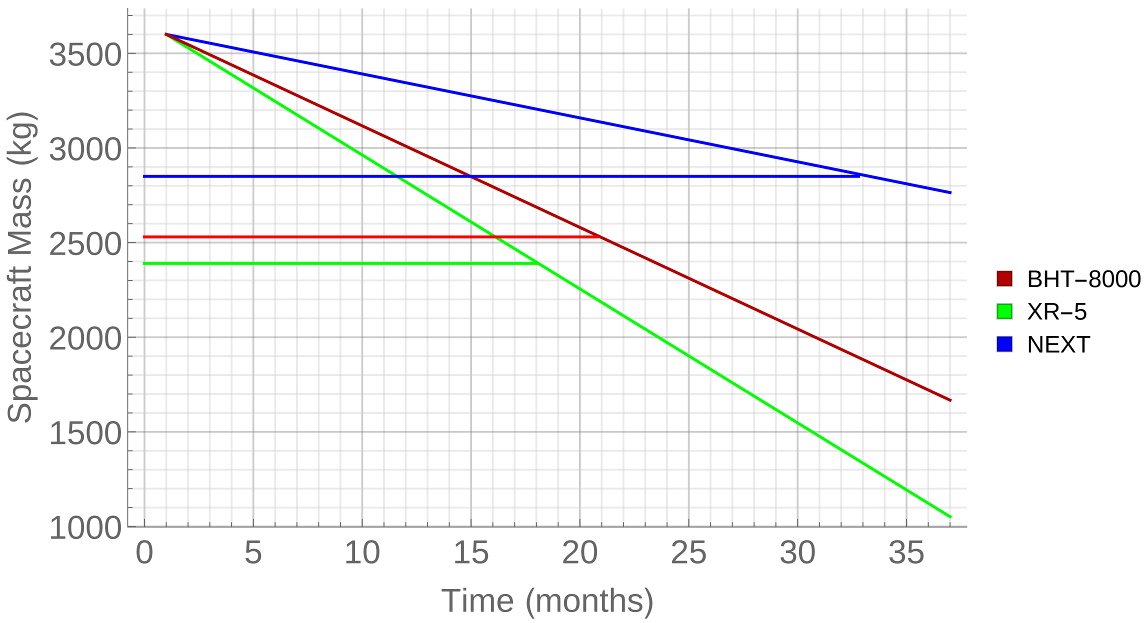


Figure 6: Comparison of the braking time required for each engine with its required initial mass.

Table 2: Comparison of engines.

Engine	Thrust (mN)	Isp (s)	Tested Lifespan (hrs)	Thruster Efficiency	Thruster Mass (kg)	Minimum Engines to Complete Mission	Reliability
XR-5 (x2)	492.4	1838	10,100	55.5	24.6	6	Tested in Space with good results
BHT-8k	449	2210	~10,000	60.8	25	3	Never tested in space
NEXT	315	3585	24,300+	71.7	13.5	1	Never Tested in Space, Based on NSTAR

TRL 6. While its reliability in space is unknown, it is being developed by the same people that made the reliable BHT-4000.

The final engine chosen for consideration was the NEXT engine, the gridded electrostatic thruster being developed by NASA. While no NEXT engines have been launched into space, they have performed well in ground test over a variety of power levels, earning it the minimum TRL of 6. [11] The NEXT is based on the NSTAR, an engine that performed perfectly on Deep Space 1 and Dawn. Being a gridded electrostatic thruster, its expected lifetime greatly exceeds that of its counterparts, allowing fewer to be aboard the spacecraft.

Table 3: Engine trade study.

Variable	XR-5	BHT-8000	NEXT	Weights
Thrust	4.5	4	2.5	2.5
Isp	2	2.5	4	3
Lifespan	2	2	3.5	4
Efficiency	2.5	3	4	2.5
Mass	2.5	2.5	3.5	1
Engines Required	1.5	2.5	5	3.5
Reliability	5	2.5	2.5	5
Score	2.57	2.27	3.03	

Out of the seven parameters shown in Table 2, thruster mass was weighted the lowest as the relatively small size of the typical ion engine prevents it from hampering the overall performance as much as poor thrust and specific impulse would. Tested lifespan was one of the most important parameters, because a longer lifespan would reduce the necessary number of backup engines and therefore reduce complexity and mass as well. One of the most important parameter was the reliability, because too many engine failures would end the mission. In order to understand how these engines would perform in the actual mission, a simulation for each engine was run to see how they compare.

The major difference between the engines in terms of their performance is the fuel and time required to stop. A higher fuel requirement reduces the dry mass of the spacecraft, leaving less room from the payload and power systems. Furthermore, less fuel used during the braking run would mean that more fuel is available for the in-system maneuvering. The engines' total fuel use was inversely proportional to the specific impulse for each engine, with the XR-5 requiring the most fuel and the NEXT using much less. Similarly, the XR-5 arrived at the Pluto system with its high thrust to power ratio, and NEXT took nearly twice as long. Engine lifespan is crucial for low-power missions, but with the shorter burn time enabled by the available power for this mission, engine lifespan is less important.

Reliability was also a significant concern. The highest performer in reliability was the XR-5, with one of the flown engines even saving a mission when its main hydrazine thrusters failed. [12] The other two engines have not flown and are at approximately the same level of development. The lifespan of the engines was also important, as engine with shorter lifespans can be expected to fail before the mission has been completed, requiring additional redundant engines. Both the Hall effect thrusters had relatively short lifespans around 10,000 hours, requiring at least two sets of them to complete the mission. [13] [12]. The NEXT, which has the longest lifespan of all the engines, lasts up to 48,000 hours when used at the nominal power levels. However, its nominal power level is 1.1 kW less than what the spacecraft can provide. One solution would be to reduce the power level to 4 kW per engine and run two engines at the same time, but this would drop the efficiency of the system by 60%. A better solution is increasing the power to beyond its nominal power level, which was extensively tested by NASA. This will reduce the expected lifespan of the system by 50%. The NEXT engine already provides sufficient longevity, so it would still last long enough to complete the mission.

5.3 Final Propulsion Configuration

The NEXT was chosen as the optimal engine to complete this mission. The engine allows a lot of fuel to be saved for in system maneuvers, as well as reduces the overall mass of the spacecraft. The engine as mentioned before, would be run at a higher than nominal power level. Since the NEXT has been tested at a power level of 7.73 kW, that was power level chosen. The specific power level also operated at 5.8 amps and 1179 volts. The NEXT's long life allows for a reduced number of engines, with one engine able to complete the mission by itself, and another on board for redundancy. Since neither engine will be able to be aligned with the center of mass while pointing in the same direction at the same time, both engine will be mounted on a gimbal. Neither engine will be run exclusively until it breaks. Instead the engines will be cycled, similarly to the three NSTAR engines on the dawn mission. They will be spaced 0.5 meters to prevent overheating each other. The two engines will be supplied with power by a Power Processing Unit (PPU) based on the 7 kW system that is being developed for NEXT, but scaled up slightly to provide an extra kilowatt of power. One will be shared between both engines, but a redundant PPU will also be present should the first fail. All other components such as the Xenon Control Assembly will also come in pairs for redundancy.

5.4 Summary

A high impulse propulsion system was necessary in order to achieve the 10 km/s delta v for Pluto orbit insertion. Chemical propulsion was ruled out because it would require excessively high propellant to payload ratios, leaving only electric propulsion. Three ion propulsion engines, one gridded electrostatic thruster, and two Hall effect thrusters were chosen and compared through their capabilities and their performance in a simulated deceleration burn. While all had acceptable performance, the NEXT when run at a higher than nominal power setting was ideal.

