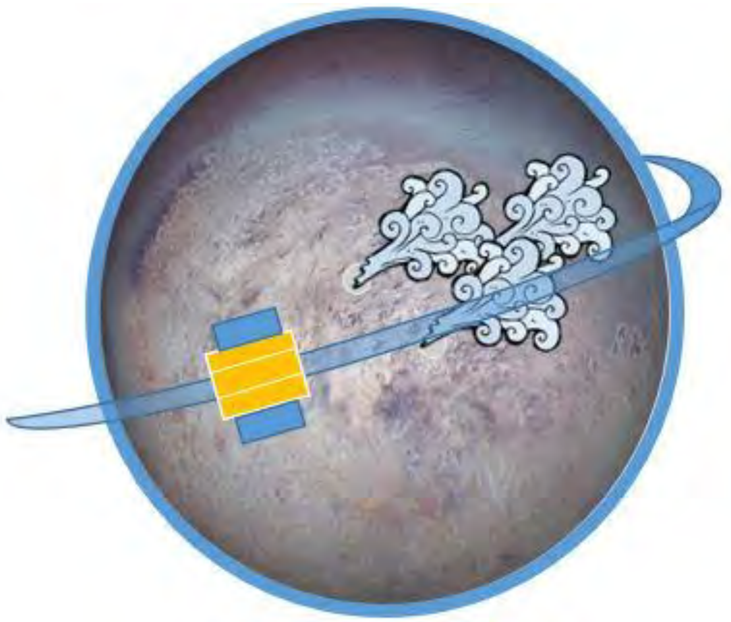
AIAA UNDERGRADUATE TEAM SPACE DESIGN COMPETITION

PROJECT TES
TRITON EXPLORATION SYSTEM

PROPOSAL FOR THE DESIGN OF AN SLS SPACE MISSION



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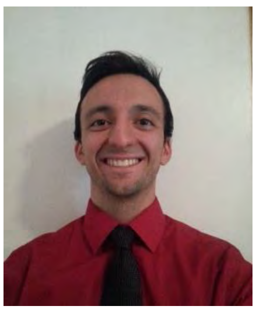

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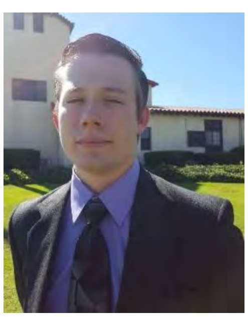

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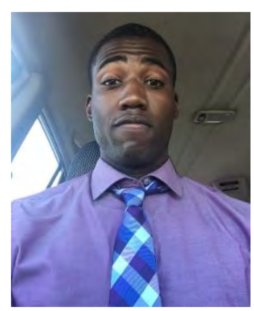

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Executive Summary

This proposal, Project TES, for a space vehicle and mission design has been created by Pulsar Enterprises in response to two requests for proposals: AIAA's Exploration Enabled by Space Launch System and JPL's Project Haukadalur: Distant Geysers, Triton Exploration. Utilizing the SLS Block 1B lifting capability, our Triton Exploration System (TES) will travel to Triton, Neptune's largest moon, to investigate its mysterious geysers and map its surface. Our space system consists of two vehicles, the TES Orbiter and the Vespucci Lander. The TES Orbiter will carry a payload suite consisting of six science instruments, which will fulfill JPL's RFP requirements. The Vespucci Lander will carry a payload suite of four instruments and provide close-range surface composition analysis of Triton's soil and never-before-seen high-definition pictures of the surface, including a 360° panorama view.

Upon approval, TES will begin detailed design in May 2017, begin manufacturing and testing in May 2019, and prepare for launch in March 2022. TES will then launch on the SLS on April 12, 2022 into a Lambert transfer to Saturn, perform a flyby of Saturn to alter its course towards Neptune on March 4, 2025, capture into a Neptunian orbit on July 12, 2035, and finally transfer and enter a circular polar orbit around Triton on September 1, 2035. TES will have begun science operations prior to arrival at Triton, but once there, the full payload suite will collect and transmit all scientific data through the Deep Space Network (DSN) back to Earth. In the summer of 2036, one year after arriving at Triton, Vespucci will detach and descend towards the surface of Triton. Once landed, its full payload suite will activate and much more additional data will be recorded and relayed back to TES. All data will be transmitted back to Earth by 2040. Once all operations are complete, TES will enter a graveyard orbit until its RTG power supply diminishes. Per the JPL RFP, the project budget was capped at $5 billion (2016 USD), and current cost analyses put Project TES at a total cost of $3.63 billion (2016 USD). Given these two RFPs, Pulsar Enterprises has successfully developed a complete interplanetary space vehicle and mission that will provide invaluable data about the outer edges of our Solar System and allow our scientists to understand much more about the universe.

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Nomenclature, Abbreviations, and Acronyms

ACS	Attitude Control System
AIAA	American Institute of Aeronautics and Astronautic
ASRG	Advanced Stirling Radioisotope Generator
ATLA	Assembly, Test, and Launch Operations
BOL	Beginning of Life
C&D	Command and Data
CCD	Critical Command Decoder
CDR	Critical Design Review
CER	Cost Estimating Relationships
CSS	Coarse Sun Sensor
DCSS	Delta Cryogenic Second Stage
DSN	Deep Space Network
DSS	Digital Sun Sensor
eMMRTG	Enhanced Multi-Mission Radioisotope Thermoelectric Generator
EOL	End of Life
EUS	Exploration Upper Stage
FOV	Field of View
FSS	Fine Sun Sensor
GPHS RTG	General Purpose Heat Source RTG
He	Helium
HGA	High Gain Antenna
HRCC	High Resolution Color Camera
IMU	Inertial Measurement Unit
IR	Infrared “or” Infrared Radiation
JPL	Jet Propulsion Laboratory
LDA	Local Dust Analyzer
LIS	Lander Infrared Spectrometer
MAGIC	Magnetic Investigator and Characterizer
MATLAB	Matrix Analyzing Tool Laboratory
MHW RTG	Multi-Hundred Watt RTG
MLI	Multi-Layered Insulation
MMRTG	Multi-Mission Radioisotope Thermoelectric Generator
MON-3	Mixed Oxides of Nitrogen-3
NASA	National Aeronautics and Space Administration
NC	Normally Closed
NICM	NASA Instrument Cost Model
NO	Normally Open
PDR	Preliminary Design Review
PDS	Planetary Data System
PDU	Power Distribution Unit
PO	Programmatic Opportunity
PR	Progress Review
PR	Programmatic Risk
PRU	Power Regulatory Unit
R&D	Research and Development
RAT	Radio Atmospheric Testing
RFP	Request for Proposal
RHU	Radioisotope Heater Unit
RTG	Radioisotope Thermoelectric Generator
SDR	Space Design Review
SDST	Small Deep Space Transponder

SLS	Space Launch System
SSPA	Solid State Power Amplifier
SSR	Solid State Recorder
TDA	Triton Dust Analyzer
TE	Thermo-Electric
TES	Triton Exploration System
TO	Technical Opportunity
TR	Technical Risk
Triple-S	Subsurface Surveyor
TRL	Technology Readiness Level
USCM8	Unmanned Spacecraft Cost Model, 8th update
USD	United States Dollars
UVS	Ultraviolet Spectrometer
VIS	Visible and Infrared Spectrometer
WBS	Work Breakdown Structure

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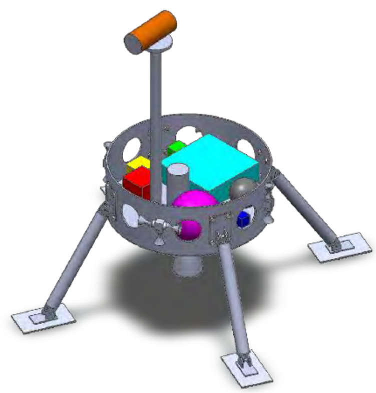
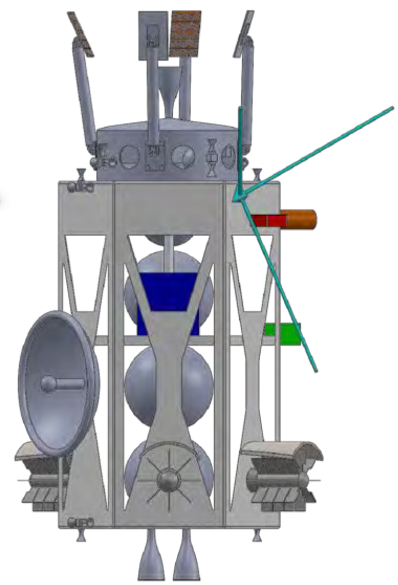
Pulsar Enterprises

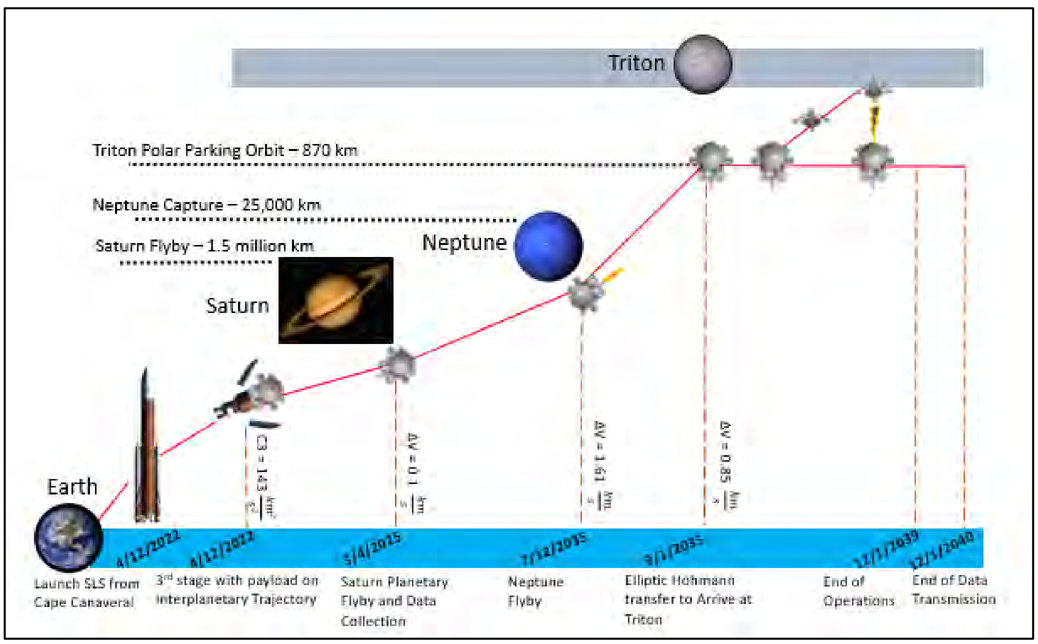
Project TES - Triton Exploration System Orbiter & Vespucci Lander

Mission Objective - AIAA	Mission Objective - JPL
Design an interplanetary space mission enabled by NASA's Space Launch System	Investigate and return data on the atmosphere, surface, and active geysers seen on the surface of Neptune's largest satellite, Triton.

Project Specifications	
Orbiter Dry Mass	763 kg
Lander Dry Mass	139 kg
Total Liftoff Mass	3068 kg
Launch Vehicle	SLS Block 1B
Spacecraft Height	5 m
Spacecraft Width	2 m
Power	2x MMRTG, 2x eMMRTG
Propulsion	Hydrazine, MON-3
Distance from Earth	4.7 billion km
Launch Date	April 12, 2022
Mission Target	Triton
Mission Duration	18 years
Program Cost	$3.63B







TES Specifications	
Subsystem	Mass, kg
Power	201
Structures	186
Thermal	20
Propulsion	160
CD&H	7.5
ACS	82.5
Telecom	70
Cabling	11.2
Payload	24
Lander	210

2016						2017	2018	2019	2020	2021	2022	2023-2040
MCR	ACR	SRR		SDR	PDR		CDR		PRR	ORR		

Mission Analysis Definition	Define Mission Req.	Concept Definition	Concept Development	Preliminary Definition	Detailed Definition & Fabrication	Mission Operations and Ground Support

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Extended Summary

Pulsar Enterprises was challenged with the task to develop a design for an interplanetary space vehicle using SLS as a launch vehicle. The design requirements were provided by AIAA's Undergraduate Student Space Design Competition, as well as a spacecraft design RFP from NASA JPL. AIAA asked for an interplanetary space vehicle enabled by the new Space Launch System (SLS) as its launch vehicle. The RFP provided by JPL had a considerably larger number of constraints, with the main goal being that a spacecraft must travel to Neptune's largest moon, Triton, and investigate unusual geysers and other thermal activities on its surface. The JPL RFP outlined 15 specific requirements, which were further broken down into primary and derived requirements. The design of this system was completed to ensure that all requirements, both for AIAA and JPL, were met. The most limiting requirement from JPL is that the spacecraft must arrive at Triton no later than December 1, 2035, meaning our launch window is very slim, leaving few options for a reasonable launch date and trajectory. To respond to both RFPs, we developed Project TES, short for Triton Exploration System.

Our spacecraft launches on April 12, 2022, giving us a development cycle of about 5 years. After completing a flyby of Saturn on March 4, 2025, our spacecraft will arrive at Neptune on July 12, 2035. With this trajectory, our spacecraft will have a C3 energy of 143.46 km/s as it leaves Earth's sphere of influence heading towards Saturn. Thanks to the Saturn flyby, the total required ΔV of the spacecraft is 2.99 km/s for its entire mission lifetime.

From the JPL RFP requirements, it became clear that our spacecraft needed the ability to directly analyze the surface of Triton. Due to this requisite, three architectures were created for Project TES, with all being capable of studying the soil, atmosphere, geysers, and overall environment of Triton, both remotely and directly. Each architecture demonstrated a vehicle that traveled to the surface of Triton, albeit in different ways, provided in our *Mission Architecture Down-Select* section. For our final architecture, we chose a system consisting of an orbiter and a lander. Thus, our spacecraft will be comprised of two main pieces: the orbiting module called *Triton Exploration System Orbiter*, which we will call *TES Orbiter* or *TES* for short, and a landing vehicle we have named *Vespucci Lander,* just *Vespucci* for short, in honor of the Italian explorer and cartographer, Amerigo Vespucci. The TES Orbiter will take scientific measurements and data from above the moon with its remote instrumentation, while Vespucci separates and lands on the surface of Triton to take scientific data and perform experiments with its direct experimentation. Vespucci's data will be transmitted to TES for processing, compressing, and transmitting back to Earth.

While a trade study of launch vehicles was performed, the requirement to use the SLS made the task of selecting a launch vehicle seemingly trivial. However, it was important to study the SLS and its launch capabilities in detail. The SLS Block 1B was the final selected version of the SLS because of its large launch capabilities and availability by 2022. The largest concern was that even with this version of SLS, we would still not have a large enough launch capability for our spacecraft. Therefore, an Orbital ATK Star 37 kick motor was added as an upper stage vehicle, raising our effective C3 capability to 160 km/s, giving us a total possible launch weight of 3375 kg.

The payload instruments for the spacecraft were chosen to directly fulfill both the primary and derived requirements from the JPL RFP. The instruments' scientific data recording focuses mainly on mapping the surface of Triton, as well as investigating the unusual geyser activity on the surface. There is a total of ten instruments implemented onto our system, with six of them located on TES and the other four located on Vespucci. The instruments on the TES Orbiter are as follows: a visible light camera for pictures of the moon and to map its surface, a visible and infrared mapping spectrometer to analyze the thermal properties and composition of the surface, an ultraviolet spectrometer which will investigate the atmosphere and geyser plumes, a dust analyzer to determine the composition of any trace atmospheric particles, a radio science payload item to further study the atmosphere, and finally, a magnetometer to create an accurate model of Triton's very strong, very anomalous magnetic field. The Vespucci Lander also includes the same visible light camera as TES, as well as a dust analyzer for surface soil analysis. There will also be a surface infrared spectrometer and a subsurface radar to investigate what is below the moon's surface.

The total dry mass of the spacecraft before launch is 973 kg, and the total launch mass is 3068 kg. This allows for a margin of 307 kg under our launch mass capability.

The power requirement of our system is 373 W. However, this is assuming that all of the systems on the spacecraft will be operating at the same time and at peak power. At the beginning of life of the mission, for RTG power units will provide 440 W to the spacecraft. By the end of the mission, the RTGs will only be outputting 217 W, thus the need for different operating modes during the length of the mission is necessary to effectively manage our available power.

The propulsion system was chosen based on the total ΔV of the spacecraft during the lifetime of the mission, which was 2.99 km/s. A bipropellant hydrazine and MON-3 system utilizes an Aerojet AMBR thruster to perform the main thruster burns. The feed system is a dual-mode system that is shared by both the main propulsion and the attitude control system on the spacecraft.

The attitude control system (ACS) shares the dual-mode feed system using a monopropellant fired through 12 MR-111C engines, giving the spacecraft complete three-axis control. That three-axis control is also maintained by four reaction wheels, two magnetic torque rods, three star trackers, and one sun sensor. The components of the ACS system provide the necessary pointing accuracy for the spacecraft's instruments and telecommunications system.

In orbit, TES must have the ability to continuously record, compress, save, and transmit the science data from the instruments back to the Deep Space Network (DSN) on Earth. The maximum data rate from the instruments to the main computer will be 10.08 megabits per second, and that data will be packaged and stored on two redundant 12-Terabyte rad-hardened hard drives using Reed-Solomon turbo coding. For 8 hours of allowed DSN usage per day, a gimbal mounted 3-meter high-gain antenna aboard TES will transmit pictures and data from Triton back to the DSN on Earth at a max data rate of 13 kilobits per second. Due to the vast distance between Earth and Triton, it will take approximately 2.84 years to deliver one full mapping of Triton's surface. In addition, when the Vespucci Lander has completed its science mission, it will transmit all of its data back to the TES Orbiter to be relayed back to the DSN.

Thermal control was necessary to keep the spacecraft within operating limits at different places in the solar system. To help balance the temperature of our spacecraft, it will be coated in a thin, lightweight film known as Kapton on 5-mil Chromium vapor deposited coating. At Earth, the spacecraft is extremely warm due to Earth's IR emissions and the close distance to the Sun. During transit, the spacecraft will be drifting farther and farther away from the Sun, and its overall temperature will begin to drop drastically. As the spacecraft reaches Neptune and transfers to Triton, it will experience its final temperature drops. Due to the temperature limits of the payload instruments, the spacecraft must be cooled during its departure from Earth and heated as it transfers to and arrives at Triton. To accomplish this, a passive thermal control system will be implemented. For heat dissipation at Earth, our spacecraft will use a radiator. For heating at Triton, temperature will be maintained with our four RTGs and 16 RHUs spread around the spacecraft.