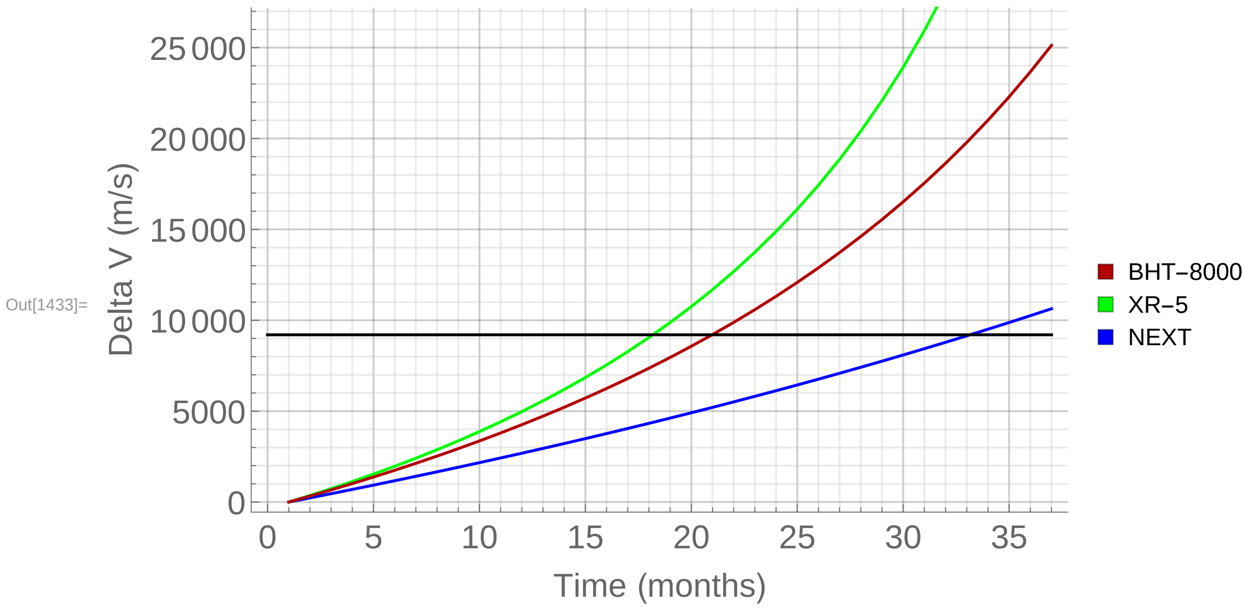


Figure 7: Comparison of time required to achieve sufficient ΔV to complete the 9.21 $\frac{km}{s}$ Pluto insertion burn. Spacecraft mass set at 3600 kg.

6 Launch Vehicle

To get to Pluto, a super heavy lift launch vehicle will be required. While there are many rockets available in the market, not many of them are of the super heavy class of launch vehicles. Luckily, many rockets are in development that would be ready by the time Orpheus would launch. The main criteria for selecting the most suitable rocket are its cost and mission success rate. The main constraint would be if the rocket had enough delta v to get Orpheus to Pluto in time.

Regardless of the launch vehicle, any spacecraft going to the outer solar system would benefit from a third stage. Typically, a solid rocket engine is used for simplicity and low cost. The Star rocket stages are a series of solid rocket stages built solely as 3rd stage boosters. One of the largest of the Stars is the 75 stage, which is capable of 2,826 m/s of delta V with the Orpheus as payload. [14] The Star 48-B is spin stabilized during its firing and is spun up by a motor that would be mounted to the second stage. Once done firing, the Star 48-B and Orpheus will be de-spun using a yo-yo de-spin technique. The Star 75 comes from the same class of booster that was used as the third stage of New Horizons.

The Atlas V 551 was the rocket used to launch New Horizons and is used as a baseline for the payload lifting capacity required. However, it was clear earlier on that the same rocket

Table 4: Launch vehicle comparison

Variable	Atlas V 551	Falcon Heavy	SLS Block 1
LEO Payload (kg)	20,520	63,800	70,000
GTO Payload (kg)	8,900	26,700	~30,000
C3 Energy for 12000 kg payload	-50	24	45
Cost (Millions $)	110	150	>400
Reliability (Success,Partial Failure,Full Failure)	76,1,0	1,0,0	0,0,0

would not be able to launch a spacecraft with 7 times the mass of New Horizons to the same location. Two other rockets either in development or recently completed were considered for meeting the minimum requirements for this mission. The first one is the Falcon Heavy, which just recently completed its first launch. While its exact C3 energy is unknown for a payload of 12,000 kg, it is known that it is capable of launching 16,800 kg to a Mars transfer orbit. [15] That corresponds to a C3 energy of approximately 11.5 $\frac{km^2}{s^2}$. This mission requires 12,000 kg to a C3 energy of 22 $\frac{km^2}{s^2}$. The difference between these two characteristic energies corresponds to a ΔV of 441 $\frac{m}{s}$. With a reduction in the payload mass by 4,800 kg the Falcon Heavy should provide the required C3 energy.

The only other possible rocket that could be used is the SLS block 1, which is currently in development by NASA. It offers slightly more payload capacity than the Falcon Heavy. SLS has been in development for a while and experienced numerous delays, and the threat of cancellation is always higher than for an already completed rocket. The price tag will also be much greater for the SLS than for the Falcon Heavy.

6.1 Summary

Orpheus will use a Falcon Heavy rocket as its launch vehicle as well as a Star 75 solid rocket motor as its third stage. The Falcon Heavy will provide Orpheus and the Star 75 with 24 $\frac{km^2}{s^2}$ of C3 energy. Burn out of the second stage will occur with a velocity of 12,206 m/s. The Star 75 will fire as quickly as possible after separation and provide a final acceleration to 14,933 $\frac{m}{s}$, which has the 98 $\frac{km^2}{s^2}$ of characteristic energy required to reach Pluto.

7 Power

The power system for Orpheus is a vital system, because it is required for every other system to function. It has direct influence on thermal management, communications, and propulsive power. Additionally, it must be able to operate in the widely varying conditions between Earth and Pluto. Finally, it must be long-lived enough to function for the entire 25-year mission duration.

The very fact that the power system must function at Pluto eliminates all forms of solar power systems as impractical. At Pluto's distance of 39.5 AU from the sun, a solar panel would be 1,500 times less efficient than at Earth, requiring 1,500 times as much solar panel area to produce the same amount of power. This applies to any power system that relies on solar radiation, rendering them useless for Orpheus. Likewise, fuel cells and batteries are not usable due to their limited lifespan.

The only remaining form of power supply that has adequate lifespan and power density is nuclear. Radioisotope Thermoelectric Generators (RTGs) are a traditional power source for spacecraft that make use of the heat from the decay of relatively short-lived radioactive isotopes (usually Plutonium-238) to generate electricity. They are highly reliable and have been used on every spacecraft to explore the outer solar system, including New Horizons. However, the only RTG currently in production for NASA is the Multi-Mission Radioisotope Thermoelectric Generator (MMRTG), shown in Figure 8. Compared to the out-of-production General-Purpose Heat Source RTG (GPHS-RTG) used on New Horizons, it has a relatively poor power-to-weight ratio, and would require four RTGs to accomplish the Pluto insertion burn. On top of that, the deceleration would take so long that Orpheus would have almost one year on orbit for scientific investigation, and barely sufficient power to send that data back. With the current rate of Plutonium production, the use of MMRTGs as the sole power source for New Horizons is impossible.

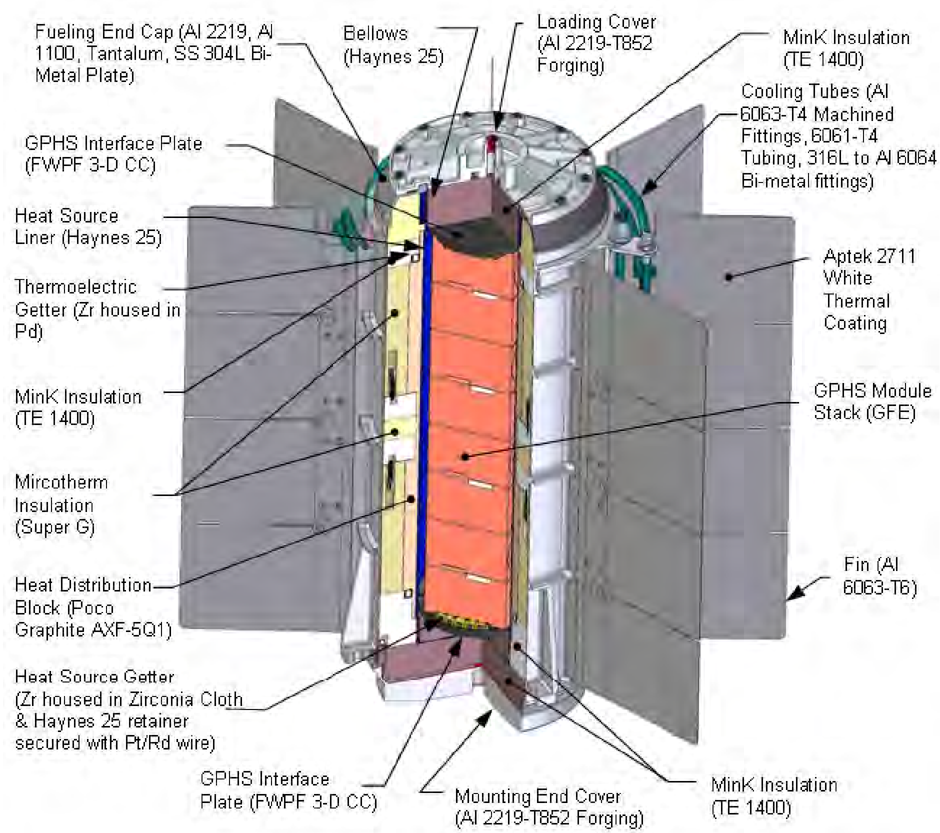


Figure 8: Multi-Mission Radioisotope Thermoelectric Generator. Weight 45 kg, initial power output 125 W. Image credit: Ryan Bechtel, U.S. Department of Energy.

Table 5: Power systems trade study

Variable	4 RTGs	5 RTGs	4 RTGs + Liquid Stage	8 kW reactor	Weights
Burn time (days)	5511	4409	3438	583	N/A
Time on orbit (years)	0.9	1.5	1.6	6	0.2187455916
Cost (billions)	1.682	1.808	1.752	1.927	0.09134261364
Data rate (bits/s)	860	1075	860	545000	0.2194859588
Political feasibility (higher better)	7	6	7	3	0.4071699571
Risk (lower better)	4	4	5	7	0.0632558789
SCORE:	3.883411591	3.58300435	3.95196435	5.68280391	

On encountering this problem, several alternatives were considered, including increasing the number of RTGs to five, adding a liquid hydrazine stage to reduce the time required for deceleration, and replacing most of the RTGs with an 8 kW fission reactor based on the recently tested Kilopower reactor. The trade study used to analyze this problem is shown in Table 5.

The Kilopower reactor, shown in Figure 9 makes use of highly-enriched metallic Uranium fuel, bypassing the Plutonium availability problem. Its high power output enables both a much faster deceleration, and corresponding increased science time, as well as greatly increased communications bit rate, allowing much more data to be sent back. While it does not have the proven flight record of RTGs, its use of heat pipes for thermal transport and high-lifespan Stirling engines for power conversion ensures that it will function correctly for the entirety of its 10-year lifespan. [16]

The use of the fission reactor substantially reduces the amount of radioactive material

Table 6: Power budget for all three flight modes.

Subsystem	Component	Power, W (Cruise)	Power, W (Propulsive)	Power, W (Scientific)
Scientific Instruments	Alice (UV spec)	0	0	4.4
	REX (radio)	0	0	2.1
	CALORI (optical camera)	0	0	10
	SWAP (solar wind)	0	0	2.3
	PEPSSI (neutral atoms)	0	0	2.5
	Thermal Spectrometer	0	0	5
	Laser Altimeter	0	0	20
	Subtotal	0	0	46.3
Propulsion	NEXT	0	7730	0
	Subtotal	0	7730	0
Thermal	Heaters x14	30	30	30
	Subtotal	30	30	30
CC&DH	Comms/Electronics	10	10	7900
	Computer	21.6	70	70
	Subtotal	31.6	80	7970
Power	RTG	-110	-110	-110
	Reactor	0	-8000	-8000
	Subtotal	-110	-8110	-8110
	Spacecraft Net Total	-48.4	-270	-63.7

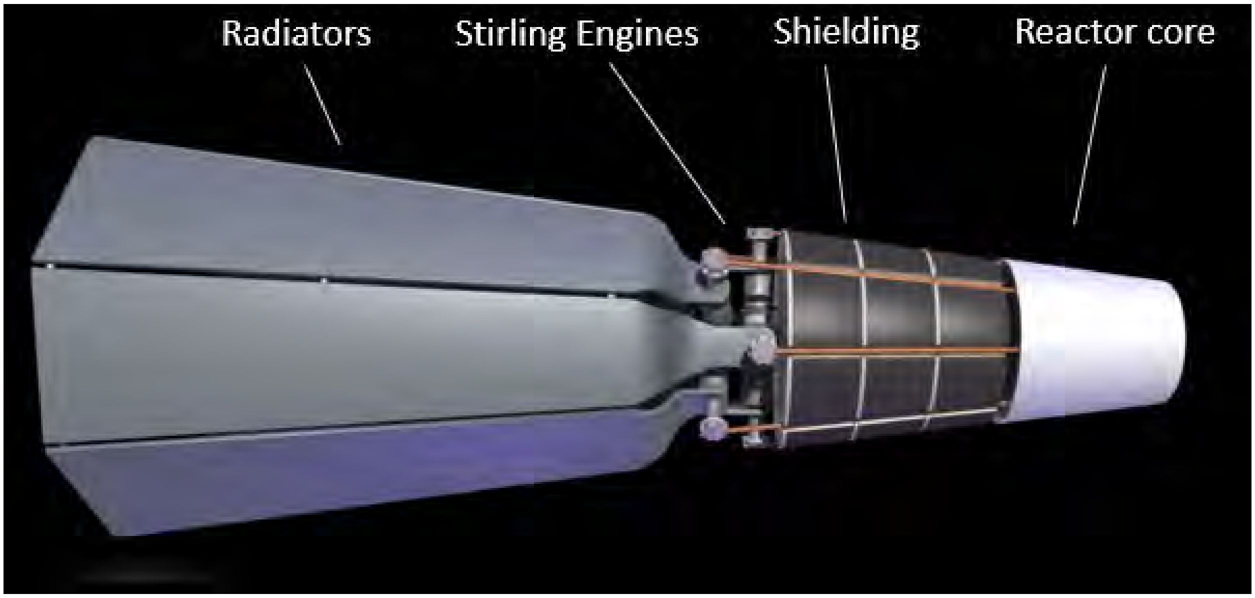


Figure 9: Kilopower nuclear fission reactor. Core and neutron reflector at far right, Stirling generators at center, radiators at left. The radiators remove heat and expel it into space and away from the spacecraft. Only a small portion of heat radiates towards the spacecraft and the exposed hydrazine tank. Image source: Los Alamos National Laboratory.

on board the spacecraft at launch, as well as the heat required to be dissipated. The reactor remains in a shut-down state until it is time to begin the deceleration burn, at which point it provides the additional power required by the engines. All power required during cruise is supplied by the single remaining MMRTG. When Orpheus arrives at Pluto, the excess power is redirected towards communications and operating the scientific instruments. The details of this power budget are shown in Table 6.

8 Thermal Management

Sending an orbiter into deep interplanetary space involves wildly varying temperatures during the mission. The mission requires keeping the spacecraft and its instruments inside its operational temperature range. Table 7 shows the operational and survival temperatures of varying instruments onboard the spacecraft. In order to ensure mission success the thermal subsystem needs to heat and cool the spacecraft depending on the position and internal temperature of the spacecraft.

Knowing the required temperature ranges the spacecraft requires, a variety of methods can be considered in order to maintain a nominal temperature. Thermal insulation on the spacecraft will be needed so that minimal heat is lost to space. Multi-layer insulation (MLI) is used in order to keep the heat flux on the spacecraft to a minimum while also keeping the mass of the thermal system low. The emissivity and the absorptivity using 30 layers of Goldized Kapton are 0.08 and 0.3 respectively while the total mass of the MLI is only 2 kg. The MLI can keep the spacecraft from losing heat, however other systems will be needed to provide internal heat to keep the spacecraft operational.