The TES bus will be a hexagonal structure that will carry all the instruments. The structure will be made of 7075-Aluminum alloy and will have a mass of 130 kg. The support structures for each system add up to a total of 56 kg. In addition, the main body will have cutouts on each side to reduce the overall mass. Also, there will be three support beams along the long axis of the bus and a solid plate in the middle to strengthen the structure and prevent it from buckling.

Total cost for the mission is restricted by the RFP to be under 5 billion US 2017 dollars. A combination of two NASA cost models have been used to estimate the total cost of the mission from start to finish. These cost models include all spacecraft components, the launch vehicle, all ground support, software costs, and development costs. Using these cost models, the total budget for Project TES came out to be $3.63 billion, well under the cap of $5 billion. When compared to similar interplanetary missions such as Cassini, New Horizons, and Juno, the estimated cost of our mission was found to be within a reasonable range.

1.0 Science Overview

1.1 RFP Background

This proposal has been created in response to two RFPs: AIAA Exploration Enabled by Space Launch System and JPL Distant Geysers Project Haukadalur. AIAA's request for proposal dictates the primary use of the NASA SLS for human space exploration beyond Earth's orbit, but its capabilities are extended to deep space mission. The AIAA RFP has allowed the project team to choose or develop its own mission as long as it utilizes the SLS. This lead to the choosing of Project Haukadalur, JPL's RFP. This RFP dictates the design of a mission to Neptune's moon, Triton, to conduct detailed scientific research based on significant findings from previous deep space missions. Our response to both of these RFPs is Project TES, which we have designed to meet both sets of requirements laid out in each RFP.

1.2 Neptune and Triton Previous Missions and Scientific Information

The Voyager 2 mission is arguably the most prominent deep space mission to date. Primarily designed to study the edge of the solar system, Voyager 2 was also able to take advantage of the alignment of the gas giants to perform studies on all of them. This spacecraft is the only man-made object to have studied Neptune at a close distance, doing so in the summer of 1989.[1] During its time with Neptune, Voyager 2 discovered five moons; of these moons, Triton is the largest. Triton is also one of the coldest known celestial bodies in our solar system, carrying a nitrogen ice "volcano" on its surface, as well as active geysers. These geysers were seen primarily in the southern hemisphere region of Triton, and the eruption clouds were speculated to be trapped beneath a thermopause in the atmosphere.[1] The geological activity on Triton is the central motivator behind Project TES.

In addition to Voyager 2, the more recent New Horizons spacecraft (whose goal was to explore Pluto and other Kuiper Belt objects) performed a distant flyby of Neptune in 2014. During this flyby, New Horizons took a few snapshots of Neptune and Triton. Analysis following this event, scientists suggested Triton could share similarities to Pluto: an icy surface, bright poles, nitrogen atmosphere, and the presence of “ice volcanoes”. Furthermore, Triton is only slightly larger than Pluto in terms of diameter.[2]

Currently, there is not much known about Triton. Its atmosphere and surface composition are all based on the minimal data taken from Voyager and New Horizons. It is believed that Triton has a pinkish color, which suggests the presence of methane gas and ice in addition to nitrogen — a feature that makes it unique from other known moons.[3] Furthermore, it is known to be the only large moon with a retrograde. This orbit could be due to being captured by Neptune from the Kuiper Belt — the same source of Pluto — rather than being a body that naturally formed near Neptune.

1.3 Mission Concept of Operations

Project TES utilizes the SLS Block 1B to propel the spacecraft into a trajectory towards Saturn, and ultimately, to Triton. Our orbiter will map and study Triton’s surface, atmosphere, and most importantly, its interesting geyser activity. This will be accomplished using the various instruments onboard the TES Orbiter. After the Vespucci Lander has detached, both spacecraft will continue to study Triton, with the Vespucci Lander conducting its operations on the surface. TES will continuously transmit back to the Deep Space Network on Earth until its end of life in December, 2040. The TES Orbiter will then dispose of itself into a graveyard orbit around Triton and fully abide by the established Planetary Protection Protocols.[4] Figure 1.3-1 portrays a broad view of the entire mission concept of operations.

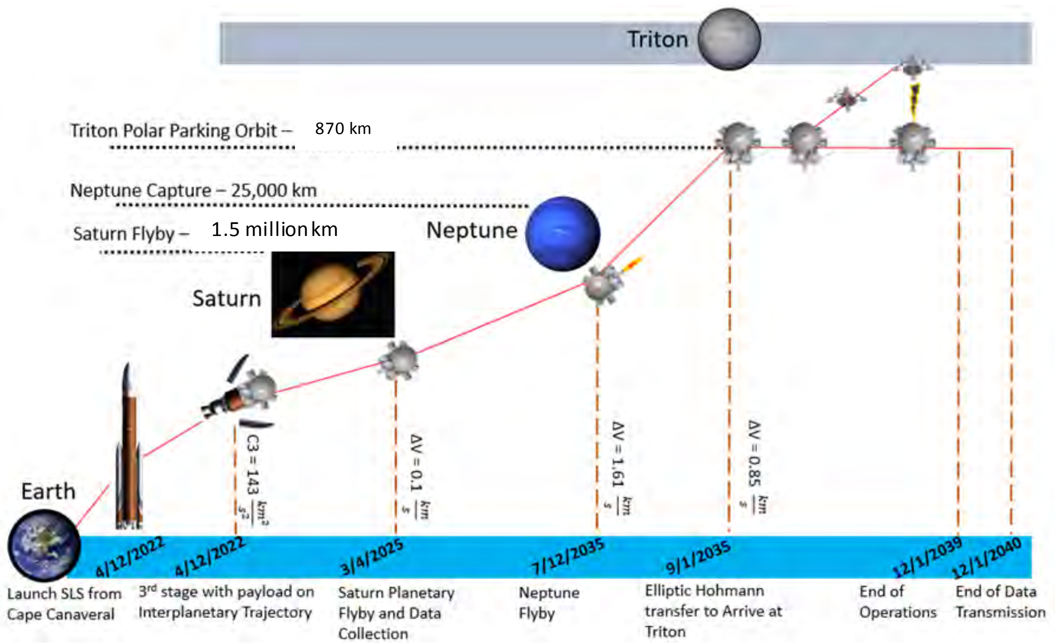


Figure 1.3-1 *Spacecraft Lifetime Concept of Operations*

TES will launch on April 12, 2022 aboard the SLS Block 1B from Cape Canaveral, Florida. The SLS ascent profile is shown in Figure 1.3-2, from the SLS Payload Planner's Guide.[5] The Orbital ATK Star 37 XFP upper stage engine will give the spacecraft a final boost towards Saturn for a total C3 energy of 143 km^2/s^2.

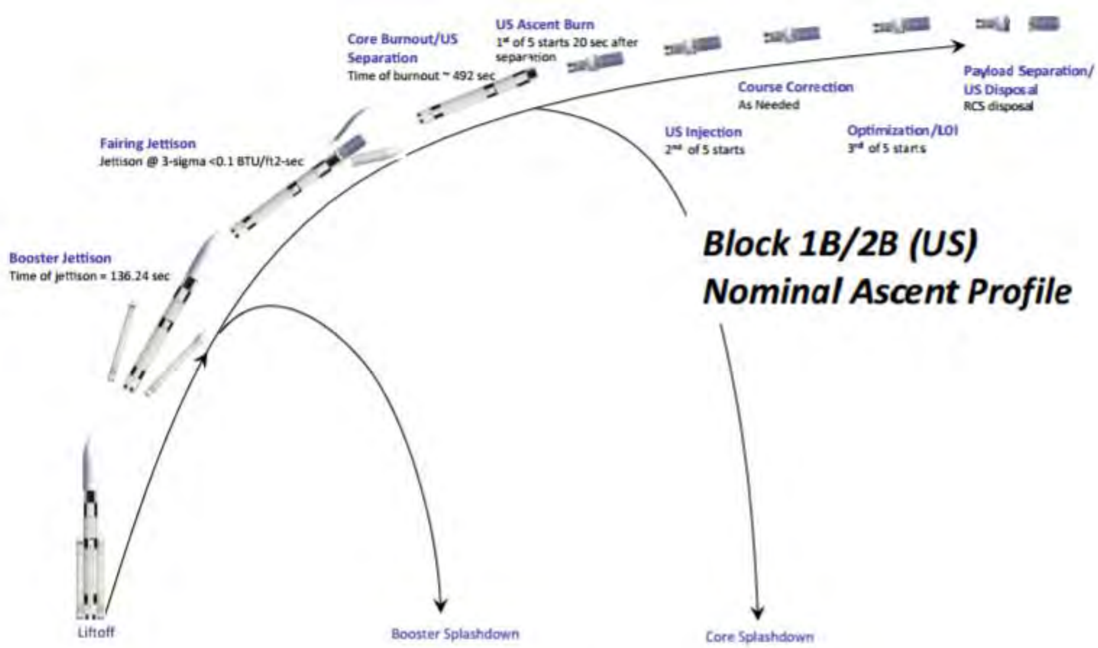


Figure 1.3-2 *SLS Block 1B Ascent Profile*

On March 4, 2025, the TES Orbiter will conduct a flyby of Saturn at 1.5 million km altitude, with a small ΔV burn of 100 m/s for course correction towards Neptune. Saturn science data will be collected and transmitted back to Earth via the DSN.

On July 12, 2035, TES will arrive at Neptune with a velocity of 8.46 km/s and perform a retrograde ΔV burn of 1.61 km/s to place itself in an elliptical orbit around Neptune. The orbital period of this orbit will be 12 days, with a periapsis of 25,000 km. Figure 1.3-3 shows the Neptune capture and Triton transfer orbits in sequential order. TES will now start collecting science data around Neptune. When Triton and Neptune align properly, TES will perform an elliptical Hohmann transfer, with total ΔV burn of 850 m/s, to place itself into polar parking, circular, polar orbit with an 870 km altitude.

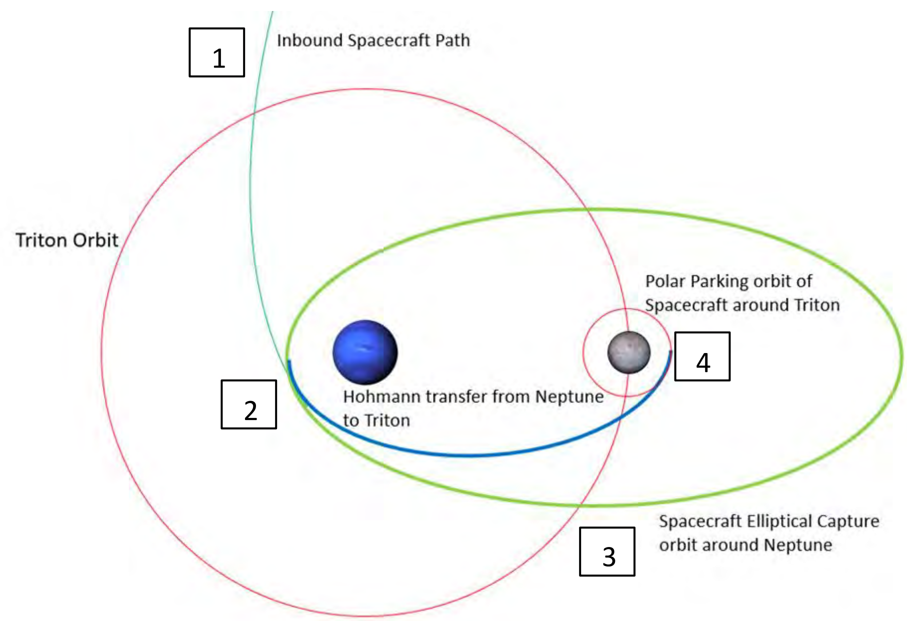


Figure 1.3-3 *Neptune Capture and Triton Transfer*

Once TES reaches its Triton orbit on September 1, 2035, it will begin taking data following the schedule of its first, beginning of life operational mode, Operational Mode 1 (see Section 1.4). The TES Orbiter will also transmit its data back to the DSN as it collects it.

In the summer of 2036, about a year after arriving at Triton, we are expected to have identified multiple landing sites for the Vespucci Lander. Once a landing site is finalized, Vespucci will detach from the TES Orbiter and initiate a ΔV burn of 105 m/s in the opposite direction of travel using its main thruster. This will put Vespucci on an elliptical trajectory towards Triton's surface. At a designated altitude of 200 m, Vespucci will perform another ΔV burn of 700 m/s to bring velocity to 5.00 m/s, relative to Triton. It will then be in free fall until it reaches 10 m, where it will conduct its final ΔV burn of 17.5 m/s in an upward direction. The kinematic energy of Vespucci at the moment of surface impact will be 0.055 J, which is well below NASA's criteria of 100 J for spacecraft survival.[6] The descent will take a total of 1.05 hours. Once Vespucci is on the surface, it will collect and transmit data for approximately 12 hours. Figure 1.3-4 presents a visual representation of the Vespucci Lander concept of operations.

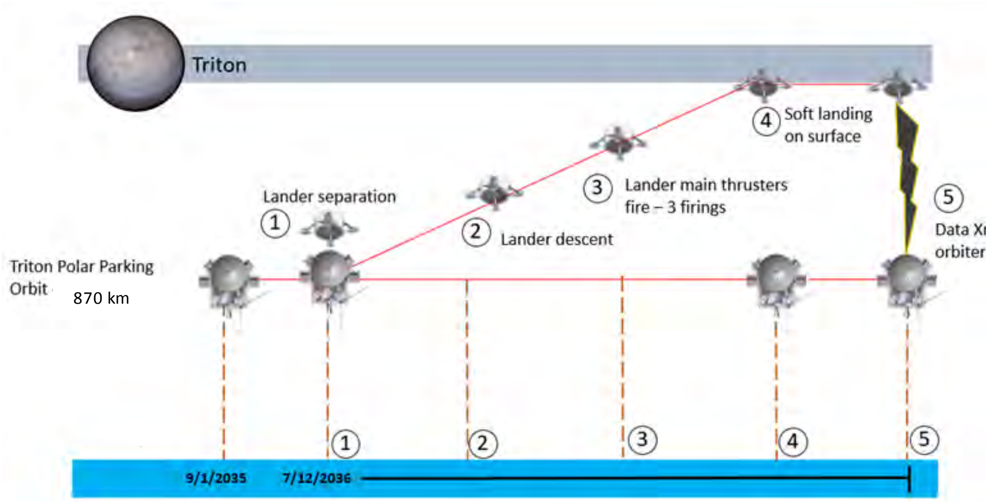


Figure 1.3-4 *Vespucci Lander Concept of Operations*

Vespucci will continuously take data and transmit all necessary data back to TES when in view for 1.51 hours per orbit, and will continue transmitting for each subsequent pass-over until all data has been transmitted or until battery power ceases. Figure 1.3-5 shows the field of view between TES and Vespucci. TES will process, store, and relay Vespucci's data back to Earth.

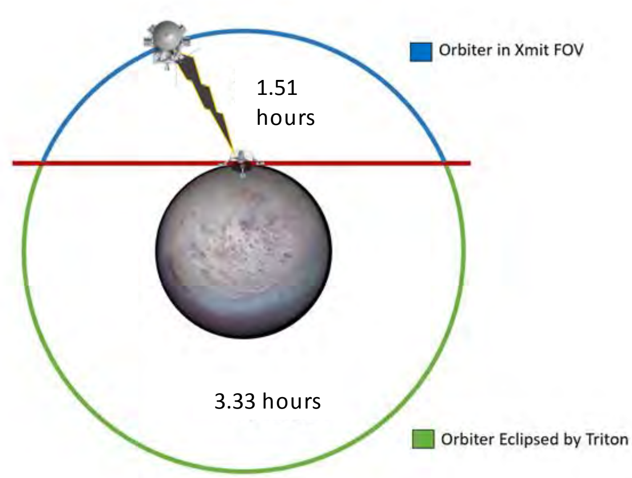


Figure 1.3-5 *Vespucci Lander Transmission Operations*

On December 1, 2039, the science data collection will end, but the telecommunications system will continue to transmit data via DSN. About a year before that, and as RTG power output continues to decay, the TES Orbiter will no longer be able to operate multiple subsystems at the same time. Thus, it will enter Operational Mode 2 (see Section 1.4). The last year is dedicated solely to transmission. Finally, on December 1, 2040, all data transmission will cease. The TES Orbiter will move into a graveyard orbit until the RTGs can no longer provide sufficient power to operate.

1.4 Operational Modes

During the various phases of the mission, different spacecraft modes must be implemented to efficiently distribute power to the instruments and other components of the spacecraft. The different modes have been broken down into two operational modes with two science modes each.

Each mode was chosen based on the spacecraft's position in orbit, our DSN usage per day, and the available power. The illumination period of our 4.84 hour orbit is 3.83 hours, leaving about 1.01 hours of eclipse period. Throughout the life of the mission, we will continuously have 8 hours of DSN usage per day. However, RTG power starts at 440 W at beginning of life (BOL), and at end of life (EOL), our RTG power is only 217 W. For these three reasons (eclipse, DSN, power), two different operational modes are necessary.

Operational Mode 1 is implemented during the BOL of the mission. Within this mode, there are two science modes. Since this is at BOL, we have enough RTG power to use all instruments at the same time. Science Mode 1 defines the use of all six science payloads aboard TES (see Section 2.2), while also simultaneously transmitting back to Earth. Science Mode 1 is broken into two sections; the first section lasts 2.67 hours and is when TES collects and transmits data at the same time. The second section lasts for 1.16 hours and only data is collected. Adding up both these times leads to a total time of 3.83 hours for Science Mode 1, which is the illumination period of the orbit. Science Mode 2 is implemented during the orbit's eclipse period of 1.01 hours. In this mode, only the dust analyzer, magnetometer, and radio science surveyor will be used. During the eclipse, the visible light camera, infrared spectrometer, and ultraviolet spectrometer will not be as useful, so to save power, they will not be used. During the 1.01 hour eclipse period, 0.07 hours will be dedicated to engineering data to check the status of the spacecraft. The rest will be used for data collection. Figure 1.4-1 shows a visual representation of our orbit's illumination and eclipse periods. Figure 1.4-2 shows Operational Mode 1, with Science Mode 1, Science Mode 2, transmission, and engineering data times shown.