The main source of heat comes from the sun and its solar irradiance. The solar irradiance throughout the mission can be shown in Figure 10 with major milestones such as launch, Jupiter flyby, and finally the Pluto encounter. The amount of solar irradiance the spacecraft will experience varies from 1,397 $\frac{W}{m^2}$ near Earth down to 0.75 $\frac{W}{m^2}$ at Pluto. A simple model was used to analyze the thermal environment on the surface and inside the spacecraft by using a cylinder with a circumference of 6.73 m and a height of 2.5 meters to model the main bus. Using the temperature ranges shown in Table 7, a range of heat values were calculated in order to keep the spacecraft within the nominal temperature range. Figure 11 outlines the heat required in order to maintain a healthy spacecraft. Table 8 shows the heat requirements for near Earth and near Pluto with the maximum temperature and near Pluto

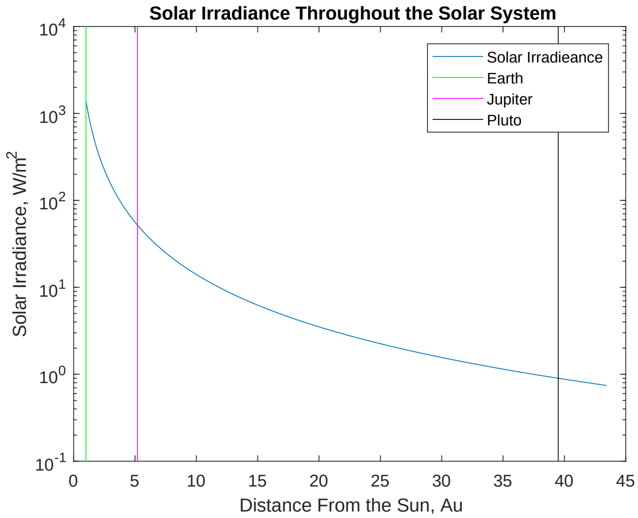


Figure 10: The solar irradiance is shown above as a function of distance from the Sun. The distances of Earth, Jupiter and Pluto are shown as vertical lines on the graph. As expected the solar irradiance decreases as the spacecraft travels further away from the Sun.

with the minimum temperature requirements. In order to satisfy the need to expel heat at the beginning of the mission a set of louvers will be used. Louvers were chosen since the panels can be opened and closed at any time and as the heat needed to expel decreases the louvers can compensate by closing the panels partially. To satisfy the 1,500 Watts excess heat near Earth a total of 4 louvers with an area of 0.84 m^2 each will be used.

Due to the survivable temperature range of the spacecraft, around 1.7 AU from the sun the louvers can completely close. At this point the spacecraft is allowed retain its heat and

Table 7: The Operating and Survival Temperatures of Significant Systems

System	Operating Temperature, K	Survival Temperature, K
Digital Electronics	273 to 323	253 to 343
Analog Electronics	273 to 313	253 to 343
IR Detectors	4 to 94	4 to 305
Particle Detectors	238 to 273	238 to 305
Hydrazine	275 to 387	275 to 387

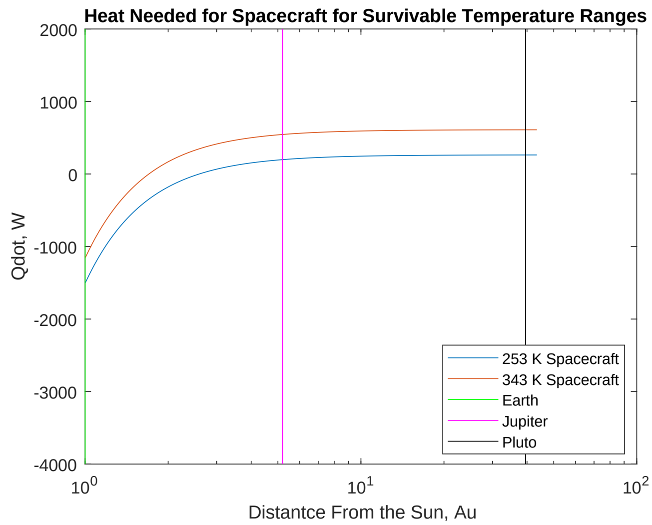


Figure 11: The amount of heat the spacecraft radiates is based on how hot the spacecraft and its surface is. The maximum temperature the spacecraft can survive is shown in red while the minimum temperature is shown in blue.

Table 8: Solar Irradiance at Earth, Jupiter, and Pluto.

Planet	Distance, Au	Solar Irradiance, $\frac{W}{m^2}$
Earth	273 to 323	253 to 343
Jupiter	273 to 313	253 to 343
Pluto	4 to 94	4 to 305

only lose heat through MLI due to radiation. When the spacecraft passes 2.6 Au from the sun, the solar irradiance becomes too small in order to remain above the minimum survivable temperature. The spacecraft passes 1.7 Au 108 days after launch and passes 2.6 Au 205 days after launch. After the spacecraft reaches 2.6 Au, a series of resistive heaters aboard the spacecraft will be turned on. These heaters will be placed on the scientific instruments and near the electronics. These heaters will be powered by the MMRTG and the reactor once it is activated.

Included on the spacecraft is an 8 kW reactor in order to power the ion engines and the communications system once on orbit. The reactor generates heat and then radiates through the attached radiators which is installed onto its side shown in Figure 9. The core of the reactor during full power reaches 1000 Kelvin and the radiators are expected to reach 500 Kelvin. While most of this heat is radiated into space, a small portion is radiated back towards the spacecraft bus. Based on the 500 Kelvin baseline for the radiators, the reactor must be a minimum of 4 meters away from the main bus in order to prevent overheating inside the spacecraft near Earth. A factor of safety of 2 is used in case a truss segment fails to deploy and therefore nominal distance between the spacecraft bus and the reactor is 8 meters.

The attitude control system, as described in Section 11 will use a set of two hydrazine tanks that also need to be heated in order to operate correctly. Hydrazine can stay a liquid between 275 Kelvin and 387 Kelvin which is required for proper thruster operation. The hydrazine tank inside the spacecraft bus is heated passively by the residual heat from the electronics and their respective heaters. The second hydrazine tank is placed just under the reactor in order to provide fuel to the RCS thrusters near the top of the spacecraft. Since the reactor will not be activated until after the flyby of Jupiter, a set of heaters will be needed to keep the hydrazine from freezing from Earth until the start of the deceleration burn near Pluto. Once the reactor activates and begins to radiate heat the hydrazine tank near the reactor will be heated to 373 Kelvin. This is below the boiling point and therefore the hydrazine remains a liquid without the need for the resistive heaters.

9 Structure

The main body of the spacecraft is a rectangular prism with diagonal corners. It is 3 meters long and 1.8 meters wide in width and length. There are large faces on each side that are 1.6 meters wide. The corners on the rectangular prism are cut, resulting in a small 0.28 meter wide side in-between all the larger sides. The majority of the instruments and external systems will be mounted at the flat corners. The MMRTG will be on the opposite side as TIS, to avoid interfering with the thermal sensors. The main instrument boom and the communications dish are on the remaining two corners. This gives each of the instruments plenty of room to move about on their boom.

Overall, the structure is larger than it strictly needs to be, to ensure that there is plenty of space for all the internals inside. The inside of the structure mostly consists of the xenon tank, which, based on the dimensions and mass of the xenon tank offered by Cobham industries [17], will measure 1.4 meters tall and 1.2 meters in diameter and keeps the xenon at a pressure of 8 MPa. Given these parameters, it would weigh approximately 50 kg. On top of the xenon tank is a hydrazine tank that holds half of the 200 kg of hydrazine on the spacecraft. It weighs approximately 15 kg and would be based on the hydrazine tanks made available by Arainegroup [18], but would be flatter, measuring 0.8 meters across and 0.3 meters thick. On top of this tank would be the main computer and relevant electronics. These electronics would be shielded from deep space radiation with layers of aluminum to block out cosmic rays. It also allows a minimal structural support for the low weight components like the computer. Below the xenon tanks will be the ion propulsion system. The bottom half meter of the spacecraft has a pit in which the two ion engines will sit, allowing them room to gimbal in the necessary direction. Just between the pit and xenon tank will be the PPUS for the engines and the rest of the control assembly. It will be partially shielded in the same way that the computers were.

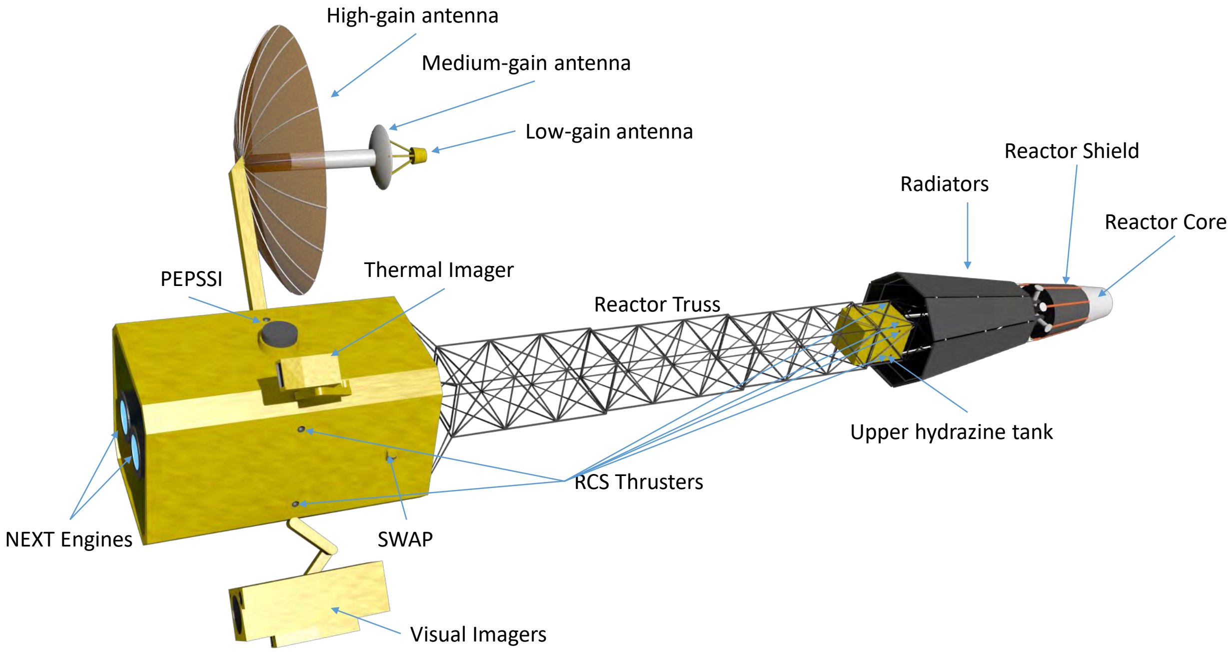


Figure 12: Spacecraft layout with all major external components labeled.

The main structure of the spacecraft is supported by a box truss made of two sections. It carries the stress of launch, both the main axial acceleration and the lateral vibrational loads, which were estimated to max out at 5 gs and 4 Gs respectively. Due to special consideration involving the truss that connects to the reactor, the box truss was chosen over the more tradition design of an aluminum honeycomb for support. With the box truss design, the support structure of the spacecraft bus can smoothly transition into the support structure of the truss supporting the nuclear reactor. The truss inside the spacecraft box is 1.5 meters of each side, which would make the corners reach all the way to the thin diagonal flat panels where the instruments are mounted. This allows the large instruments to be directly connected to the main support structure. At the bottom the truss connects to a stageable support structure that will turn the four points of connection on the bottom of the spacecraft into a single connection point on the other side that will allow the spacecraft to interface with the third stage.

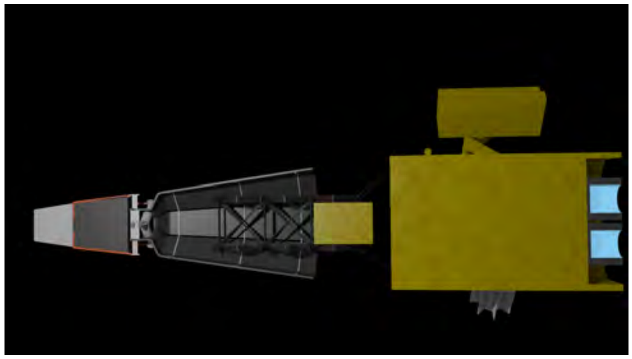


Figure 13: Cutaway of the launch configuration of the spacecraft.

The main support structure will be made of a material known as carbon fiber reinforced cyanate ester. [19] The material is incredibly strong, with a compressive strength of 965 MPa and a tensile strength of 300 ksi. Its thermal properties are also incredibly good, with a thermal expansion coefficient of 1 $\frac{\mu_{strain}}{^{\circ}C}$. Therefore thermal contraction while approaching Pluto due to the reduced temperature is not a substantial problem. Certain parts of the truss are hollowed out for less weight while keeping most of the moment of inertia, otherwise the space of the cross-sections of the individual beams are square. The thickness of each individual beam is determined for each individual beam. The total weight of the system was determined through an approximate method, so a sizable factor of safety of 1.6 was used when determining the total mass of the system.

The reactor has to be separated from the spacecraft bus for thermal concerns, by 8 meters. However, at that length, the entirety of the spacecraft wouldn't fit inside the payload fairing for launch, which is only 11 meters long. Between the 4 meter reactor, the 3 meter spacecraft bus, and the 2.5 meter third stage solid rocket motor, the 7 meter truss had to contract into a space of only 1.5 meters. As a result, the reactor truss is divided into four segments, each a box truss with three segments and 3 meters long. Each truss will be 0.1 meter wider than the one before it, starting out with a small 0.6 meter truss and working the way up to a 0.9 meter truss.

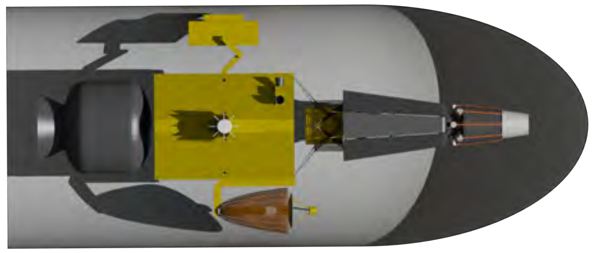


Figure 14: Orpheus stowed in Falcon Heavy payload fairing.

However, since each truss is 2 meters long, having them all stacked together would result in the truss taking up 2 meters of space when they can take up no more than 1.5 meters of space. Therefore, the actual effective length of the truss has been reduced by stuffing as much as possible of it into the gap between the radiators that are part of the reactor. With this method, 1.4 meters of the truss can be stuffed into the radiators, making the effective space taken up by them to only be 0.6 meters, since the reactor will not be on, there is no issue with anything being in that space on the reactor. Once the spacecraft has deployed, the truss will extend, and stay extended forever.

In order to save weight, only one section of the truss will be strong enough to hold the weight of the reactor during launch. Since the reactor truss will be collapsed during launch, the smallest truss can connect directly to the spacecraft bus and the reactor, carrying the loads through it while the other truss pieces will be bypassed. This will allow the individual beams to be much thinner and take up less mass. Thus the other beams will only be strong enough to handle their deployment and the trivial stress experienced while maneuvering the spacecraft.

The load will still have to be transferred to the truss in the spacecraft bus in order be properly supported. Since the load bearing truss is 0.6 meters wide and the spacecraft bus truss is 1.5 meters wide, a diagonal strut at a 45 degree angle connects the corners of the

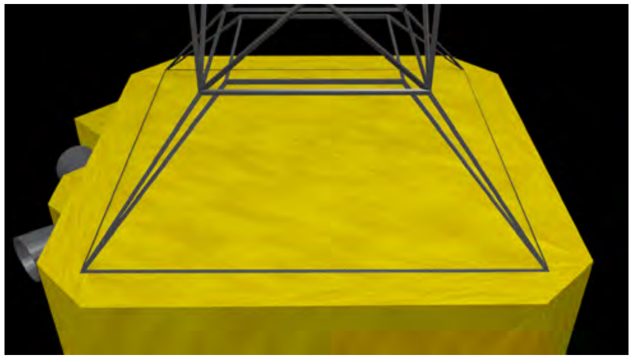


Figure 15: Base of the reactor truss. Note that only the inner section is required to bear launch loadings.

bottom of the reactor truss to the top of the spacecraft bus truss. A groove at the top of these struts allows a solid connection between the strut and the smallest reactor truss segment. Another similar strut connects the largest segment to the same spot of the spacecraft bus to hold the spacecraft together when the truss is fully extended.