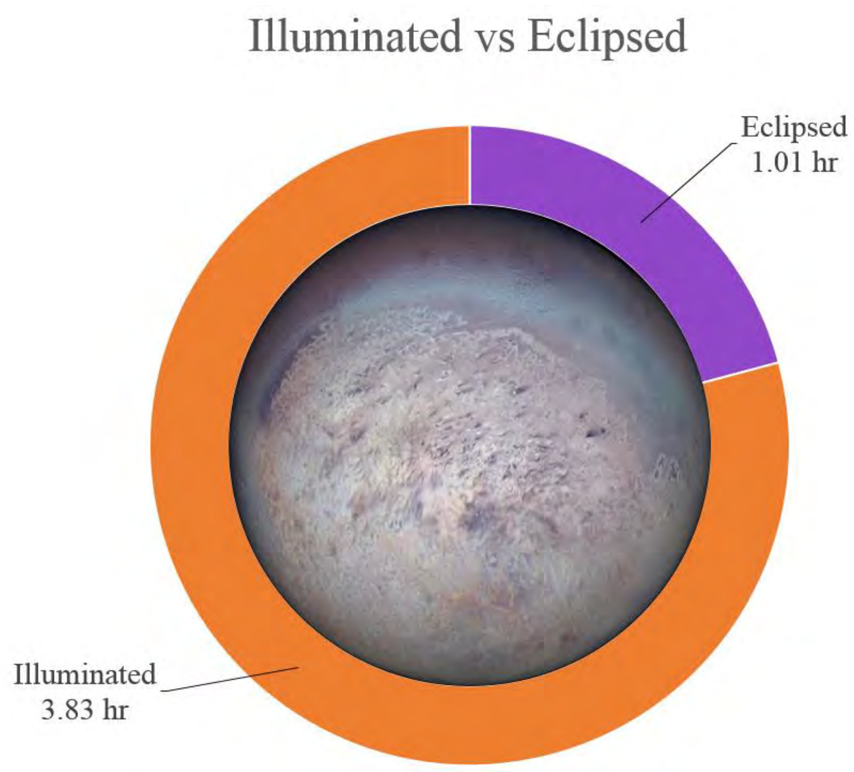


Figure 1.4-1 *Illuminated vs Eclipse Orbital Periods*

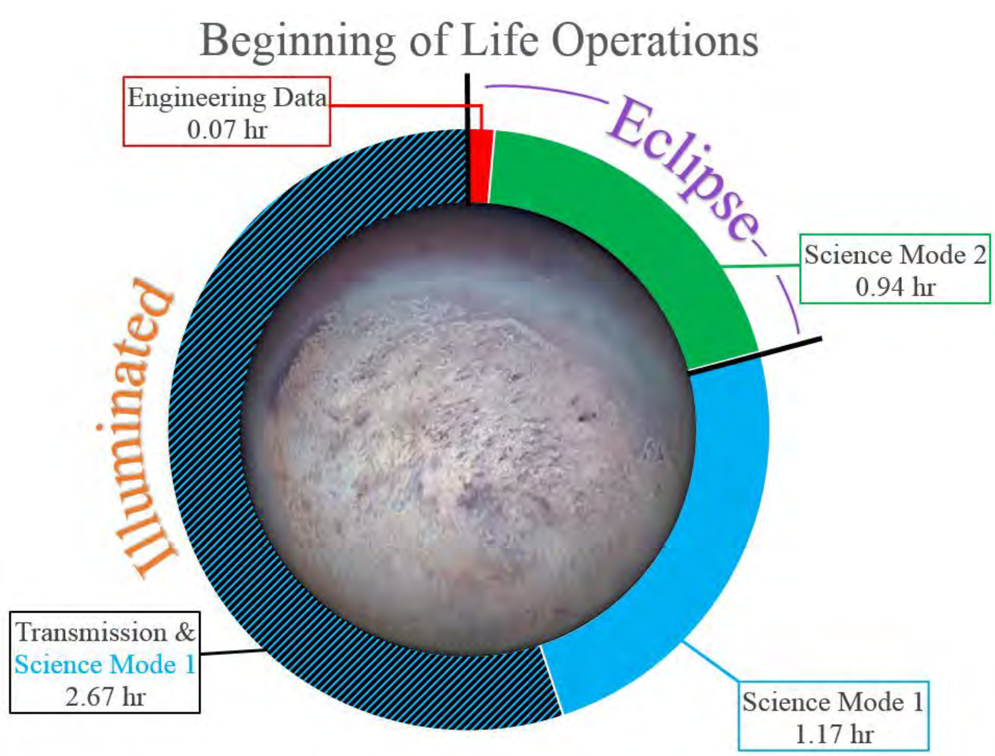


Figure 1.4-2 *BOL Operations showing Science Modes 1 & 2 per Orbit*

Near EOL, power must be allocated more carefully because of our RTG degradation. During this part of the mission, TES will enter Operational Mode 2. There are also two science modes in this operational mode: Science Mode 1E and Science Mode 2E. Due to less power being available, transmitting data and collecting data must be done separately. This time, during the illumination period of TES' orbit, it will spend 1.61 hours just transmitting data to Earth. For the next 2.22 hours, TES will enter Science Mode 1E and operate all of its payload items at once. The following eclipse period is the same at EOL as it was for BOL. TES will spend 0.94 hours in Science Mode 2E, using only its dust analyzer, magnetometer, and radio science surveyor, and 0.07 hours doing engineering data. Figure 1.4-3 shows Operational Mode 2 at EOL.

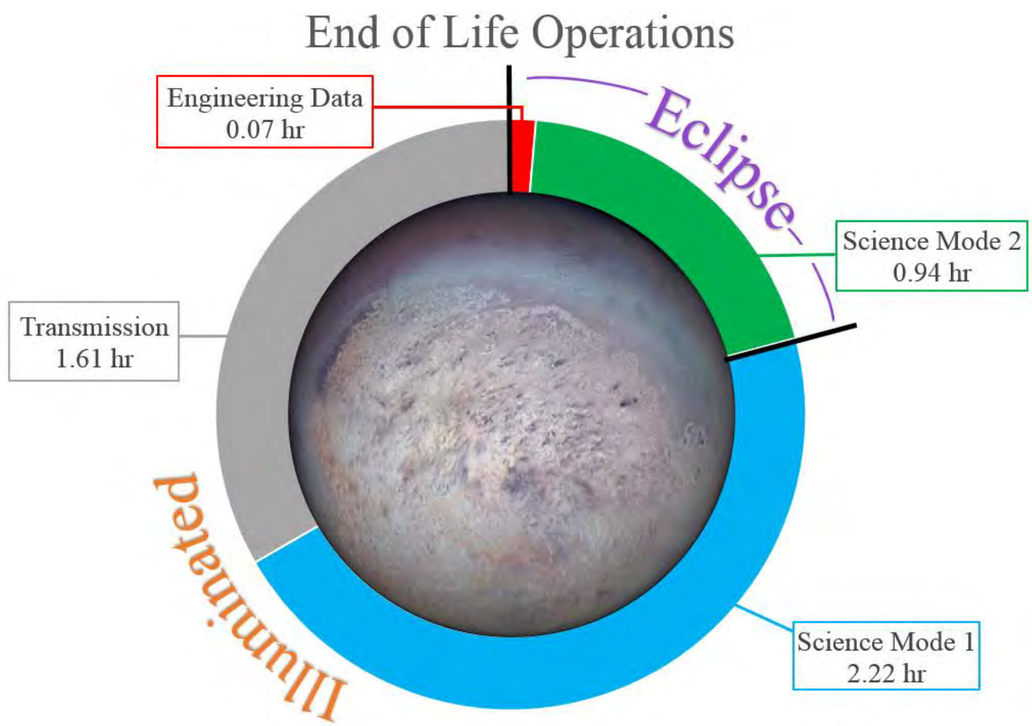


Figure 1.4-3 *EOL Operations showing Science Modes 1 & 2 per Orbit*

During descent, Vespucci will operate its dust analyzer to receive additional atmospheric particle data and its visible light camera to watch its descent. Once Vespucci is on the surface of Triton, it will begin full operation of all of its payload items, including its cameras and spectrometers. Vespucci is designed to operate for 12 hours at maximum power, which relates to at least two orbits made by TES. Vespucci will collect and transmit data to TES simultaneously.

1.5 Science Overview

As previously mentioned, in 1989, Voyager 2 detected eruption plumes from Neptune's moon Triton during its flyby. Due to the duration of the flyby, no further data was collected about these active geysers. Given that Neptune is the furthest planet from our sun, it places Triton in one of the coldest places in our solar system. Discovering an active geyser on one of the satellites of Neptune is considered very unusual, due to the cold temperatures at the outskirts of the solar system. The geyser activity that was observed only occurs at high temperatures, making this a significant topic of interest. This suggests that Triton has some thermal activity beneath the surface, creating some internal heating. Pulsar Enterprises is proposing Project TES to further investigate this unusual, dynamic icy moon and provide more scientific data to our planetary scientists to learn more about the edge of our solar system and the origins of those celestial bodies.

With the everlasting prospect of discovering life outside of our own planet, the initial findings of Triton's atmospheric composition and geothermal activity are a promising candidate in finding organic material. According to Irwin's *Assessing the Plausibility of Life on Other Worlds*, Triton is classified as a Category III body, which is defined as a "world where conditions are physically extreme, but possibly capable of supporting exotic forms of life unknown on Earth."[7]

Another topic of interest within the astronomical community is discovering the origins of our solar system. Due to the nature of Triton's retrograde orbit, it is possible that Triton was a separate body originating from outside of our solar system that was captured by Neptune's sphere of influence. Both Pluto and Triton originated as prograde satellites of Neptune, only to experience a catastrophic gravitational interaction.[8] Exploring the mechanics of Triton's geysers, as well as its atmospheric and surface composition could potentially lead to answers about the origin of our solar system. The scientific evidence obtained from Project TES would help develop a more accurate theoretical model for the early life of our solar system.

To investigate the details of the geyser phenomenon on Triton, the following science studies will be conducted: surface composition study, driving material of the exhaust, composition of the atmosphere, and geological data of Triton. Investigating these will enable scientists to understand more about Triton and what is causing the eruption plumes. Project TES will surely demonstrate the feasibility of deep space science observations.

1.6 Science Objectives

The primary observational target for this mission are the geysers on Triton's surface in the region near its south pole. Additional targets include the atmosphere of Triton and its surface.

For the active geysers on the surface, scientific measurements must be taken of both the gaseous exhaust and the solid material that is expelled from the geyser to gain a clear understanding them. Pressure and volume density measurements of the gaseous exhaust must be taken, as well as composition and particle size of the solid material in the plume. These measurements will be taken remotely using TES, and directly with Vespucci as the geyser precipitate falls upon its dust analyzer.

Triton's entire surface must be fully mapped, with emphasis on its geyser region. This geological mapping should include enough data to understand surface composition, surface history, mineral history, and thermal characteristics. Some of the surface composition analysis should be performed in a clean area not affected by the geyser zone or any nearby geyser plumes.

Regarding Triton's atmosphere, measurements must be taken for the composition, pressure, temperature, and density. These measurements will be taken from above the thermopause.

1.7 Primary Requirements

The project's primary requirements are interpreted directly from the given RFPs from JPL and AIAA. Table 1.7-1 is a summary of the both RFPs' primary requirements, showing their respective numbers and their condition. These top-level requirements form the backbone of our mission planning and spacecraft architecture design.

Table 1.7-1 *Primary Requirement Breakdown*

RFP	Req. #	Requirement Description	Required/Optional
JPL	T0.1	Provide data for mineral, surface history, and thermal mapping of Triton's geyser zone with a resolution of 10 m	Required
JPL	T0.1.1	Full surface mapping at 1 m resolution	Optional
JPL	T0.2	Determine composition of the geyser zone surface in a clean area not covered by geyser precipitation	Required
JPL	T1.1	System shall be capable of differentiating between areas of the geyser zone that are covered or not covered by geyser precipitation	Required
JPL	T1.2	System shall be capable of analyzing surface soils	Required
JPL	T0.3	Determine the composition, particle size, and particle volume density of the solid material released by Triton's geysers	Required
JPL	T0.4	Determine the composition of the geyser-driving exhaust and its pressure and volume density in the eruption plume	Required

JPL	T0.5	Determine the composition, temperature, pressure, and density of Triton's atmosphere from 20 km above its thermopause to the surface at 100 m intervals in altitude	Required
JPL	T0.5.1	Map atmosphere 100 km above the surface at 10 m intervals	Optional
JPL	T1.3	System shall have the capability to analyze the composition of a trace atmosphere	Required
JPL	T0.6	Atmospheric measurements shall be made 75 km upwind of any geysers in the region or 40 km crosswind and outside the periphery of the geyser region	Required
JPL	T1.4	System shall be capable of orienting its instrumentation to avoid interference by geyser plumes	Required
JPL	T0.7	Arrive at Triton by December 2035, completion of operations before December 2039 and data delivery by December 2040	Required
JPL	T0.8	All scientific data shall be delivered to the Planetary Data System	Required
JPL	T1.5	System must be capable of interfacing with the Deep Space Network	Required
JPL	T1.6	Spacecraft shall be capable of using its instruments in deep interplanetary space	Required
JPL	T0.9	System must be capable of performing science operations prior to arrival at Triton	Required
AIAA	T0.10	System must be capable of launching aboard a currently available launch vehicle including SLS	Required
JPL	T0.11	System shall abide by NASA's planetary protection protocols	Required
JPL	C0.1	Project cost cap is $5 billion	Required
JPL	M0.1	Mission Concept Review is due the last half of March 2017	Required
AIAA	M0.2	Preliminary Design Review is due the last half of May 2017	Required

1.8 Derived Requirements

Based on the primary requirements outlined above, derived requirements have been developed that specify how the primary requirements shall be applied to our mission. Table 1.8-1 lists all derived requirements that drive the design of each of our spacecraft's subsystems. Although each subsystem has its own derived requirements, a full requirements breakdown of every subsystem is beyond the scope of this report, so only top level derived requirements that influence multiple subsystems have been included below in Table 1.8-1.

Table 1.8-1 *Derived Requirements Breakdown*

RFP	Req. #	Requirement Description	Required/Optional
JPL	T1.1	System shall be capable of differentiating between areas of the geyser zone that are covered or not covered by geyser precipitation	Required
JPL	T1.2	System shall be capable of analyzing surface soils	Required
JPL	T1.3	System shall have the capability to analyze the composition of a trace atmosphere	Required
JPL	T1.4	System shall be capable of orienting its instrumentation to avoid interference by geyser plumes	Required
JPL	T1.5	System must be capable of interfacing with the Deep Space Network	Required
JPL	T1.6	Spacecraft shall be capable of using its instruments in deep interplanetary space	Required

2.0 Scientific Payload

2.1 Payload Overview

Our payload suite consists of ten science instruments. The payload items tie directly to our JPL RFP, and to complete our science objectives, the correct instrumentation for each requirement had to be selected. When determining which instruments to use, three main design drivers were specified: mass, power, and ground resolution. Furthermore, each requirement listed in the RFP was decomposed into basic objectives and categorized with an accompanying type of instrument.

Our analyses identified that there were six specific types of payloads that were essential to the completion of the requirements listed in the RFP. These included visible light cameras, infrared spectrometers, ultraviolet spectrometers, visible and infrared mapping spectrometers, particle and dust analyzers, and radio science and magnetometer experiments. As we conducted further trade studies, however, it became clear that to minimize mass and power usage and to increase the simplicity and efficiency of our system, our goals could be accomplished by merging two redundant instruments together by using only five of the six different types of payload. In this case, we decided to omit the infrared spectrometer and instead rely on our visible and infrared mapping spectrometers to cover this measurement.

All instruments will be produced specifically for this mission, with their specifications catered towards efficiency and quality. To correctly estimate the mass, power, resolution, data rate, and temperature limits of each of our instruments, the following trade study was conducted. First, the equivalent instruments from previous space missions were compiled based on year, and their mass and power properties were listed.[9-28] The properties of each instrument were then plotted on a line graph versus time to develop a rough trend of how the technology changes over time (very large outliers were omitted). From here, a more detailed trend line was applied to the data points and projected forward to our expected production year of 2022. Following the trend line, we obtained a value for the mass, power, resolution, and data rates for our new instruments. The mass, power, and resolution trend lines were then overlaid onto each other and onto one plot to determine the point at which they intersect. This intersection point helped verify that all the new values for our instruments agreed with each other so that no one value was inflated or exaggerated. It also confirmed that the trend lines were accurate and pointed to the same conclusion. In addition, using this method, we can classify all our payload items as a Technology Readiness Level of 9, based on NASA's definitions, since we are just improving their efficiencies from over time.[29]

The instruments we are using are distinguished between our TES Orbiter and our Vespucci Lander payloads. TES consists of six instruments: a visible light camera, an infrared spectrometer, an ultraviolet spectrometer, a dust analyzer, a radio science instrument, and a magnetometer. Each instrument satisfies a particular requirement in our compliance matrix, while also providing some useful additional data not specifically requested in the RFP. Vespucci consists of four instruments: a visible light camera identical to the one on the orbiter, a dust analyzer, a ground penetrating radar, and a multi-purpose infrared spectrometer capable of looking at Triton's surface up-close. With TES and Vespucci combined, there is a total of ten scientific instruments. Each instrument is listed below and further elaborated upon.

2.2 TES Orbiter Payloads

2.2.1 Visible Light Camera, "HRCC"

Our main instrument for taking pictures of the surface of Triton is our visible light camera placed on TES, abbreviated HRCC, for High-Resolution Color Camera. It is a panchromatic camera with a resolution of 2048 x 2048 pixels, a mass of 2.5 kg, and a power usage of 4 W. This camera will give us the clearest pictures of Triton ever taken. It will satisfy requirements T0.1 and T0.2 as listed in our compliance matrix. This camera will take pictures at an altitude of 870 km above the surface of Triton and is the main camera that will be used for mapping the surface, taking pictures, and identifying potential landing sites for our lander.

2.2.2 Visible and Infrared Mapping Spectrometer, "VIS"

There will be one visible and infrared mapping spectrometer, VIS, on TES. Its main purpose is to analyze the atmosphere and geysers of Triton and determine their composition, density, temperature, and in the case of the geysers, the driving force and exhaust material. It is also a visible mapping spectrometer to analyze the composition of the surface materials. It will operate within the spectral ranges of 300-2900 nm and 5800-50000 nm, which is in the visible range of the spectrum as well as the mid/far range of the infrared spectrum. This instrument will specifically satisfy requirements T0.1, T0.2, T1.1, T1.2, T0.3, T0.5, T1.3, and T0.6 in our compliance matrix. VIS has a mass of 8 kg and a power usage of 15 W.

2.2.3 Ultraviolet Spectrometer, "UVS"

The TES also contains one ultraviolet spectrometer, which will be used to conduct further, detailed analyses of Triton's trace atmosphere. More specifically, it will be used to understand the density, temperature, energy, and composition of the particles in the atmosphere. It will also be used to study the composition of the surface and geyser plume material. It operates in the spectral range of 50-195 nm, with a mass of 5.5 kg and a power usage of 6 W. Our UVS satisfies requirements T0.4, T0.5, T0.6, T1.3, and T1.4 as listed in our compliance matrix.

2.2.4 Dust Analyzer, "TDA"

There will be two distinct direct sensors in our system. Both are particle analyzers, but take measurements differently, as one is placed on TES and one is on Vespucci. This first dust analyzer is called the TDA, or Triton Dust Analyzer, and is placed on TES. It has a mass of 1 kg and a power usage of 3 W. As TES orbits through the trace atmosphere of Triton, some of the particles will be caught in this instrument. TDA will then analyze the energy, mass, and speed of the particles caught in its sensors. It is our only direct sensor on our orbiter and will help us better understand the properties of Triton's atmosphere. TDA satisfies requirements T0.5 and T1.3 in the compliance matrix.

2.2.5 Radio Science, "RAT"

The TES Orbiter also has a radio science instrument onboard, called RAT, or Radio Atmospheric Testing. It will study the atmosphere of Triton with radio waves. This will give us additional data on the profile of the atmosphere, which can then be correlated with the data from the ultraviolet and infrared spectrometers to increase the accuracy of our findings. RAT has a mass of 4 kg and a peak power usage of 7.5 W, and will help satisfy requirements T0.5, T0.6, T1.3, and T1.4.

2.2.6 Magnetometer, "MAGIC"

The final scientific instrument aboard our orbiter is a magnetometer, named MAGIC, standing for Magnetic Investigator and Characterizer. MAGIC will allow us to study the anomalous magnetic field of Triton and potentially the magnetic field of Neptune as well. Although this objective is not explicitly stated in the RFP, it would be interesting to learn more about Triton. MAGIC has a low mass of just 3 kg and low power usage of 3 W, thus, the tradeoff for additional useful data is not severe at all. The only consideration when choosing to include this payload item was the fact that a potentially heavy and long boom would be necessary. The boom for the magnetometer has a mass of 2 kg.

2.3 Vespucci Lander Payloads

2.3.1 Visible Light Camera, "HRCC"

This is the same camera that is also placed on the TES orbiter. When placed on Vespucci, HRCC will help us fulfill requirement T0.1, as it will give us more close-up, high-resolution pictures of the surface. After landing, Vespucci will first take a 360° panorama picture of Triton's surface.

2.3.2 Dust Analyzer, "LDA"

The secondary particle analyzer on Vespucci will study the particles and precipitate from the geyser plumes. LDA, or Local Dust Analyzer, can measure the mass, speed, and energy of the particles that fall on it from the exhaust material. It has a mass of 1.5 kg and a power usage of 4 W. This is the sole direct sensor on our Vespucci Lander and will help us further understand the composition of the material jettisoning from the geysers on the surface of Triton. LDA satisfies requirements T0.2 and T0.4.

2.3.3 Subsurface Surveyor, "Triple-S"

Also on Vespucci is a ground penetrating radar, called Triple-S, for Subsurface Surveyor. It will use radio waves to study the surface and subsurface of Triton. More specifically, it will help us better understand the composition of the surface soil, the soil underneath, and any potential geyser precipitate resting on the surface. It has a mass of 3 kg and a power usage of 5 W. The instrument satisfies requirement T0.1, T0.2, T0.3, T1.1, and T1.2.

2.3.4 Surface Infrared Spectrometer, "LIS"

The final instrument on our Vespucci Lander is another infrared spectrometer, called LIS, standing for Lander Infrared Spectrometer. It operates in the spectral ranges of 5000-29000 nm. This instrument will study surface dust and plume precipitate up-close, and will provide very accurate data because of how close it is to the sample being studied. Much of our data from Vespucci will come from this instrument. LIS has a mass of 2 kg and a power usage of 5 W. It will satisfy requirements T0.1, T0.2, T0.3, T1.1, and T1.2 in the compliance matrix.

2.4 Payload Summary

The payloads for each spacecraft are tabulated below, showing mass, power, and various other properties of each instrument. The totals are shown at the bottom of the table. Table 2.4-1 below shows the TES Orbiter instruments, and Table 2.4-2 shows the instruments for our Vespucci Lander.

Table 2.4-1 *TES Payload (Orbiter)*

Instrument	Mass [kg]	Peak Power [W]	Ground Resolution [m/pixel]	Resolution [pixel x pixel]	FOV [mrad x mrad]	Max Data Date [kbps]	Design Life [years]	TRL
HRCC	2.5	4	7	2048 x 2048	16.5	10066.33	25	9
UVS	5.5	6	8	1024 x 128	9.4	6.43	25	9
VIS	8	15	15	256 x 256	4.4	0.85	25	9
TDA	1	3	-	-	-	2.22	25	9
RAT	4	7.5	-	-	3141.59	0.80	25	9
MAGIC	3	3	-	-	Point Source	4.05	25	9
Total	**24**	**38.5**						

Table 2.4-2 *Vespucci Payload (Lander)*

Instrument	Mass [kg]	Peak Power [W]	Ground Resolution [m/pixel]	Resolution [pixel x pixel]	FOV [mrad x mrad]	Max Data Date [kbps]	Design Life [years]	TRL
HRCC	2.5	4	7	2048 x 2048	16.5	10066.33	25	9
Triple-S	3	5	-	-	3141.59	1.08	20	9
LIS	2	5	$5x10^{-6}$ (0.2 m above ground)	256 x 256	0.366	0.85	20	9
LDA	1.5	4	-	-	-	0.08	20	9
Total	**9**	**18**						

2.5 Scientific Payload Operations Scenario

As mentioned above in Section 1.4, there are two operational modes for TES, with two science modes each. In September of 2035, upon Triton arrival, main science operations will begin. However, prior to arrival at Neptune during cruise, basic science operations will still be done. For example, there are opportunities to begin payload operations during the Saturn flyby, Neptune capture, or asteroid belt flyby. During our orbit's illumination period of 3.83 hours, the HRCC will continuously take photos of the surface of Triton (while it is exposed to sunlight), while the VIS and UVS will operate in tandem in order to analyze the composition, density, and pressure of the atmosphere, and also study the geyser plume material. The imaging instruments will also be responsible for providing potential landing spots for the Vespucci Lander. However, during eclipse periods, which last approximately 1.01 hours per orbit, the HRCC, VIS, and UVS shall be made idle. Here further studies will be conducted by TDA, RAT, and MAGIC. While it is an analysis of a trace atmosphere at an 870-km altitude, the TDA shall analyze the properties of any particles its sensors can catch; meanwhile, MAGIC shall continuously study magnetic fields and RAT shall collect atmospheric data.

The Vespucci Lander shall provide another set of data about Triton. Each of the instruments will contribute in their own ways to form a detailed model of Triton's surface. For example, LDA will give a more accurate reading of the composition of the materials projected from the geysers. Meanwhile, Triple-S will analyze the soil materials and geyser components that rest on and beneath the surface. Given the low power requirements of the four instruments on the lander, each can remain fully operational for as long as there is power available to run the lander (approximately 12 hours). Vespucci will continuously take data. In terms of transmitting data to the Planetary Data System (PDS), there will be about a 1.51 hour window when TES and Vespucci will be in direct contact with each other, allowing the lander's data to be transmitted to the orbiter, and eventually relayed back to the PDS.

The data collected from the instruments of TES shall continuously be transmitted back to the PDS from the time the instruments are activated until they are decommissioned in December 2040 – totaling up to just over 5 years of analysis of Triton. However, each instrument shall cease its operation one year before decommission in December 2039, at which time the disposal process shall commence.

2.6 Science Traceability Matrix

The payload instruments to be used in the spacecraft will be carefully chosen to fulfill each of the science requirements given in the RFP. Table 2.6-1 gives a layout of each science objective, as well as the corresponding goals, instrument(s) and purpose of said instrument(s) in relation to how it achieves the objectives.

Table 2.6-1 *Science Traceability Matrix for Objectives and Instruments*

Science Objective	Required Ref. #	Goals	Instrument
Geyser zone surface shall be fully mapped to a ground resolution of 10 meters	T0.1 T0.1.1	Fully mapped to a ground resolution of 1 meter	HRCC VIS Triple-S LIS
Composition of geyser zone not covered in precipitate shall be studied	T0.2 T1.1 T1.2	-	HRCC VIS LDA Triple-S LIS
Measurements shall be taken for composition, particle size, and particle volume density of geyser solid material	T0.3	-	VIS Triple-S LIS
Composition of geyser exhaust and pressure and volume density shall be taken	T0.4	-	UVS LDA
Composition, temperature, pressure, and density of atmosphere shall be taken at 20 km above thermopause at 100 m intervals; 75 km upwind and 40 km crosswind of geysers	T0.5 T0.5.1 T1.3 T0.6	100 km altitude at 10 m intervals	VIS UVS TDA RAT

3.0 Mission Overview and Implementation

3.1 Mass Budget

Initial mass estimates were performed using Brown's estimation criteria.[30] As our design process progressed, each subsystem was further refined to get more accurate numbers. Finally, the exact mass values for each subsystem were determined and compiled in Table 3.1-1 below. The mass allocation percentages for each subsystemare shown in Table 3.1-2.

Table 3.1-1 *Summarized Spacecraft Mass Statement*

Subsystem	Mass (kg)
Power	201.74
Structures	186.00
Thermal Control	20.15
Propulsion	159.98
Command and Data Handling (C&DH)	7.52
Attitude Control System (ACS)	82.49
Telecommunications	70.00
Cabling	11.21
Payload	24.00
Subsystem Total	**763.09**
Lander Mass	210.40
On-Orbit Dry	**973.49**
Liquid Bi-Propellant	1825.38
Pressurant	8
On-Orbit Wet	**2806.54**
Adapter	261.89
Launch Mass	**3068.43**
Launch Mass Margin	306.57
Launch Mass Capability	**3375.00**

Table 3.1-2 *Spacecraft Subsystem Mass Percentage Allocation*

Subsystem	Allocation (%)	Mass (kg)
Power	6.57%	201.74
Structures	6.06%	186.00
Thermal Control	0.66%	20.15
Propulsion	5.21%	159.98
Command and Data Handling (C&DH)	0.25%	7.52
Attitude Control (ACS)	2.69%	82.49
Telecommunications	2.28%	70.00
Cabling	0.37%	11.21
Payload	0.78%	24.00
Lander	6.86%	210.40
Liquid Bi-Propellant	59.49%	1810.74
Pressurant	0.25%	7.67
Payload Adapter	8.54%	260.05
Total	**100.00%**	**3068.43**

3.2 Power Budget

Similar to the mass budget, initial power estimates were performed using Brown's estimation criteria. [30] After many iterations, final power values were determined for each subsystem. The final power statement is shown in Table 3.2-1, and the allocation percentages for each subsystem is in Table 3.2-2.

Table 3.2-1 *Spacecraft Power Statement*

Subsystem	Power (W)
Power	45.00
Structures	0.00
Thermal Control	1.50
Propulsion	76.58
Command and Data Handling (C&DH)	27.50
Attitude Control System (ACS)	108.24
Telecommunications	62.00
Cabling	14.00
Total	344.52
Payload	38.50
Total On-Orbit Dry	373.32
BOL Margin	**66.68**
EOL Margin	**-156.32**
MMRTG Power BOL	**440**
MMRTG Power EOL	**217**

Table 3.2-2 *Spacecraft Subsystem Power Percentage Allocation*

Subsystem	Allocation (%)	Power (W)
Power	12.05%	45.00
Structures	0.00%	0.00
Thermal Control	0.40%	1.50
Propulsion	20.51%	76.58
Command and Data Handling (C&DH)	7.37%	27.50
Attitude Control System (ACS)	28.99%	108.24
Telecom	16.61%	62.00
Cabling	3.75%	14.00
Payload	10.31%	38.50
Total	100.00%	373.32

3.3 Complete Mass and Power Statement

The complete, detailed mass and power statement is shown in Table 3.3-1. Each subsection is broken down and the corresponding mass and power for each component is shown.

Table 3.3-1 *Complete System Mass and Power Statement*

Components	Mass (kg)	Power (W)
Power Subsystem		
Batteries	0	0.00
(4) MMRTGs with Attachment Hardware	174.4	0.00
Power Regulatory Unit, Power Distribution unit, Shunts	25.14	45.00
Structures Subsystem		
Bus Mass	365.00	0.00
Thermal Control Subsystem		
Heat Pipes	9.99	0.00
(2) Radiators	7.88	0.00
(3) Thermal Switches	0.06	1.00
(16) RHUs	0.64	0.00
Multi-Layers Insulation	1.49	0.00
(3) Temperature Sensors	0.09	0.50
Propulsion Subsystem		
Fill and Drain Valve	0.226	0.00
Fuel Tank	21.48	0.00
Oxidizer Tank	13.86	0.00
Pressurant Tank	36.54	0.00
Mounting Hardware	63.75	0.00
Pressure Transducer	0.68	6.00
Temperature Sensor	1.36	3.00
Open Pyro Valve	1.23	0.00
Closed Pyro Valve	1.08	0.00
Solenoid Valve	2.04	19.16
Latching Valve (No Relief)	1.02	3.00
Pressure Regulator w/Filter	2.08	0.00
Check Valve	1.09	0.00
Relief Valve	1.70	0.00
Flow Balance Orifice	0.46	0.00
System Filter	0.57	0.00
Main Thruster	5.40	45.00
Piping	12.68	0.00
Command and Data Subsystem		
C&DH Processor Card	0.15	5.00
Solid State Recorder (SSR) Card	7.10	17.00
Instrument Interface Card	0.07	2.50
Critical Command Decoder (CCD) on Uplink Card	0.10	2.50
Downlink Formatter on Downlink Card	0.10	0.50
Attitude Control Subsystem		
ACS Thrusters	3.96	27.28
Reaction Wheel	34.00	28.00
Magnetic Torque Rod	11.20	7.91
Gyroscopic Controls	13.50	25.00
Star Trackers	19.41	19.80
Sun Trackers	0.38	0.25
Telecommunications Subsystem		
Electra Radio and UHF Antenna	7.00	10.00
Telecom Panel	20.00	0.00
Antenna (High Gain and Low Gain)	30.00	0.00
SDST	10.00	2.00
(2) SSPA	3.00	50.00
Cabling Subsystem		
Electrical Cabling	11.21	14.00
Payload		
Visible Light Camera	2.50	4.00
UV Spectrometer	5.50	6.00
Visible/Infrared Mapping	8.00	15.00
Particle Energy Analyzer Spectrometer (PEAS)	1.00	3.00
Magnetometer	3.00	3.00
Radio & Plasma Wave Science	4.00	7.50
Lander		
Power Subsystem	22.00	23.00
Structures Subsystem	21.00	0.00
Thermal Control Subsystem	5.00	65.00
Propulsion Subsystem	33.00	2.00
Command and Data Subsystem	8.00	40.00
Attitude Control Subsystem	19.00	47.00
Telecomunications Subsystem	15.00	54.00
Cabling Subsystem	7.00	2.00
Payload: Visible Light Camera	2.50	4.00
Payload: Subsurface Surveyor	3.00	5.00
Payload: Surface Spectrometer	2.00	5.00
Payload: Dust Analyzer	1.50	4.00
Monopropellant	70.90	0.00
Cold Gas Proppellant	0.50	0.00
Propellent		
Liquid Bi-Propellant	1825.38	0.00
Pressurant	7.67	0.00
Launch Adapter		
Adapter	261.89	0.00

3.4 Spacecraft Diagrams

As mentioned on the concept of operations, the spacecraft will be in four different configurations during its mission. The first configuration is during the launch phase of the spacecraft. The spacecraft will be stowed and mounted on the payload adapter. Figures 3.4-1 to 3.4-4 show the 3-view drawing along with the isometric view of the spacecraft with its fairing for the first two figures, and without it for the last two.