10 Communications

Due to the capabilities of the selected instruments and the mission timeline, the amount of data retrieved from the mission will be constrained by the communications system, making the data transmission rate the main driver for this system. The DSN was selected as the ground station due to the distance to Pluto. The 70 m diameter parabolic antennas were used for this analysis, but they will be replaced by arrays of four 34 m diameter parabolic antennas by the time of this mission. However, the new antenna arrays should be able to match or exceed the capabilities of the 70 m antennas. [20]

A 3 dB margin is used for all communication modes in order to obtain the highest data rate possible while still leaving some room for error. The target bit rate error is 10^{-5} dB. BPSK modulation with RS and Viterbi decoding was selected in order to achieve a required E_b/N_0 of 2.7, which is the lowest of all considered modulation techniques. [21] The maximum distance from Earth during the planned mission is used in order to account for space losses.

The communications system was designed for two phases of operation. The first is the low power mode which will be used on the journey to Pluto for telemetry and command. The main driver for this mode is the small amount of power available for the system, 10 Watts. This mode will be used until the end of the slowdown burn, at which point there will be more power available for the system. The second mode is the data transmission mode, which will be used while in system to transmit data back to Earth. For this mode, the nuclear reactor can be mostly dedicated to communications, amounting to 7900 Watts.

A 3 m diameter parabolic high gain antenna will be used for the main data transmission. This size is optimal because large antennas allow for higher data rates, but have more mass and volume. The antenna array on the spacecraft will have a pointing error of only 0.005° in order to maximize data rates. A 0.5 m diameter medium gain parabolic antenna is included as a backup to decrease reliance on this high pointing accuracy. A low gain antenna is also

Figure 16: All of the antennas are mounted on the same array, on a boom connected to the spacecraft bus. The array can rotate independently of the spacecraft using a gimbal system.

included for near Earth communications.

The X-band and Ka-band were both considered based on compatibility with the DSN. The Ka-band was chosen for the primary data downlink because it allows a higher data rate with the antenna array's intended pointing accuracy. The backup system uses the X-band because it allows communications to be maintained with a higher pointing error.

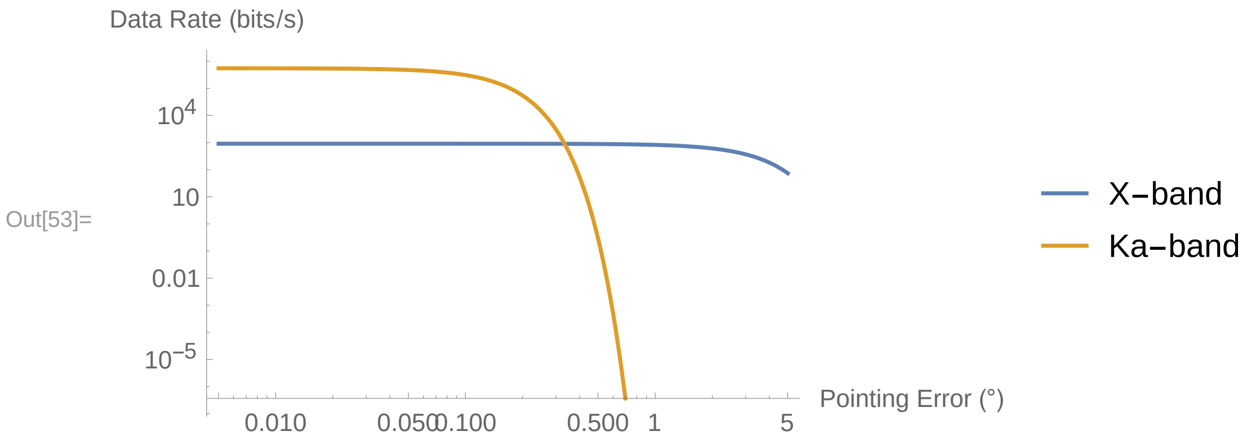


Figure 17: This plot shows how lowering pointing error increases the data rate. For low pointing error, the Ka-band achieves higher data rates, while for higher pointing error the X-band achieves higher data rates. It also shows that decreasing pointing error provides diminishing returns. Finally it demonstrates that even the X-band does not enable communications with pointing errors over 5°.

Table 9: Link budget for cruise, propulsive, and science modes.

	Low Power Mode	Data Transmission Mode	Backup System
Frequency	32 GHz	32 GHz	8 GHz
Power	10 Watts	7900 Watts	7900 Watts
Transmitter Line Loss	-0.5 dB	-0.5 dB	-0.5 dB
Spacecraft Antenna Diameter	3 meters	3 meters	0.5 meters
Transmitter Beamwidth	0.875 deg	0.875 deg	5.25 deg
Transmitter Antenna Pointing Error	0.005 deg	0.005 deg	4.689 deg
Transmitter Gain	57.50 dB	57.50 dB	20.32dB
Path Distance	6.5E9 km	6.5E9 km	6.5E9 km
Space Loss	-318.8 dB	-318.8 dB	-306.8 dB
Propagation/Polarization Loss	-0.51 dB	-0.51 dB	-0.51 dB
Receiver Antenna Diameter	70 m	70 m	70 m
Receiver Pointing Error	0.0005 deg	0.0005 deg	0.0005 deg
Receiver Antenna Efficiency	0.75	0.75	0.75
Receiver Beamwidth	0.0375 deg	0.0375 deg	0.0375 deg
Receiver Gain	74.2122 dB	74.2122 dB	74.2122 dB
System Noise Temperature	800	800	552
Data Transfer Rate	475 bps	375000 bps	100 bps
Margin	3 dB	3 dB	3 dB

11 Attitude Determination and Control

Orpheus uses the same attitude determination solution as New Horizons, two star trackers. These provide redundancy if one fails. However, Orpheus has far more strict pointing requirements with potentially higher rates of rotation. Therefore, instead of a single Inertial Measurement Unit (IMU), Orpheus will contain redundant units to ensure that the orientation is always known to at least some precision.

Attitude control is driven by the pointing requirements of the antenna array, payload instruments, and engines. The engines were placed on the opposite side of the spacecraft bus from the reactor truss in order to align them as closely as possible with the spacecraft's center of mass. The engines must be pointed in the correct direction during the slowdown burn, which will be away from the direction of Earth. The antenna array must always point at Earth to maintain communications, with a desired accuracy of within 0.005°, and a minimum accuracy of within 4.75°. Since the antenna array must be pointed in the opposite direction of the engines, it is placed on a boom to avoid the reactor truss. The payload instruments with pointing requirements are also placed on booms so they can be placed around the spacecraft to move the center of mass closer to the desired location without restricting their view.

Three-axis stability is required because of the different pointing requirements. Sixteen thrusters are used in order to provide some redundancy. Eight of these thrusters are located on the main spacecraft bus, while the other eight are located near the reactor in order to achieve moments without altering the spacecraft's orbit. The two sets of thrusters necessitates having two separate hydrazine fuel tanks in order to avoid having to pipe fuel along the truss, which would introduce significant heating problems to prevent the fuel from freezing. Due to the mass of the spacecraft and the length of the mission, the necessary fuel was estimated at 200 kg of hydrazine, which is divided into two tanks with 100 kg each. Small helium tanks are included for pressure regulation of the hydrazine thrusters, amounting to 5 kg of total mass.

To supplement the thrusters and increase pointing accuracy, four reaction wheels are included in the spacecraft bus, one for each rotational axis and a fourth offset from the others to act as a backup. The thrusters will be used to de-spin the reaction wheels periodically.