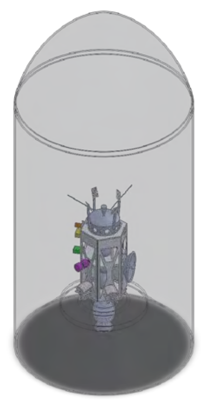


Figure 3.4-1 *Isometric View of Stowed TES in Fairing*

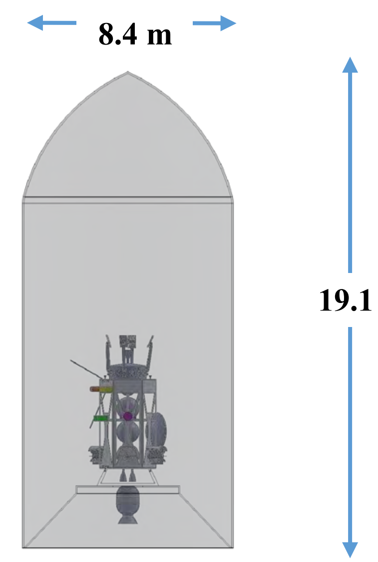


Figure 3.4-2 *Front View of Stowed TES in Fairing*

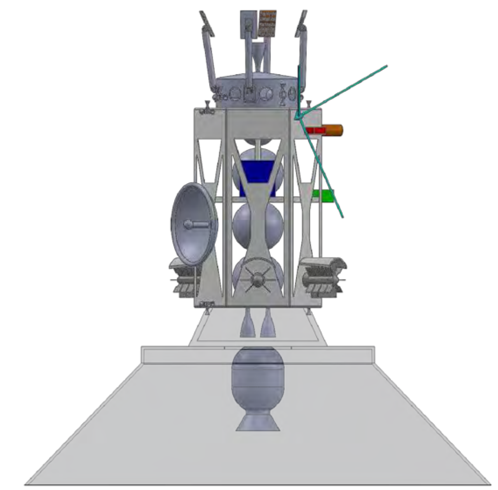


Figure 3.4-3 *Front View of Stowed TES on Payload Adapter*

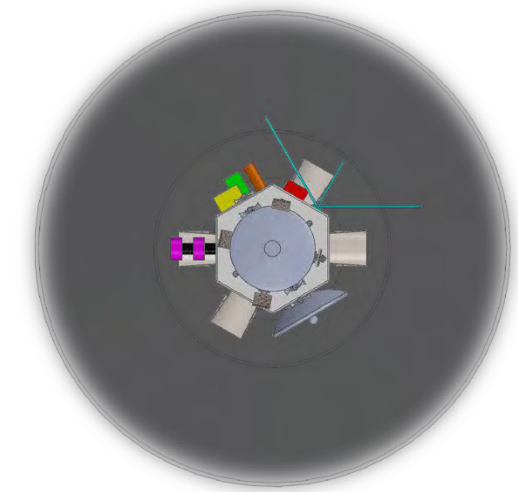


Figure 3.4-4 *Top View of Stowed TES*

Once the launch vehicle's fuel is depleted, the spacecraft will jettison off the launch vehicle, and the spacecraft is shown below with the upper stage in Figures 3.4-5 to 3.4-8.

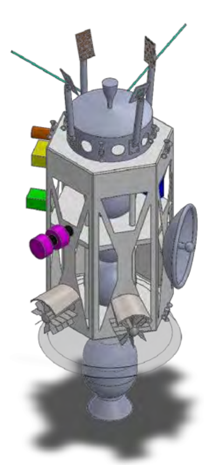


Figure 3.4-5 *Isometric View of TES Upper Stage Configuration*

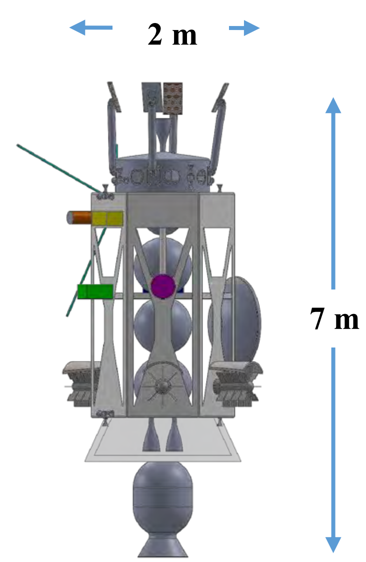


Figure 3.4-6 *Front View of TES Upper Stage Configuration*

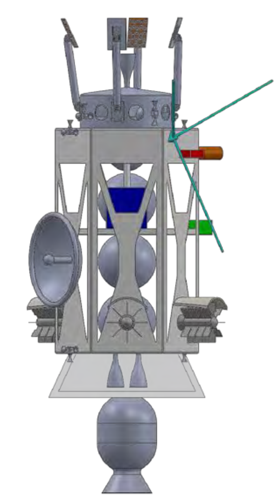


Figure 3.4-7 *Back View of TES Upper Stage Configuration*

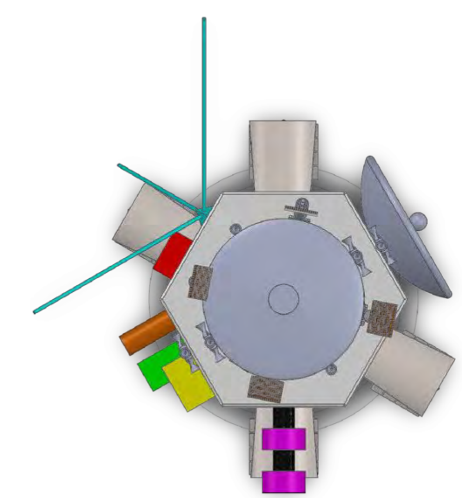


Figure 3.4-8 *Top View of TES Upper Stage Configuration*

After the solid motor upper stage completes its burn, the upper stage adapter is jettisoned off of the spacecraft and the stowed spacecraft in transit to Neptune is shown on Figure 3.4-9 to 3.4-12.

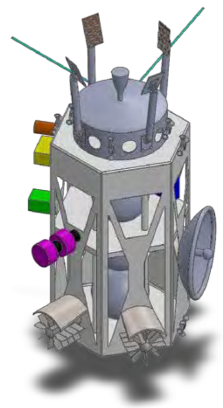


Figure 3.4-9 *Isometric View of TES Upper Stage Jettisoned*

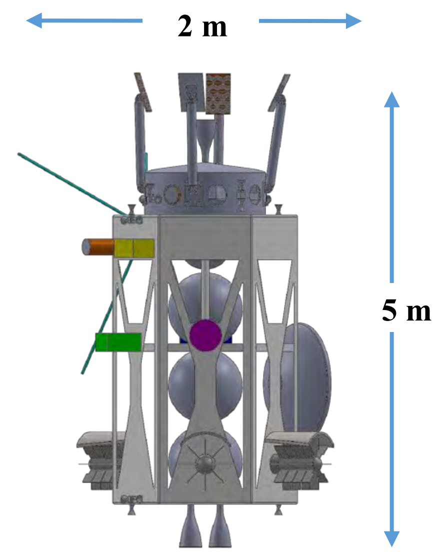


Figure 3.4-10 *Front View of TES Upper Stage Jettisoned*

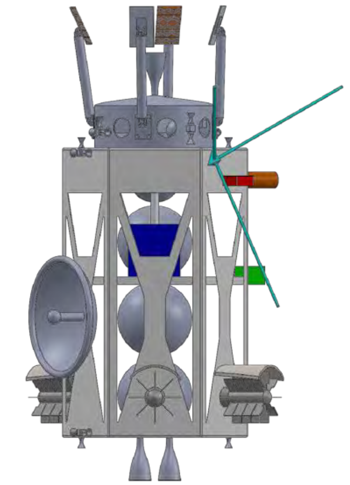


Figure 3.4-11 *Back View of TES Upper Stage Jettisoned*

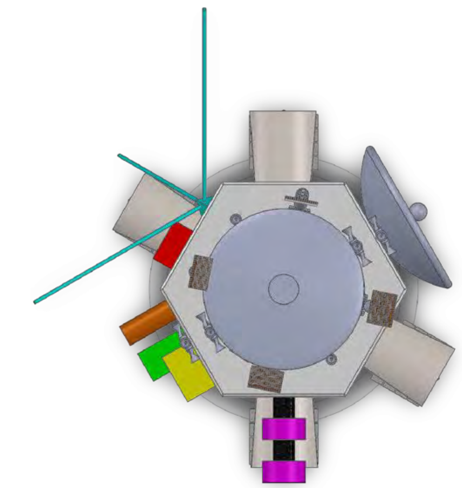


Figure 3.4-12 *Top View of TES Upper Stage Jettisoned*

When the spacecraft arrives at Triton, the lander is jettisoned off the spacecraft and the magnetometer boom is fully deployed shown on Figure 3.4-13 to 3.4-16.

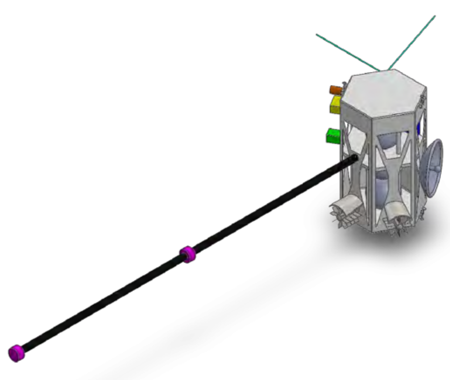


Figure 3.4-13 *Isometric View of TES Deployed*

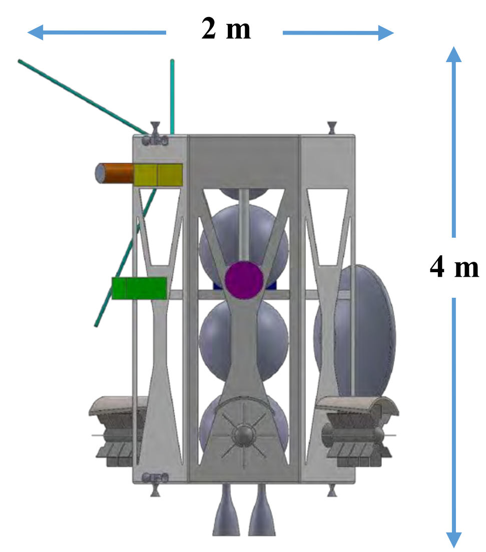


Figure 3.4-14 *Front View of TES Deployed*

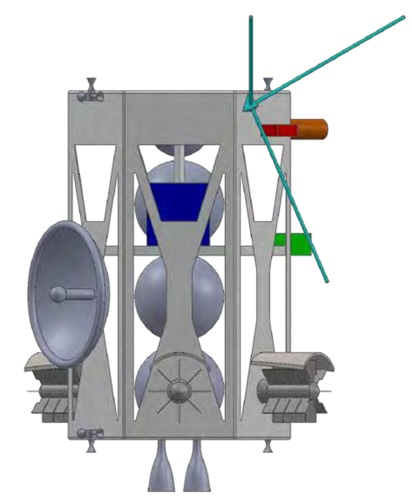


Figure 3.4-15 *Back View of TES Deployed*

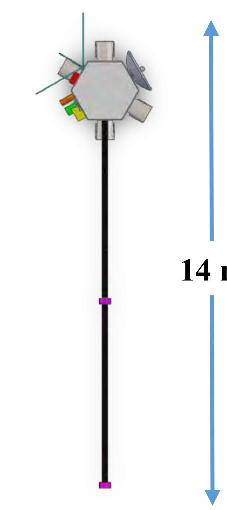


Figure 3.4-16 *Top View of TES Deployed*

3.5 Lander Diagrams

When Vespucci is released from TES, all systems are stowed. 3-view drawings are shown in Figures 3.5-1 to 3.5-3. The various instruments are highlighted.

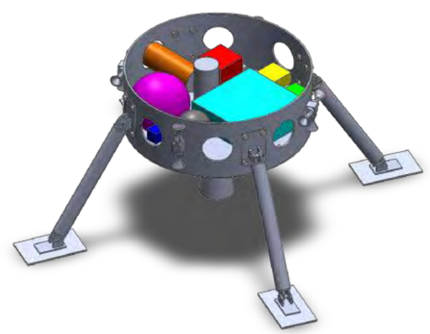


Figure 3.5-1 *Isometric View of the Vespucci Stowed Configuration*

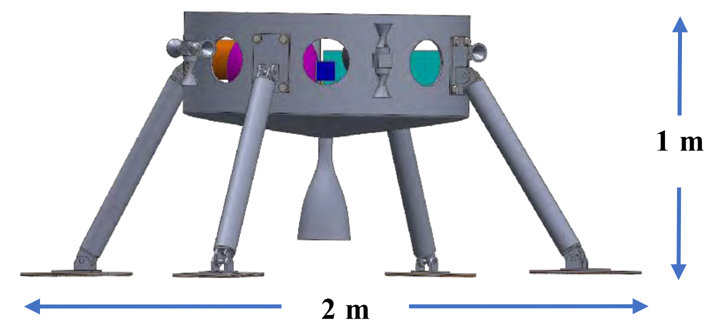


Figure 3.5-2 *Side View of the Vespucci Stowed Configuration*

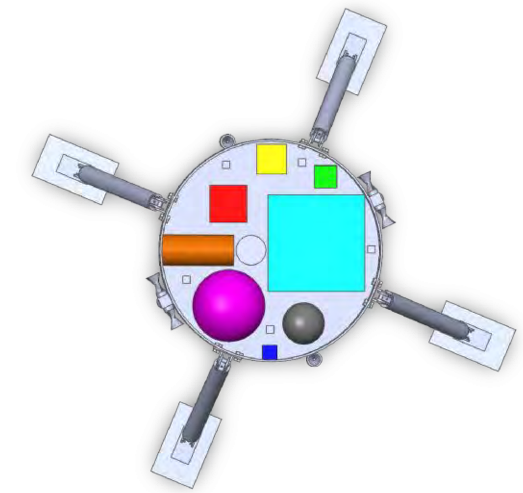


Figure 3.5-3 *Top View of the Vespucci Stowed Configuration*

Once Vespucci landed on the surface of Triton, HRCC will rise from the lander to take a 360 degree panorama picture. 3-view drawings are shown below on Figures 3.5-4 to 3.5-6.

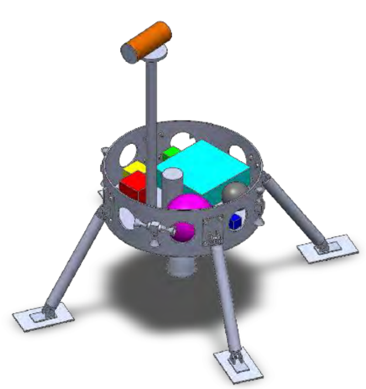


Figure 3.5-4 *Isometric View of the Vespucci Deployed*

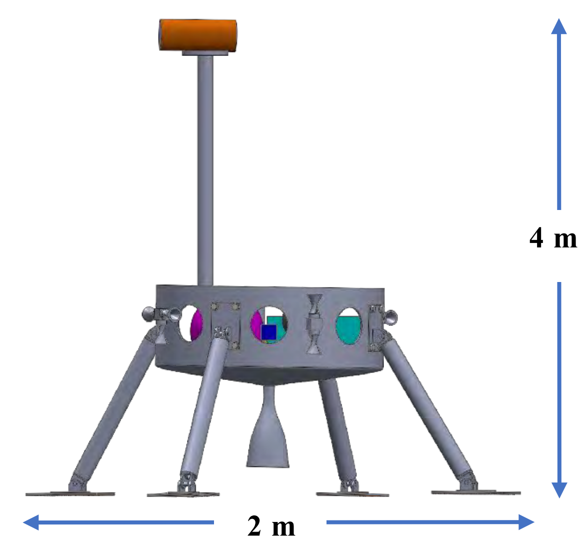


Figure 3.5-5 *Side View of the Vespucci Deployed*

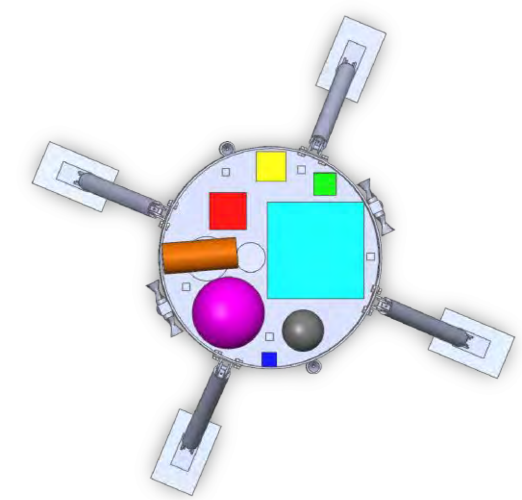


Figure 3.5-6 *Top View of the Vespucci Deployed*

4.0 Mission Trajectory

4.1 Launch Vehicle Selection

The launch vehicle issued by the AIAA RFP is the Space Launch System (SLS). For performance analysis purposes, a launch vehicle trade study was conducted shown below in Table 4.1-1. The launch vehicles in comparison are readily available or currently in development. The payload capacity to certain planets are shown. Also, the size of the payload fairing for each launch vehicle is included to portray the similarities between them. The cost per launch for the current launch vehicles are estimated based on previous missions, while the SLS is estimated similar to the Saturn V launch vehicle due to its size and launch capability.

Table 4.1-1 *Launch Vehicle Trade Study*

Launch Vehicle	LEO (kg)	GTO (kg)	Mars (kg)	Jupiter (kg)	Saturn (kg)	Fairing Diameter (m)	Fairing Length (m)	Cost ($M in 2016)
Falcon Heavy	54400	22200	13600	4080	0	5.2	13.1	135
SLS Block 1B	105000	50000	30000	9000	0	8.4	19.1	1269
Delta IV Heavy	28790	14220	8571	2571	0	5	19.1	435
Ariane V	21000	10050	4500	1800	0	5.4	18.1	220

The SLS variant that this mission will utilize is the SLS Block 1B. It features 2 solid rocket boosters, four RS-25 liquid propellant engines for the core stage, the Exploration Upper Stage (EUS) with four RL10 liquid propellant engines, and outputting at liftoff thrust of 8.8 million pounds.[31] The SLS Block 2, which is identical to the Block 1B except for advanced rocket boosters, an upper stage, and a larger payload fairing, is shown in an exploded view below in Figure 4.1-1. A figure of Block 1B was unavailable, so the next closest SLS was used.

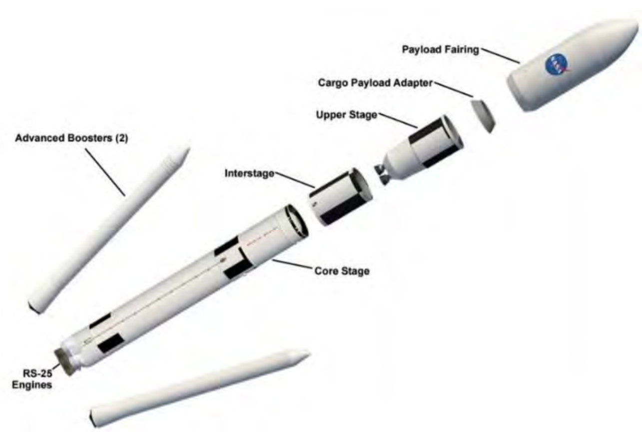


Figure 4.1-1 *Space Launch System Block 2 Exploded View*

The payload capacities at various C3 capabilities were obtained through the launch vehicle payload planner guides. None of the sources contained capabilities past Jupiter, therefore trends to Saturn had to be independently developed. Figure 4.1-2 is shown below to prove that the SLS is currently the most capable of launch vehicles for deep space interplanetary travel.

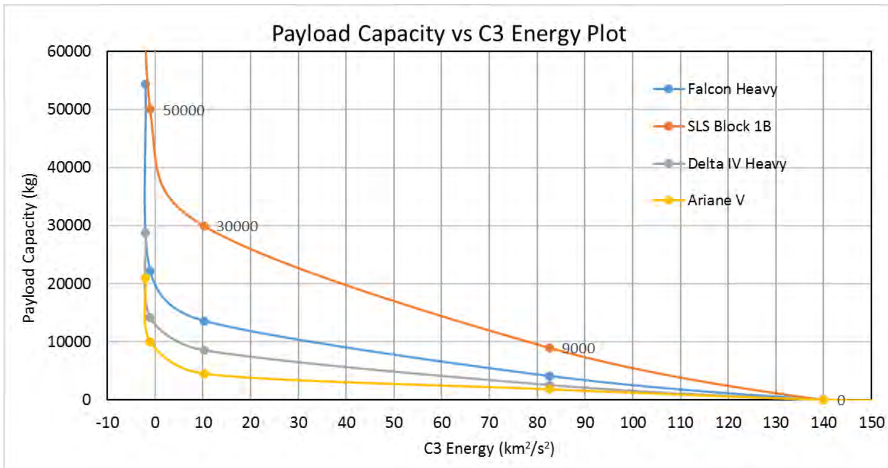


Figure 4.1-2 *Launch Vehicle Payload Capacity versus C3 Capability*

4.2 Upper Stage Selection & Integration

In Figure 4.1-2 shown above, all the launch vehicles do not reach past a C3 of 140 km^2/s^2. This is a big problem because the trajectory analysis done in Section 4.3 allocated a C3 of 143 km^2/s^2. Thus, it was decided to implement an additional upper stage or kick motor.

Referencing the SLS Payload Planner Guide[31], the payload fairing does not have any restrictions on additional stages inside the fairing bay. With that noted, upper stages were explored to increase the payload capacity for this mission. The dimensions, availability, cost, and C3 capability were taken into consideration for assessing the most ideal choice for this mission.

Liquid bipropellant upper stages such as Centaur[32], the DCSS[33], and the EUS[34] were originally considered, but all had a large excess of C3 capability, on top of a high cost. To save on cost and stay within a reasonable C3 capability increase, a kick motor was considered.

The Orbital ATK Star 37 XFP was chosen based on its availability and experience in prior space missions.[35] The Orbital ATK Star XFP addition increases the overall SLS C3 capability to 160 km^2/s^2 and increases the payload capability at a C3 of 143 km^2/s^2 to 3375 kg. This new payload capability will act as the launch mass capability to determine our launch margin.

To integrate the payload and upper stage into the SLS payload fairing, two different payload attach fittings were implemented. One of the fittings connected the SLS to the payload directly, while the other fitting attached the Thiokol to the payload. The two fittings overlap each other when connecting to the payload and the Star 37 resides inside the larger payload attach fitting. Figure 4.2-1 portrays the Thiokol integration into the payload configuration.

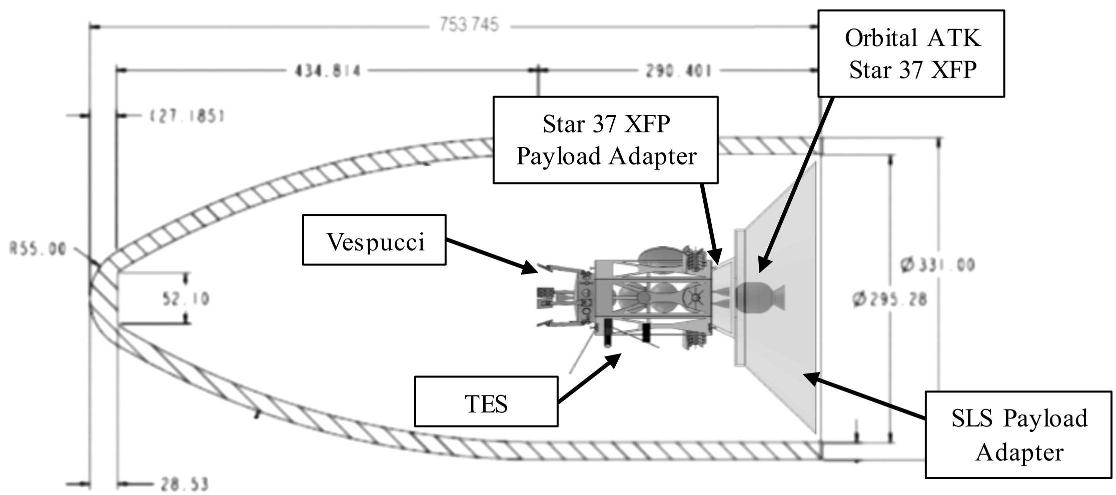


Figure 4.2-1 *Orbital ATK Star 37 XFP Upper Stage Kick Motor Integration*

4.3 Mission Trajectory Optimization

The importance of mission trajectory selection cannot be overstated. The trajectory that a spacecraft takes over its mission lifetime determines the spacecraft's fuel needs, time of flight, lifetime environmental conditions, and constraints of possible launch vehicles and windows that could accommodate such a mission. The most important constraint is the spacecraft mass, therefore, it is important to put a great amount of effort into trajectory selection and optimization. In our case, arrival date was also a prominent constraint.

It is required that this mission reach Triton by the end of 2035. Given that initial planning for this mission began in late 2016, which leaves 19 years for mission conception, development, production, launch, and flight. A typical low energy transfer trajectory for a spacecraft flying between the Earth and Triton would take just over 30 years to reach Triton. The 19-year time constraint requires a reduced production time and flight time. It would be expensive in terms of labor costs, but by shaving time off production, we can lengthen the time of flight, decrease the fuel costs for our mission, and increase the allowable spacecraft mass. Running under an accelerated production schedule of 5 years, the estimated time that launch would occur is 2022. That gives us a flight time of 13 years.

Neptune has a low synodic period with Earth, just over a year. However, with the shortened flight time, any feasible trajectory between the Earth and Neptune's sphere of influence will likely need to be assisted by one of the outer planets. Jupiter and Saturn have very large masses and good secondary scientific potential for flyby observations, but both have larger synodic periods than Neptune, which will decrease the number of flyby opportunities that will be possible. With these low number of flyby opportunities, missing our launch window will likely have a devastating effect on the mission's timetable, so alternative backup trajectories will also likely have to be selected.

It was important to analyze many different trajectory options, and have a tool that can generate plausible trajectories quickly. We opted to utilize a genetic algorithm. Genetic algorithms are a good way to quickly generate many different possible trajectories and down-select to the most optimum trajectory. However, there are several downsides to using a genetic algorithm for this purpose. They tend to be good at inducing artificial creativity by using a random number generator to create slight alterations to a plausible solution, but they usually do not converge on the same optimum solution every single time. Also, because a genetic algorithm depends on a random number generator to generate new plausible solutions they tend to not analyze the full range of possibilities. Therefore, it may take running a genetic algorithm many times to solve for a fully optimal solution.

The tool used was Trajectory Optimization Tool v2, developed by Adam Harden and published through GitHub.[36] This tool uses the J2000 ephemeris and a Lambert Solver to generate possible interplanetary trajectories. The tool allows the user to select an order of interplanetary flyby targets, time limits, and select weighting values for solution fitness calculation. Solutions are selected based on their C3, flyby ΔV, and sphere of influence relative arrival velocity. The output is a text file with launch, flyby, and arrival times, ΔV requirements, and orbital elements for each leg of the mission plan.

To find the true optimum solution, at least 150 runs were conducted. The algorithm was run with many different planetary flyby sequences to see what effects each flyby opportunity had on the overall trajectory performance. For each input flyby sequence, the algorithm was run multiple times to account for the inaccuracy of the genetic algorithm. Using this method, 43 unique plausible trajectories were generated. The next step was to further down select these partially optimized trajectory solutions. All solutions that required a launch before 2022 were immediately discounted from consideration. The rest of the solutions were then analyzed in terms of arrival date at Neptune's sphere of influence and the mass requirements that each solution would impose on the mission.

To find the mass requirements imposed on the mission, the launch capability of the SLS Block 1B, the required C3 of the solution, the required interplanetary ΔV, and the sphere of influence excess arrival velocity were all combined using Tsiolkovky's rocket equation to calculate the maximum spacecraft mass that could arrive at Triton. This was a measure of effectiveness that we referred to as "deliverable mass to Triton", and it was the main measure used to down-select to our final trajectory solution. In Figure 4.3-1 below, each trajectory solution is represented by a single data point.

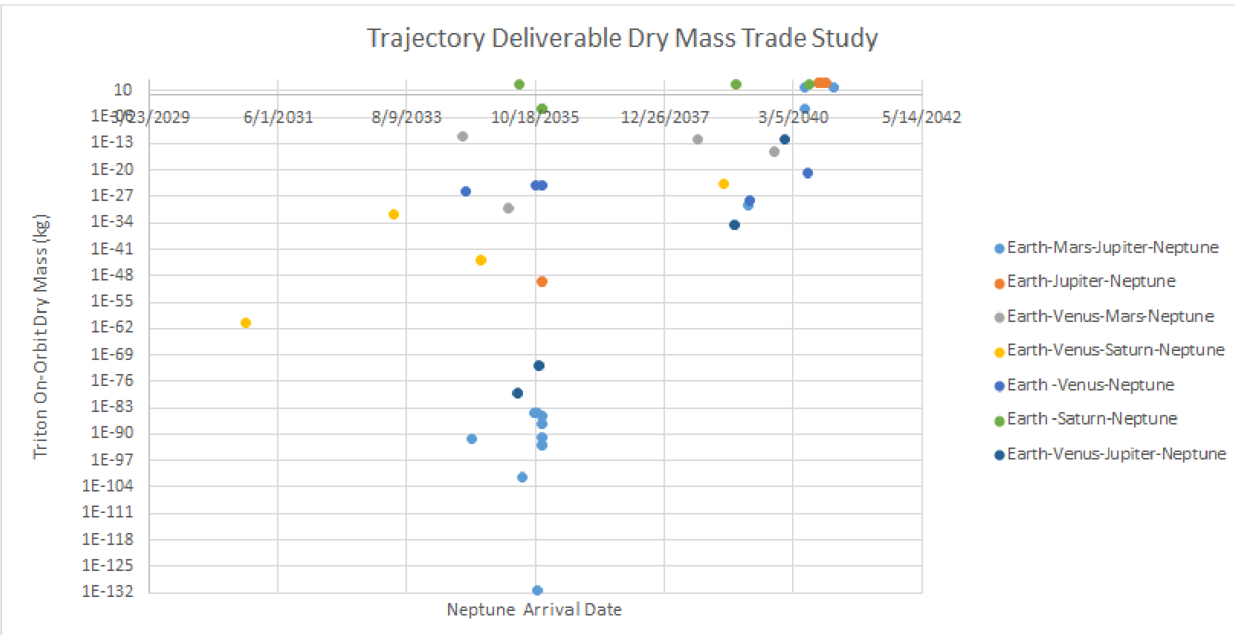


Figure 4.3-1 *Deliverable Mass Trade Study*

The data points are color coded based on the order of planetary bodies used as a flyby opportunity. The horizontal axis represents the arrival date at Neptune's sphere of influence. The vertical axis represents our deliverable mass to Triton on a logarithmic scale. As can be seen, most of our trajectory solutions are woefully incapable of delivering a reasonably sized spacecraft to Triton. We can get a better look at our trajectory options by eliminating any solution that cannot deliver at least 1 kilogram of mass to Triton.

Figure 4.3-2 shows how the revision drastically removes most of our plausible trajectories. From 43 plausible trajectories, we have down selected to 8 possible trajectories. Most of these trajectories involve flybys of Saturn and

Jupiter. But this plot does not represent the time constraints on the mission. Figure 4.3-3 implements the 2035 deadline, represented below by the red line, and the estimated dry mass of our spacecraft, represented by the purple line below.

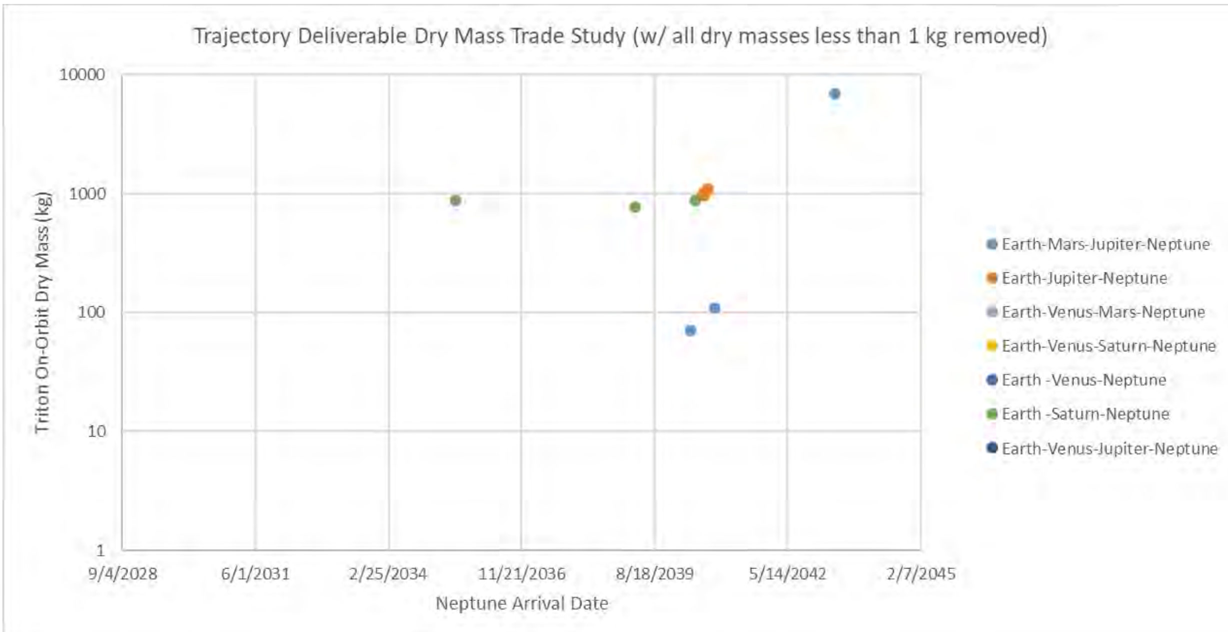


Figure 4.3-2 *Revised Deliverable Mass Trade Study with Only Masses 1 kg And Above*

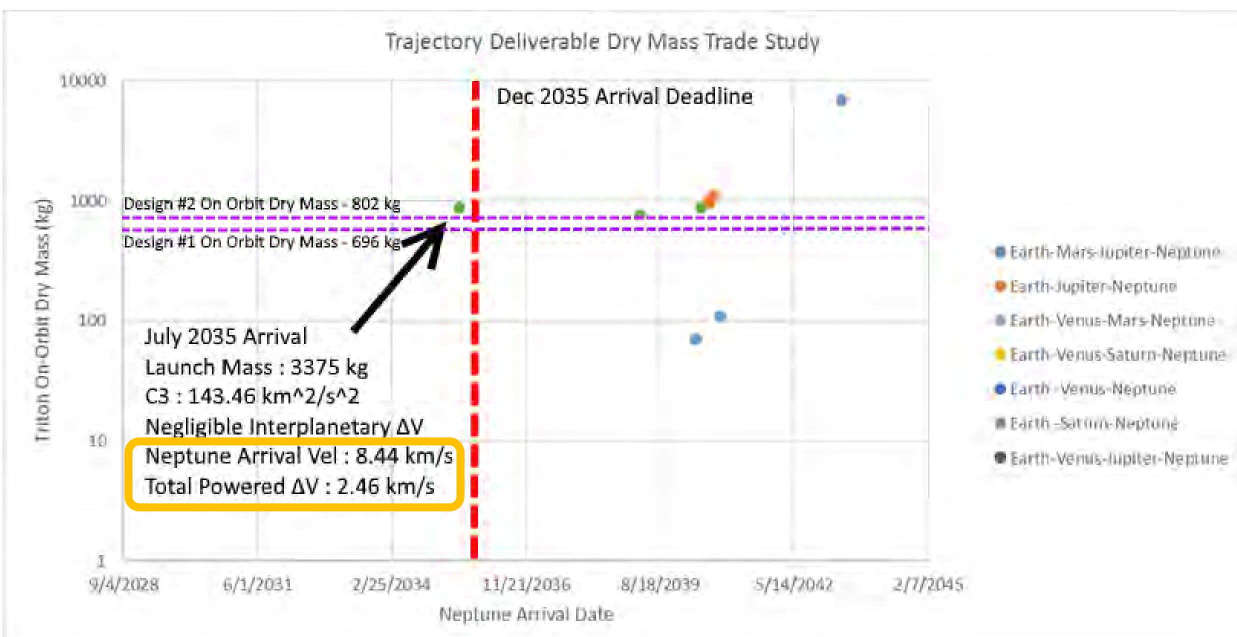


Figure 4.3-3 *Final Revision of Deliverable Mass Trade Study*

Following this criteria, we are left with only one trajectory solution which meets all our requirements. Although the solutions to the right of the plot do not meet the arrival deadline requirements, they do represent several plausible backup trajectories that have similar mass requirements if our program misses their original launch window. Figure 4.3-4 shows the final trajectory for our system.