12 Data Handling

Solid state drives on-board the spacecraft will be used to store the data gathered by the scientific instruments before being sent back to Earth. For redundancy, two RAD 750 computers will be used to control the spacecraft during operations and handle the data gathered by the instruments. They have been used on numerous spacecraft and are currently one of the best radiation-hardened processors to use. With two computers, it is easier to double check the data and account for errors without shutting down the spacecraft. Onboard processing is crucial to increase efficiency of data transfer and to control the spacecraft.

13 Risk Analysis

The various risks for this project are presented in Figure 18. Failing to launch, missing the trajectory, communication failure, and impact with other objects, pose the largest threat to the mission, but are very unlikely to occur. Single engine failure, the lowest risk, takes into consideration only one engine failing at any given time, and not all three. Computer errors are more than likely to happen, but may be fixed remotely, which may not jeopardize the mission. Having only one RTG reduces the need for plutonium, thus reducing its risk and impact on the mission. It isn't likely for the gimbals to fail, but even if they do, not much can be done once the spacecraft has been launched. This could restrict the functionality of the instruments and would cause the ion engines to be locked in place. The chance of political issues arising due to the fission reactor is likely. Due to the benefits of the fission reactor, we have decided that this is an acceptable risk. Strategies for reducing the various risks, such as redundancy and proper testing, are described in Table 10.

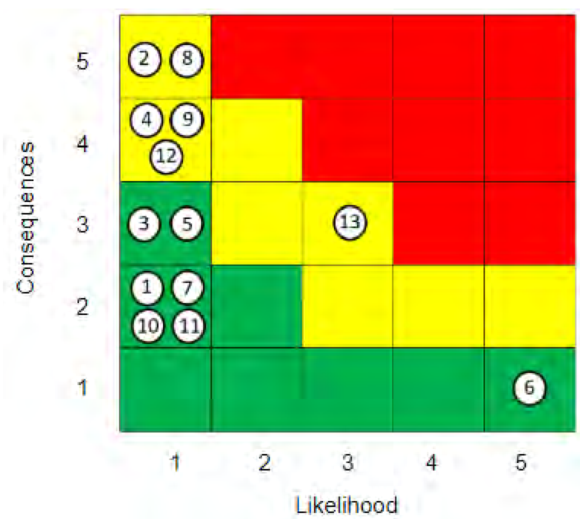


Figure 18: Risks associated with this project: 1. Single Engine Failure 2. Launch Failure 3. Missing Launch Window 4. Trajectory misses 5. Communication Failure 6. Computer Error 7. Attitude Control Failure 8. Impact/Destruction 9. Structural Failure 10. Payload Gimbal Failure 11. Engine Gimbal Failure 12. Plutonium Availability 13. Political Issues.

Table 10: Strategies for risk mitigation.

	Risk	Mitigation
1	Single Engine Failure	Having one backup thruster for redundancy reduces risk of engine failure.
2	Launch Failure	Proper testing and procedures greatly reduce this risk.
3	Missing Launch Window	Extra onboard fuel allows a wider launch window, backup window.
4	Trajectory Misses	Extra fuel to correct trajectory.
5	Communication Failure	Medium gain dish can complete mission with lower data rate, or can be used to fix the error.
6	Computer Error	More than likely to occur, but most errors are fixable. Redundant onboard computer.
7	Attitude Failure	One off-axis backup reaction wheel, as well as redundant thrusters.
8	Impact/Destruction	Careful planning to avoid large objects.
9	Structural Failure	Environmental tests can greatly reduce this risk.
10	Payload Gimbal Failure	Using reliable components reduces chance of failure. The entire spacecraft can be rotated to point the instruments.
11	Engine Gimbal Failure	Using reliable components reduces chance of failure. Limited corrective capabilities based on the RCS.
12	Plutonium Availability	Only needing one RTG minimizes plutonium requirement.
13	Political Issues	Out of our control; acceptable risk.

14 Compliance

Orpheus fulfilled all major RFP requirements, as described by AIAA. These requirements are described in Table 11. The data requirements also described by AIAA are listed in Table 12.

	Requirement	Compliant	Section
1	Collect data for at least one year while in orbit in the Pluto system	✓	4.2
2	Maximum mission time of 25 years	✓	4.1
3	Must use a single launch	✓	6
4	Propulsion system must be of TRL 6 or higher	✓	5

Table 11: Orpheus compliance matrix

	Requirement	Page numbers
1	Key trade studies and a justification for selection of the overall concept and each of the major subsystems	1-43
2	Description of proposed flight sequence and mission timeline	9-13
3	Details of propulsion, vehicle sizing, trajectory, loads, structural, and payload capability analysis. Critical technologies and their current Technology Readiness Level (TRL). Discussion of any required technological breakthroughs or plans for developing technologies to the required maturity	4-43
4	Discussion of design and concept of operation. Systems that are unique to the proposed design, such as vehicle(s), propulsion subsystem(s), propellant and power subsystems, thermal protection subsystem, and communication subsystems should be addressed in considerable detail	9-43
5	Subsystems, such as avionics, guidance, navigation, and control which are not the focus of this project, do not require much attention, unless their mass fraction is expected to have significant mission architecture implications	28-43
6	Discussion of risk mitigation strategies for key technical and programmatic risks	46
7	Drawings of the overall vehicle(s) and key components or subsystems	27,34-37,39
8	Estimate of development and operation life cycle cost	50

Table 12: Orpheus data compliance matrix

This page intentionally left blank

A Mass Budget

Subsystem	Component	Number of Units	Mass per Unit, kg	Total Mass, kg	Margin	Total Mass with Margin, kg	Total Subsystem Mass, kg	Total Subsystem Mass with Margin, kg
Scientific Instruments	Alice (uv spec)	1	4.5	4.50	1.00%	4.545	76.40 (2.14%)	77.16 (2.08%)
	REX (radio)	1	0.1	0.10		0.101		
	Laser Altimeter	1	10	10.00		10.1		
	LOC	1	45	45.00		45.45		
	SWAP (solar wind)	1	3.3	3.30		3.333		
	PEPSSI (neutral atoms)	1	1.5	1.50		1.515		
	66 um Thermal Spectrometer	1	12	12.00		12.12		
Propulsion	Xenon	1	1250	1,250.00	5.00%	1312.5	1,459.80 (40.95%)	1532.79 (41.36%)
	Tank	1	40	40.00		42		
	NEXT	2	13.9	27.80		29.19		
	Xenon Control	1	50	50.00		52.5		
	8.0 kW PPUs	2	40	80.00		84		
	Gimbals	2	6	12.00		12.6		
Attitude Control	Reaction Wheels	4	10	40.00	10.00%	44	270.00 (7.57%)	297 (8.01%)
	Hydrazine	2	100	200.00		220		
	Hyrazine Tank	2	15	30.00		33		
Thermal	MLI	1	2	2.00	5.00%	2.1	2.42 (0.067%)	2.54 (0.068%)
	Heaters	14	0.03	0.42		0.441		
Structure	Spacecraft Bus Structure	1	102.92	102.92	5.00%	108.066	411.40 (11.54%)	431.97 (11.66%)
	Aluminum Shielding	1	50	50.00		52.5		
	Reactor Outer Truss Strucuture	1	168.8	228.48		239.904		
	Reactor Inner Truss Structure	1	30	30.00		31.5		
CC&DH	Comms	1	100	100.00	5.00%	105	160.92 (4.51%)	168.97 (4.56%)
	CPU	2	0.5	1.00		1.05		
	Wiring	1	24.92	24.92		26.166		
	PCU	1	17.5	17.50		18.375		
	Regulators/Converters	1	17.5	17.50		18.375		
Power	RTG	1	44	44.00	1.00%	44.44	1,184 (33.21%)	1195.84 (32.26%)
	8 kW Fission Reactor	1	1140	1,140.00		1151.4		
	Spacecraft Total Mass			3,564.94		3706.27		
	Spacecraft DNE Mass			4,400.00		4400		

B Cost Budget

Component	Cost (Millions $)	Margin	Number of Units	Total Cost (Millions $)	Total Cost with Margin (Millions $)
SpaceCraft Bus	550	20%	1	550	660
Instrumentation	60	30%	1	60	78
MMRTGs	110	15%	1	110	126.5
8 kW Fission Reactor	500	50%	1	500	750
Xenon Propulsion System	40	10%	1	40	44
Falcon Heavy	90	30%	1	90	117
Star 75	20	50%	1	20	30
Ground Maintenance	2.5	20%	30	75	90
Ground Communications for 3 years	125	5%	1	125	131.25
Total Cost				1570	2026.75

[22] [23] [24]

C Requirements Flowdown

1	Return useful data from Pluto for at least one year
1.1	Spacecraft must be able to collect data from the system while in orbit
1.1.1	Spacecraft must have 3-axis stability attitude control
1.1.1.1	Spacecraft must have star trackers and sun sensors
1.1.1.2	Spacecraft must have attitude control thrusters and momentum wheels
1.1.1.2.1	Spacecraft will need hydrazine fuel for RCS
1.1.2	Spacecraft must transit the side of Pluto opposite Earth
1.1.3	Spacecraft must examine full globe of Pluto and Charon
1.1.4	Spacecraft must return data on Plutos outer moons
1.1.4.1	Spacecraft must make a close approach to each of the outer moons
1.2	Spacecraft must have scientific instruments
1.2.1	Spacecraft must be able to record visible-light images
1.2.1.1	Cameras must have unobstructed views of Pluto during operation
1.2.1.1.1	Extendable boom will be mounted on the spacecraft with cameras attached
1.3	Spacecraft must communicate data to Earth
1.3.1	Spacecraft must be compatible with the Deep Space Network
1.3.1.1	Spacecraft must communicate on X-band while at Pluto
1.3.1.1.1	Spacecraft must have an antenna with 3 meter diameter
1.3.2	Spacecraft must be able to store 16 Gigabytes of data
1.3.3	Spacecraft must be able to transmit a minimum 20 Bps at all times
1.3.3.1	The Spacecraft must have a low gain omnidirectional antenna as a backup
1.3.4	Spacecraft must be able to transmit a minimum of 500 Bps for data transfer
1.3.5	Spacecraft must be able to point antenna at Earth
1.3.4.1	The spacecraft antenna must be able to point with a maximum 0.01 degree error

1.3.4.1.1	Double worm gear system provides 2 axis pointing ability
2	Mission must complete within 25 years
2.1	Spacecraft must enter and maintain orbit around Pluto
2.1.1	Spacecraft must have onboard propulsion capable of orbital insertion
2.1.1.1	Onboard propulsion must be capable of 10 km/s dV
2.1.2	Spacecraft must have onboard station-keeping capabilities
2.2	Spacecraft must survive conditions at all points in mission
2.2.1	Thermal protection system must handle incoming heat flux between 0.9 and 1371 w/m2
2.2.1.1	Spacecraft must have heaters
2.2.1.1.1	Small thermal resistors will provide heat to scientific instruments and electrical components
2.2.1.2	Spacecraft must have radiators
2.2.1.2.1	Louvers provide cooling to the spacecraft for near Earth operations
2.2.2	All components must be able to survive the entire mission
2.2.2.1	The computer system must have a complete backup due to radiation concerns
2.3	Spacecraft must accomplish orbital plane change to match Pluto inclination
2.3.1	Spacecraft must conduct a flyby of Jupiter
2.3.1.1	Launch must occur in December 2028
2.3.1.2	Spacecraft must conduct targeting burn for Jupiter flyby
3	Mission must launch on a single launch vehicle
3.1	Launch C3 energy must be below 150 km2/s2
3.1.1	Atlas V 551 will launch the spacecraft onto a Jupiter flyby trajectory
3.1.1.1	Spacecraft must fit within a 4.57 meter diameter fairing
3.1.1.2	Spacecraft must survive launch loads

3.1.1.2.1	Spacecraft must survive acceleration loads
3.1.1.2.1.1	Spacecraft must survive axial acceleration loads of 6 gs
3.1.1.2.1.2	Spacecraft must survive lateral acceleration loads of 4 gs
3.1.1.2.2	Spacecraft must survive vibrational loads
3.1.1.2.2.1	Spacecraft must survive axial vibrational loads of 15 Hz at 0.6 Gs
3.1.1.2.2.2	Spacecraft must survive lateral vibrational loads of 8 Hz at 0.4 gs
4	Propulsion hardware must be TRL 6 or higher
4.1	All propulsion hardware must be flight-proven
4.1.1	Propulsion must be chemical or ion

References

[1] Nola Taylor Redd. Clyde Tombaugh: Astronomer Who Discovered Pluto. Feb 2013.

[2] Rosanna L. Hamilton. Dwarf Planet Pluto. http://solarviews.com/eng/pluto.htm.

[3] Tricia Talbert. New Horizons: The First Mission to the Pluto System and the Kuiper Belt. https://www.nasa.gov/mission_pages/newhorizons/overview/index.html, Aug 2017.

[4] NASA. HIRISE: High Resolution Imaging Science Experiment. https://marsoweb.nas.nasa.gov/HiRISE/.

[5] John F Cavanaugh, James C Smith, Xiaoli Sun, Arlin E Bartels, Luis Ramos-Izquierdo, Danny J Krebs, Jan F McGarry, Raymond Trunzo, Anne Marie Novo-Gradac, Jamie L Britt, Jerry Karsh, Richard B Katz, Alan T Lukemire, Richard Szymkiewicz, Daniel L Berry, Joseph P Swinski, Gregory A Neumann, Maria T Zuber, and David E Smith. The Mercury Laser Altimeter Instrument for the MESSENGER Mission. http://www-geodyn.mit.edu/cavanaugh.mla.ssr07.pdf.

[6] General Mission Analysis Tool (GMAT) User Guide. http://gmat.sourceforge.net/docs/R2016a/help.html#ReleaseNotesR2016a.

[7] Press Release NL-45. https://history.nasa.gov/conghand/propelnt.htm, 1958.

[8] V. Rachuk and N. Titkov. The First Russian LOX-LH2 Expander Cycle LRE: RD0146. http://rocket-propulsion.info/resources/articles/AIAA-2006-4904_RD0146.pdf, Jul 2006.

[9] Cassini Mission Plan. http://caps.space.swri.edu/caps/tourInfo/Documents/MP_0.pdf, Aug 2005.

[10] Electric propulsion data sheets. http://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/Electric%20Propulsion%20Data%20Sheets.pdf.

[11] Michael J. Patterson and Scott W. Benson. NEXT Ion Propulsion System Development Status and Performance. Jul 2007.

[12] Busek's Hall Effect Thruster Technology saves Air Force AEHF satellite. http://www.busek.com/news__201203__aehf.htm.

[13] Leading the Way to Electric Propulsion in Belfast. https://www.surrey.ac.uk/ssc/research/astrodynamics/spaceplan2020/files/SP2020%20Presentation%2007.pdf, February 2014.

[14] Propulsion Products Catalog. Oct 2016.

[15] http://www.spacex.com/falcon-heavy.

[16] Marc A Gibson, Steven R Oleson, David I Poston, and Patrick McClure. NASA's Kilopower Reactor Development and the Path to Higher Power Missions. https://ntrs.nasa.gov/search.jsp?R=201700020102018-01-22T23:22:13+00:00Z.

[17] Xenon propellant tank. http://www.cobham.com/mission-systems/composite-pressure-solutions/space-systems/xenon-propellant-tank-datasheet/docview/.

[18] Hydrazine propellant tanks. http://www.space-propulsion.com/spacecraft-propulsion/hydrazine-tanks/index.html.

[19] CES Edupack.

[20] Thuy Mai. Deep Space Network Aperture Enhancement Project (DAEP). url:https://www.nasa.gov/directorates/heo/scan/services/networks/txt_daep.html, 2014.

[21] Wiley J. Larson and James R. Wertz. *Space Mission Analysis and Design.* Kluwer Academic Publishers, Microcosm Inc.

[22] Final Report of the New Horizons II Review Panel. May 2005.

[23] David Manzella. Low Cost Electric Propulsion Thruster for Deep Space Robotic Missions.

[24] James E Werner, Stephen G Johnson, Carla C Dwight, and Kelly L Lively. Cost Comparison in 2015 Dollars for Radioisotope Power SystemsCassini and Mars Science Laboratory, July 2016.