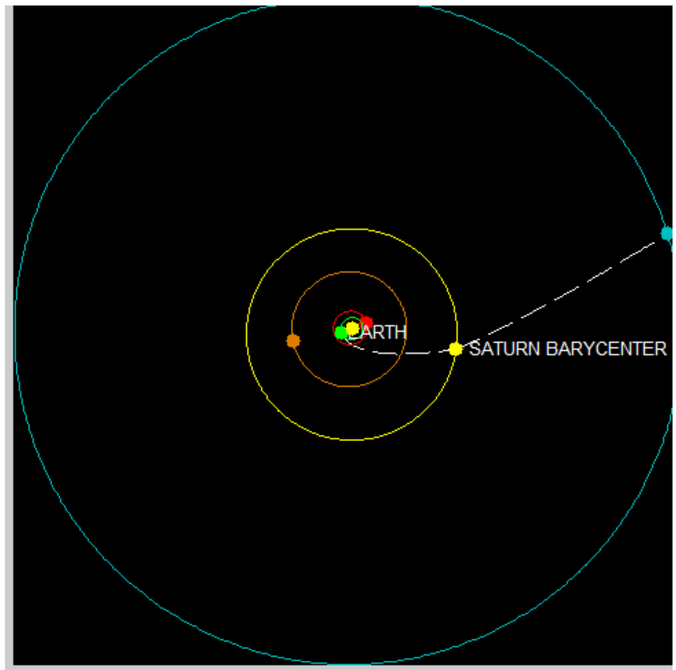


Figure 4.3-4 *Visual Representation of Chosen Earth-Saturn-Neptune Trajectory*

The trajectory launches from Earth in April 2022, flies by Saturn in March 2025, and arrives at Neptune's sphere of influence in July 2035. Due to the time constraints of the mission, our spacecraft uses highly elliptical orbits to travel to Neptune at a very high speed similar to outer-planetary missions. We were fortunate that there was an ideal flyby opportunity with Saturn during this launch flight period, as many other planets that could be used for a slingshot maneuver were not in favorable positions to give us flyby chances.

4.4 Mission Trajectory

This mission can be broken up into the following segments: launch, departure from Earth, transfer to Saturn, flyby through the Saturnian system, transfer to Neptune, capture around Neptune, phasing orbit around Neptune, transfer to Triton, capture around Triton, and parking orbit around Triton. While this is a low number of maneuvers for an interplanetary mission of this scale, this concept of operations was chosen to minimize flight time to meet our strict flight time requirements, and to limit the number of unique environments that our spacecraft would have to survive in. An overview of the transfer developed in the Trajectory Optimization Tool in MATLAB is shown in Figure 4.4-1.

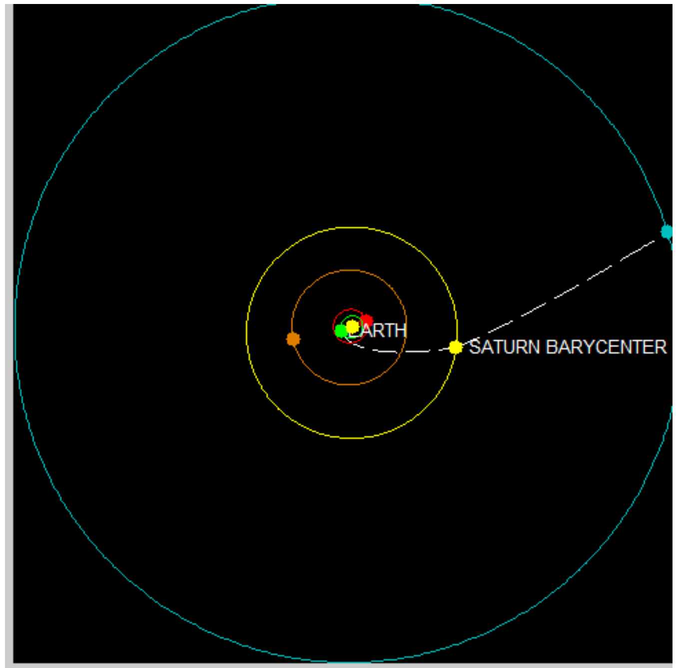


Figure 4.4-1 *Overview of Mission Trajectory*

Launch will be handled by the SLS Block 1B. The spacecraft will take off from Cape Canaveral on April 12th, 2022, and be lifted onto a transfer trajectory towards Saturn with a C3 of 143.46 km^3/s^2. To achieve this, the full lifting power of the SLS Block 1B cargo variant will be used in conjunction with an extra solid propellant kick motor. Our spacecraft will then coast directly to Saturn with nominal minor course corrections along the way.

Once our spacecraft reaches Saturn, it will perform a flyby of Saturn on March 4th, 2025. This flyby will be performed on the trailing side of Saturn with a flyby periapsis of 149 million kilometers. This will be a coasting flyby, requiring a negligible amount of ΔV. This flyby is a significant one, however, in that it will allow us to perform scientific observations on much of the Saturnian system for the first time since the retirement of the Cassini mission.

From this flyby around Saturn, our spacecraft will continue to coast on to Neptune, which it will reach on July 12th, 2035. It will approach Neptune's sphere of influence at a relative velocity of 8.45 km/s. Mid-course corrections on this approach shall line up the spacecraft on the b-plane to match the inclination with the planned Triton interception point and approach Neptune at a periapsis radius of 25,000 kilometers. At periapsis, the spacecraft shall perform a retrograde burn to capture into Neptune's sphere of influence on a phasing orbit that will delay the transfer to Triton until Triton is at the interception point This will be a critical point in the mission, as this will be the first significant burn that the spacecraft's onboard propulsion systems will be handling themselves, and it is happening very late in the mission lifetime, leading to a relatively high risk of malfunction.

After the phasing orbits are completed around Neptune, a low energy Hohmann transfer will occur between our 25,000 km periapsis location at Neptune and an altitude of 870 kilometers above Triton. The phasing orbit burns and Hohmann transfer insertion burns shall have a net ΔV of 1.61 km/s. At the end of the Hohmann transfer, a small 0.85 km/s burn shall be used to insert the spacecraft into a circular polar orbit around Triton at an approximate altitude of 870 kilometers.

At this point, much of the work done by the propulsion system is complete. Small course corrections and inclination changes can be done using the remaining fuel from our fuel margin, and shall be dictated by the changing scientific desires of mission control on Earth. Otherwise, this polar orbit is where our spacecraft is expected to stay for the entirety of its 5-year observation of Triton and its geyser zone. Minimal station keeping requirements are expected, due to the thin, trace atmosphere on Triton, and atmospheric drag should not have a significant effect on our spacecraft's orbit during its observation period.

5.0 Spacecraft Subsystems

5.1 Propulsion Subsystem

The propulsion system is based on the mission operations and trajectory analysis previously stated. The total mission requirement for ΔV is 2.99 km/s, which is broken down in Table 5.1-1. The interplanetary trajectory does not require any powered ΔV except for trimming the trajectory for optimal fly-by entry.

Table 5.1-1 *Mission Lifetime ΔV Requirements*

ΔV Requirements	
Fly-by Trim (km/s)	0.1
Neptune Capture Burn (km/s)	1.61
Triton Capture Burn (km/s)	0.85
Orbital Maneuvers (km/s)	0.43
Total (km/s)	**2.99**

Several requirements come into play based on the mission operations. Due to the capture into Neptune and Triton along with the strict time constraints, a high thrust propulsion system is ideal. Orbit maneuvers and attitude control don't necessarily need a high thrust, but instead low impulse bits and sufficient pointing accuracy. To choose the best propulsion system candidates, a trade study was made shown below in Table 5.1-2. The performance characteristics highlighted in red are the most significant design factors towards deciding the ideal system for main propulsion and attitude control.

Table 5.1-2 *Propulsion System Trade Study [30]*

	Cold Gas	Monopropellant	Bipropellant	Solid	Electric
Typical Application	Attitude Control	Orbit/Attitude Control	Trajectory/Orbit	Trajectory/Orbit	Trajectory/Orbit
Total Impulse (N-s)	<2,500	<45,000	>45,000	>45,000	>1,000,000
Specific Impulse (s)	50-120	180-245	200-468	300	200-8,000
Min Impulse Bit (N-s)	0.0005	0.005	0.025	N/A	0.000001-2
Thrust Level (N)	0.01-1	0.2-4,500	25-8,000,000	50,000-20,000,000	0.013
Feed System	high pressure gas supply	high pressure liquid supply	turbo pump (large) pressure feed (small)	N/A	high pressure gas supply
Propellant Type	N2,Ar,Kr,H2,He	Hyrdrazine w/Catalyst	NTO/N2H4	Al,Cl,OH,O2,O3	Ar,Xe,H2,N2,Bi
# of Restarts	several thousand	several thousand to hundreds of thousand	1 to 4	0	hundreds of thousand
Total Firing Duration	several hours	seconds to minutes	seconds to few minutes	seconds to few minutes	hours to years
Shortest Firing Duration (s)	<0.5	<0.5	<5	N/A	<2
Life in Space	10+ years	10+ years	10+ years	10+ years	15+ years

The high thrust requirements immediately expelled cold gas and electric because it would take too long to capture into both Neptune and Triton. Solid propellants were expelled because of its inability to throttle, which is vital to prevent over or undershooting the capture burns. Lastly, monopropellants were expelled because of its low specific impulse, which could lead to shorter than necessary burns during capture. This leaves bipropellants as the most ideal for the main propulsion system.

Bipropellant systems utilize either cryogenic or storable propellants. Due to the long mission time, cryogenic propellants would require too much weight for insulation and too much power for a cooling system. The chosen storable propellants are hydrazine and variants of nitrogen tetroxide. Hydrazine is chosen because it is the best performing storable propellant due to its hypergolic properties, specific impulse, and density impulse.[37][38] Thrusters were researched before concluding on the propellant types because thrusters require different propellants. In Table 5.1-3, a main bipropellant thruster trade study was done to determine not only what thruster would best fit our mission, but also what propellants are necessary for that thruster. The most significant design factors for this trade study are, again, highlighted in red.

Table 5.1-3 *Main Propulsion Thruster Trade Study*[39][40][41]

Engine	Manufacturer	Fuel	Oxidizer	Mass (kg)	Power (W)	Thrust (N)	Isp (s)	It (kN-s)	It-min (N-s)	O/F	Pc (Bar)	Flow Rate (kg/s)	Experience (# of Missions)
HiPAT	Aerojet	MMH	NTO	5.44	46	445	323	20016	35.6	1	9.4	0.141	100
R-4D	Aerojet	MMH	NTO	4.31	46	490	315.5	20016	15.6	1	7.45	0.158	850
AMBR	Aerojet	Hydrazine	MON-3	5.4	45	623	333	5586		1	13.8	0.204	650
TR-308 Dual Mode	NGC	Hyrdrazine	NTO	4.76	46	472	322	11418	-	1	14.13	0.149	N/A
Leros 1b	MOOG	Hydrazine	MON	4.5	46	635	317	13018	-	0.85	16.2	0.204	70
Leros 1c	MOOG	Hydrazine	MON	4.3	46	458	324	14198	-	0.85	16.5	0.144	70

Based on the design factors shown, the best balance between thrust, specific impulse and mission experience, the Aerojet AMBR thruster was chosen. The propellant it uses is hydrazine for fuel and mixed oxides of nitrogen (MON-3) for oxidizer. The thruster dimensions are with a diameter of 0.362 meters and a length of 0.663 meters. The nozzle has an expansion ratio of 400:1. One significant property of this thruster is the dual-mode capability.

A very important aspect of hydrazine is that it is capable of being used in a dual-mode system, which is for both the main propulsion and the attitude control. Utilizing a dual-mode system saves weight and cost of producing compared to two separate systems. Just as the Juno and Mars Global Surveyor missions featured dual-mode propellant systems, so will our spacecraft.[42][43] Since a dual-mode propellant system combines the main propulsive thrusters with the attitude control thrusters, the attitude control propulsive system will be featured in this section. The non-propulsive attitude control will be featured in its own section.

Now that the thrusters have been chosen, the quantity of propellant can be calculated. This was based on the Δv requirements stated earlier as well as a desaturation burns for the reaction wheels throughout the mission. The desaturation burns allocate 20.5 kg of fuel and the rest of the attitude control burns allocate 64 kg of fuel. A fuel contingency of 5.5% the initial fuel mass was allocated for any potential risks occurring during transit. Using the rocket equation shown in Brown[31], the total propellant required including contingency for main burns is 1826 kg. The Aerojet AMBR thruster requires a fuel-to-air ratio of 1.0. Adding the fuel for the attitude control system, the total fuel mass is 940 kg and the total oxidizer mass is 802 kg.

For propellant tank sizing and material selection, the main factor that came into play was the mass. Several material types were researched including space-grade aluminum, titanium, and carbon fiber composite. Carbon fiber composite was the most ideal because it provided the highest strength-to-density ratio. The tensile strength of the composite is 2450 MPa and the density is 1770 kg/m^3. Though the manufacturing composite tanks is more complex, it is one-third of the mass of titanium 6Al-4V tanks.

The composite tank design focused on manufacturing for the best performance. It is important to mention that since there is more fuel than oxidizer and the density of fuel is lower than the oxidizer, the tank volume for the fuel tank is going to be larger. It was decided that, to save space and lessen the manufacturing complexity, the fuel would be split into 2 tanks. Gathered from strength tests and composite behavior analysis, the optimum winding angle for which to lay the fibers was 54.7 degrees.[44] Per industry standards, a factor of safety of 2.0 was implemented during these tank design calculations. Assuming a 60% fiber volume, the wall thickness of the fuel tanks are 2.05 mm, the oxidizer tank is 2.23 mm, and the pressurant tank is 12.75 mm. This converts to 5 fiber layers in the fuel and oxidizer tanks and 29 fiber layers in the pressurant tank assuming a 0.7 mm thickness per layer. The masses of each tank including the hardware can be seen in Table 5.1-4.

Table 5.1-4 *Propulsion Tank Sizing*

Tank Type	Mass Per	Volume	Wall Thickness	Tank Radius
	(kg)	(m^3)	(mm)	(m)
Fuel	10.7	0.478	2.05	0.487
Oxidizer	13.9	0.616	2.23	0.530
Pressurant	36.5	0.16	13.8	0.352

As mentioned before, the chosen feed system is a dual-mode system to save weight and cost, and allow the thrusters to utilize the same system. The system was based on the Mars Global Surveyor propulsion feed system, but components are altered to lower the weight and complexity. [43] Figure 5.1-1 portrays the feed system with descriptions of each component.

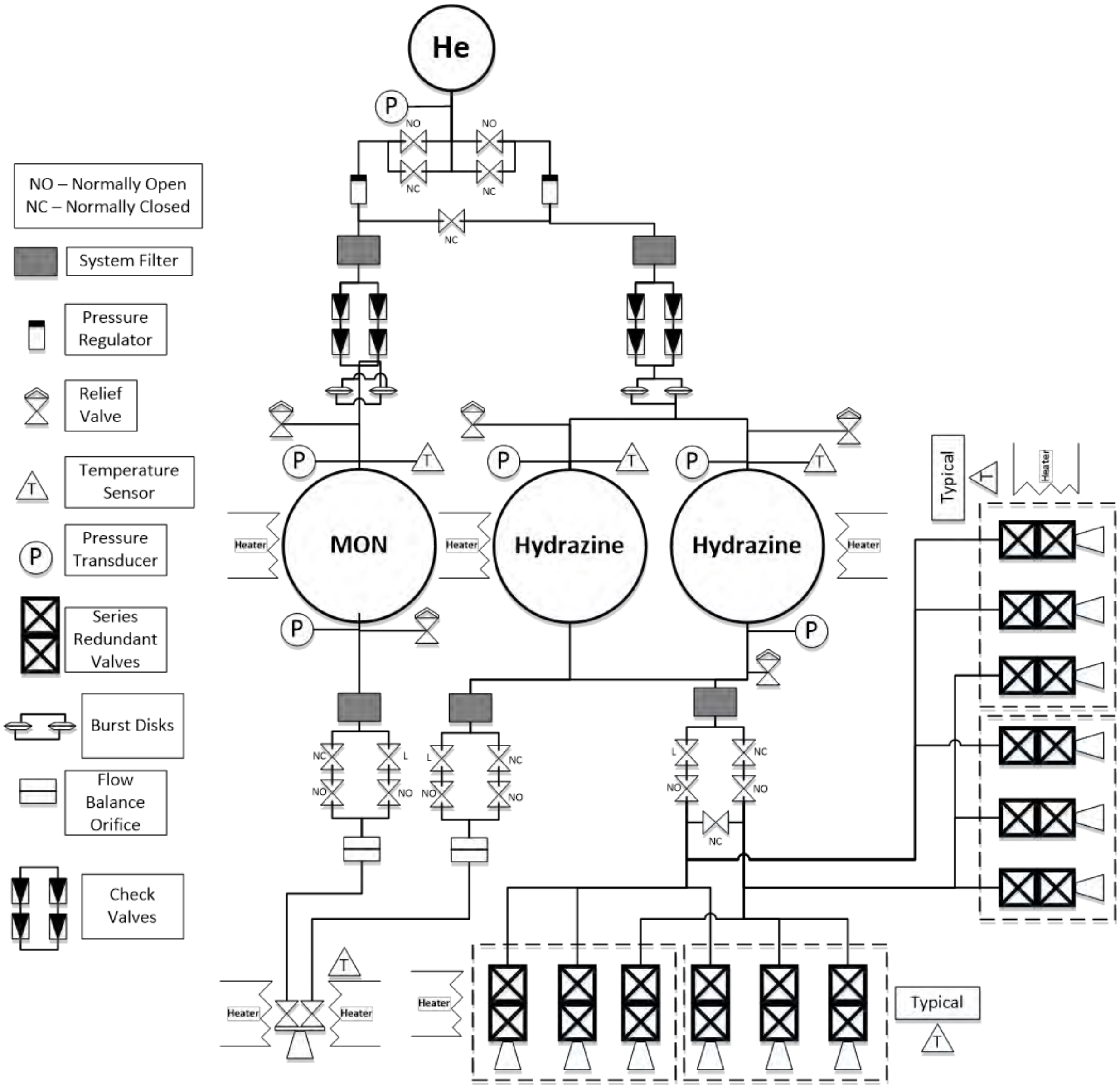


Figure 5.1-1 *Spacecraft Propellant Feed System*

Each component was researched to keep the mass and power requirements low without losing out on reliability. Table 5.1-5 shows a breakdown of the feed system and propellant tanks ultimately providing the entire propulsion subsystem estimation.

Table 5.1-5 *Propulsion System Component Breakdown*[45][46]

Item Listing	Quantity	Mass (Kg)	Power (W)	Supplier	Total Mass (Kg)	Total Power (W)
Fill and Drain Valve	2	0.113	0	VACCO	0.23	0
Fuel Tank	2	10.74	0	NASA	21.5	0
Oxidizer Tank	1	13.86	0	NASA	13.9	0
Pressurant Tank	1	36.54	0	NASA	36.5	0
Mounting Hardware	1	56.51	0	NASA	56.5	0
Pressure Transducer	6	0.113	1	GP:50	0.68	6
Temperature Sensor	6	0.226	1	GP:50	1.36	3
Open Pyro Valve	8	0.154	1	VACCO	1.23	0
Closed Pyro Valve	7	0.154	1	VACCO	1.08	0
Solenoid Valve	24	0.085	9.8	VACCO	2.04	19.6
Latching Valve (No Relief)	3	0.34	1	VACCO	1.02	3
Pressure Regulator w/Filter	2	1.04	0	VACCO	2.08	0
Check Valve	8	0.136	0	VACCO	1.09	0
Relief Valve	5	0.34	0	VACCO	1.70	0
Flow Balance Orifice	4	0.114	0	VACO	0.46	0
System Filter	5	0.114	0	VACO	0.57	0
Main Thruster	1	5.4	45	MOOG	5.40	45
Piping	1	12.68	0	NASA	12.7	0
				Total Mass (Kg)	**160.0**	
				Total Power (W)		**76.6**

5.2 Attitude Determination and Control System

The requirements for the attitude control systemare 0.1 degree pointing accuracy at Earth, a 0.001 degrees pointing accuracy at Triton, a maximum response time of 100 seconds to rotate the spacecraft about each axis 180 degrees, and maintain stability throughout the mission. These requirements were derived from data trends that were presented in Brown [3] and from scientific payload requirements. Based on the pointing accuracy requirements, the only viable method of attitude control is 3-axis stabilized because it provides a high enough pointing accuracy.

The 3-axis stabilization and pointing accuracy will be achieved using a series of different attitude control sensors. These sensors will determine the orientation of the spacecraft at any given time with respect to multiple stars and Triton's orbit. The types of sensors to be used include inertial measurement units, sun sensors, and star trackers.

The IMUs are units that each use 3-4 magnetometers and gyroscopes in order to measure and report the spacecraft's angular rate, specific force, and magnetic field. A trade study of IMUs can be seen in Table 5.2-1 with the major design factors highlighted in red.

Table 5.2-1 *Inertial Measurement Unit Trade Study.*[47][48][49]

	SIRU™	SIRU™-L	SIRU™-E	MIMU	CIRUS
Manufacturer	NGC	NGC	NGC	Honeywell	Space & Navigation
Mass (kg)	7.1	7.1	7.1	**4.7**	13.5
Power (W)	43	43	43	32	**25**
Volume (mm^3)	180x149x289	180x149x289	180x149x289	233 (diameter)x169	396 (diameter)x203
Bias Stability (deg/hr)	0.0015	0.0015	**0.0005**	0.005	**0.0005**
Scale Factor Short Term Stability (ppm)	<5	10	<5	≤1	<2
Scale Factor Non-Linearity (ppm)	<20	<40	<20	-	20
High Accuracy Mode Rate Range (deg/sec)	± 7	± 7	**± 3**	-	**± 3**
Noise Equivalent Angle (arcsec)	<3	<3	**<1**	-	-
Angle White Noise (arcsec/√hr)	0.003	0.009	0.0015	-	**0.000025**
Angle Random Walk (deg/√hr)	0.00015	0.0002	**0.00005**	-	0.00025
Operational Temp Limits (°C)	-10 to 60	-10 to 60	-10 to 60	-30 to 65	-20 to 60

Based on the following design factors, the CIRUS by Space & Navigation was chosen due to lower power requirements and higher resistance to white noise error. The CIRUS does have a larger mass, but the lower power consumption more than compensated for its disadvantages.

The STRs will measure the position of the stars to further achieve a high degree of accuracy for the spacecraft's orientation. An STR trade study can be seen in Table 5.2-2, where the major design factors include field of view, number of stars each can track simultaneously, frequency errors, and lifetime in years. Based on this data, the AA-STR by Leonardo will be used for its superior field of view and trackable stars compared to the other low-mass and low-power options.

Table 5.2-2 *Star Tracker Trade Study*.[50][51][52]

	OG-STR	A-STR	AA-STR	SED26
Manufacturer	Officine Galileo	Leonardo	Leonardo	Sodern
Mass (kg)	2.85	3.55	**2.6**	3.7
Power (W)	8.5-10.5	8.9-13.5	**5.6-12.6**	9.9-13.5
Volume (mm^3)	158x146x355	195x175x290.5	164x156x348	170x160x350
Dynamic Range (Mi)	1.5-5.5	1.5-5.5	1.5-5.5	-
Field of View (deg)	16.4x16.4	16.4x16.4	**20x20**	-
Trackable Stars	10	10	**15**	10
Tracking Rate (deg/sec)	1	2	2	-
Acquisition Time (sec)	<10	<6	<9	<3
Bias Accuracy (arcsec)	13 (all axes)	8.25 (pitch), 11.1 (roll)	8.25 (pitch), 11.1 (roll)	11 (all axes)
Low Frequency Error (arcsec)	7 (pitch)	3.6 (pitch)	**3.3 (pitch)**	4 (pitch)
	30 (roll)	21 (roll)	**15.6 (roll)**	20 (roll)
Random Error (at 0.5 deg/sec; arcsec)	15 (pitch)	6 (pitch)	**6 (pitch)**	-
	135 (roll)	63 (roll)	**49.4 (roll)**	-
Update Rate (Hz)	10	10, 4	10, 8, 5, 4	10
Operational Temp Limits (°C)	-25 to 60	-30 to 60	-30 to 60	-30 to 60
Storage Temp Limits (°C)	-	-35 to 70	-35 to 65	-40 to 70
Lifetime (yrs)	12	18	**18**	15-18

The sun sensors placed on the spacecraft will essentially be a specialized star tracker that exclusively analyzes the spacecraft's position relative to the sun. These sensors shall be used for gyro updating for the other star trackers, as well as a failsafe to be used in case of star tracker malfunction. A trade study of three sun sensor candidates can be seen in Table 5.2-3 with major design drivers highlighted in red. Based on the amount of information found for the Fine Sun Sensor (FSS) manufactured by Bradford that will be the sun sensor of choice for this spacecraft.

Table 5.2-3 *Sun Sensor Trade Study*.[53][54][55]

	Coarse Sun Sensor (CSS)	Fine Sun Sensor (FSS)	Digital Sun Sensor (DSS)
Manufacturer	Moog	Bradford	NewSpace Systems
Mass (kg)	0.215	0.375	0.035
Power Consumption (W)	Negligible	0.25	.0375 to .375
Volume (mm^3)	110 x 110 x 30	108 x 108 x 52.5	34 x 32 x 20
FOV (deg)	102	138	140
Resolution (deg)	-	0.03	-
Noise Equivalent Angle (deg)	Negligible	0.05	-
Alignment Accuracy (deg)	0.16	0.05	0.1
Temperature Limits (°C)	-80 to 120	-50 to 85	-25 to 75

Since the spacecraft will also be utilizing a dual-mode system, the thrusters must use the same fuel type, hydrazine. Using Table 5.1-2 in the propulsion section, the most ideal type of propulsion system for the attitude control system is monopropellant due to the high number of pulses as well as the large range of thrust. Using Table 5.2-4 shown below, the ideal thruster was chosen based on the most significant design factors highlighted in red.

Table 5.2-4 *Attitude Control Thruster Trade Study*[56][57][58][59]

Engine	Manufacturer	Fuel	Catalyst	Mass (kg)	Power (W)	Thrust (N)	Isp (s)	It (kN-s)	It-min (N-s)	# of Pulses	Pc (Bar)	Flow Rate (kg/s)
				(kg)	(W)	(N)	(s)	(kN-s)	(N-s)	()	(Bar)	(kg/s)
MR-103M	Aerojet	Hydrazine	S405	0.16	10.9	0.99	221	122	6.70E-06	515,344	5.9-20.7	0.00045
MRM-103D	Aerojet	Hydrazine	S405	1.27	8.25	1.02	224	128	0.03	210,238	5.9-23.4	0.00045
MRM-106D	Aerojet	Hydrazine	LCH-207/202	2.7	20.1	40	234	91	0.63	7629	17.2/11.0	0.0174
MR-106L	Aerojet	Hydrazine	S405/LCH-202	0.590	41.7	34	229	561	0.15	120,511	4.1-13.4	0.0151
MR-111C	Aerojet	Hydrazine	S405	0.33	13.64	5.3	215	260	0.08	420,000	3.4-12.1	0.0025
MRE-5.0	NGC	Hydrazine	N/A	1.5	30	36	232	-	-	28,512	4.83-32.75	0.0158
MER-1.0	NGC	Hydrazine	N/A	0.5	15	5	218	-	-	457,849	0.55-38.96	0.0023
Monarc-5	MOOG	Hydrazine	N/A	0.49	18	N/A	226.1	614	0.003	205,000	5.5-29.0	N/A
Monarc-22-6	MOOG	Hydrazine	N/A	0.72	30	N/A	229.5	534	0.312	230,000	4.8-27.6	N/A

Based on power consumption, thrust level, minimum total impulse and number of pulses, the MR-111C was chosen as the main attitude control propulsive system. Though the power consumption was not the lowest, it provided a sufficient thrust level and number of pulses. 12 of these thrusters are located around the spacecraft to provide full 3-dimensional rotation. Figure 5.2.1 portrays the locations of the attitude thrusters on the spacecraft with their respective torque arms.

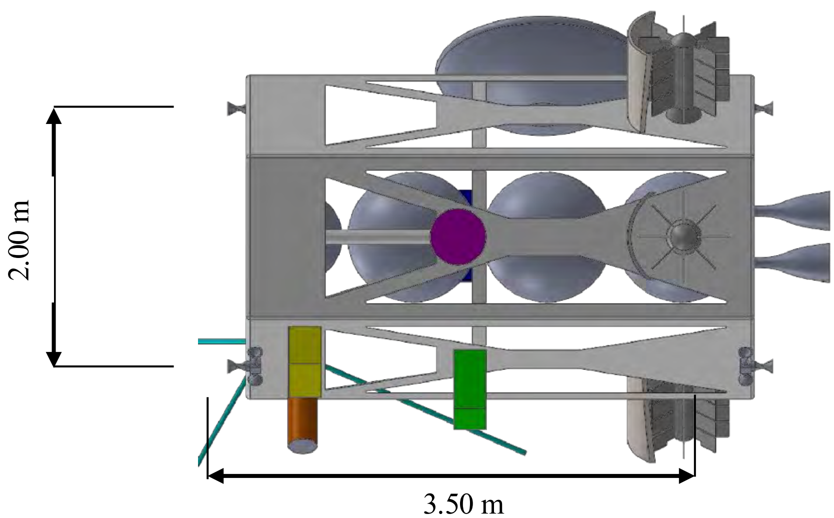


Figure 5.2-1 *Attitude Control Thruster Locations*

The response time and pointing accuracy for each axis is based on the moment of inertia of each spacecraft configuration and is presented in Table 5.2-5 below. There are two spacecraft configurations for this mission; the first is the spacecraft with the lander attached, and the second is the spacecraft after the lander has detached. For most of the mission life, the configuration will be the first configuration until an ideal landing location is found.

Table 5.2-5 *Attitude Control Propulsive System Characteristics*

	Configuration with Lander			Configuration without Lander		
Axis of Rotation	XX	YY	ZZ	XX	YY	ZZ
Moment of Interia (kg-m²)	66934	23453	45865	56682	21535	37379
Input Turn Angle (deg)	180	180	180	180	180	180
Pointing Accuracy (deg)	1.43E-05	4.08E-05	1.19E-05	1.69E-05	4.44E-05	1.46E-05
Response Time (s)	92.2	54.6	101.0	84.9	52.3	91.2

The propulsive attitude control requires propellant to function, but due to the mission consisting of orbiting a body with disturbances and strict mass limitations, non-propulsive attitude control methods were implemented. There are several types of non-propulsive control systems available today, all having their advantages and disadvantages. Table 5.2-6 shows a brief trade study on the three most common types with the most significant design factors highlighted in red.

Table 5.2-6 *Non-Propulsive Attitude Control Trade Study*[31]

	Reaction Wheels	Control Moment Gyros	Magnetic Torque Rods
Complexity	Medium	High	Medium
Reliability	Medium	Medium	High
Mass	Medium	Medium	Low
Avg. Power Consumption	High	Medium	Low
Torque Capability	0.001-10	0.1-100	0.001-1
Consistency	High	High	Medium
Desaturation	Yes	Yes	No

The most appealing type of non-propulsive attitude control seems to be the magnetic torque rods because it provides the same torque capability as the reaction wheels at a fraction of the power consumption, but there are some issues with torque rods that lead to choosing the reaction wheels instead. Triton's magnetic field is unknown because the only data we have on it came from the Voyager mission[60]; Neptune's magnetic field is known to be chaotic and is very likely to affect any use of torque rods around Triton[61]; and torque rods tend to have a delayed response because it takes time for the rods to react to the magnetic field. Because the torque rods are low mass and low power, we will be adding two for experimental control and leaving the main control to the reaction wheels.

Deciding what reaction wheels would most benefit our system came down to the significant design factors shown in Table 5.2-7 highlighted in red. Reaction wheels tend to fail or malfunction, therefore reliability is one of the most important to consider. Another method to prevent failure is to include a backup wheel mounted on a gimbal to replace any of the original three wheels if any were to fail. Momentum capacity is directly proportional to the saturation time of the wheels due to orbital disturbances, so wheels with high momentum capacity is preferred so that the ACS thrusters don't have to desaturate constantly. Reaction wheels also typically consume a lot of power, so the trade-off between power consumption and momentum capacity was chosen carefully.

Table 5.2-7 *Reaction Wheel Trade Study*[62][61][62][63]

Name	Manufacturer	Mass (kg)	Peak Power (W)	Steady Power (W)	Wheel Speed (rpm)	Momentum (Nms)	Torque (Nm)	Life (years)	Vibration (Grms)
HR12	Honeywell	7	105-195	22	6000	25	0.1-0.2	15+	13.8
HR14	Honeywell	8.5	105-195	22	6000	50	0.1-0.2	15+	13.8
HR16	Honeywell	10.4	105-195	22	6000	75	0.1-0.2	15+	13.8
HR0610	Honeywell	4	80	15	6000	8	0.055	10+	19.8
RWA-15	L3	14	230	17	2200	20	0.68	7+	-
RW8	Blue Canyon	3.6	80	7	6000	8	0.11	10+	-
RSI 12-75/60	Rockwell Collins	4.85	90	20	6000	12	0.075	15+	-
RSI 68-170/60	Rockwell Collins	9.5	150	20	6000	68	0.17	15+	-

The Blue Canyon RW8 was chosen because the power consumption remained too high until the momentum capacity reached around 10 N-m-s and the spacecraft's power is on a strict budget. The trade-off between momentum capacity and power consumption was close where the momentum capacity of the Honeywell HR16 provided a saturation time 9.4 times the saturation time of the Blue Canyon RW8, but the power consumption was 10.7 times the consumption of the RW8. It was decided that the power consumption was more important because the power budget is too strict. The RW8 mission life isn't ideal, but the implementation of another reaction is expected to extend the overall life of the non-propulsive attitude control system. The reaction wheels were also tested to determine the pointing accuracy and response time to turn the spacecraft about each axis by 90 degrees. Table 5.2-8 portrays those response times for each spacecraft configuration. The response times are unreasonably high, but if any quick axis turns are necessary, then the attitude control thrusters must take over.

Table 5.2-8 *Reaction Wheel Pointing Accuracy & Response Time*

	Configuration with Lander			Configuration without Lander		
Axis of Rotation	XX	YY	ZZ	XX	YY	ZZ
Moment of Interia (kg-m²)	66934	23453	45865	56682	21535	37379
Input Turn Angle (deg)	90	90	90	90	90	90
Pointing Accuracy (deg)	2.12E-08	6.05E-08	3.09E-08	2.50E-08	6.58E-08	3.79E-08
Response Time (s)	978	579	809	900	555	731

As for the experimental torque rods, we looked for one that would require low power consumption of the reaction wheels so that our experimentations don't put us over the already strict power budget fairly and a large torque output to prevent any delayed response that would cause an excessively slow response time. This leads to creating a trade study shown in Table 5.2-9 where the significant design factors are torque and power

Table 5.2-9 *Magnetic Torque Rod Trade Study*[66]

MTB Name	Dipole Moment Guaranteed (Am²)	Torque (N-m)	Current (mA)	Mass (kg)	Power (W)	Length (m)
MTB200-L-28V	213	0.158	184	4.06	2.74	1.07
MTB240-L-28V	231	0.172	190	5.00	3.34	1.13
MTB300-L-28V	273	0.203	184	5.62	3.48	1.25
MTB450-L-28V	450	0.335	236	8.93	3.95	1.39
MTB550-L-28V	524	0.390	228	9.79	4.06	1.53
MTB650-L-28V	683	0.509	356	12.3	5.36	1.46
MTB750-L-28V	692	0.515	281	13.2	5.21	1.63

The torque rods chosen are the MTB300-L-28V because they are half the power consumption and double the torque output of the reaction wheels chosen. The spacecraft will mainly be in a polar orbit so the torque rods will be oriented into Triton's core and perpendicular to its orbit so that it can conduct longitudinal changes and in-plane twist maneuvers.

The final component list of the attitude determination and controls are shown below in Table 5.2-10. The total power represents all the components being on at the same time, but the torque rods would turn on once the reaction wheels turn off for experimentation.

Table 5.2-10 *Attitude Determination & Control Component List*

Item Listing	Quantity	Mass (Kg)	Power (W)	Supplier	Total Line Mass (Kg)	Total Line Power (W)
ACS Thrusters	12	0.33	13.64	Aerojet	3.96	27.28
Reaction Wheel	4	8.5	7	Blue Canyon	34	28
Magnetic Torque Rod	2	5.6	4.0	Cayuga Astronautics	11.2	7.91
Gyroscopic Controls	1	13.5	25	Space & Navigation	13.5	25
Star Trackers	3	6.47	9.9	Sodern	19.41	19.8
Sun Trackers	1	0.375	0.25	Braford	0.375	0.25
				Total Mass (Kg)	**82.49**	
				Total Power (W)		**108.24**
				Power (Prop Mode) (W)		**72.33**
				Power (Non-Prop Mode) (W)		**73.05**

5.3 Command and Data Handling

A significant bottleneck in our data acquisition mission is our telecom system. While our instruments are more than capable of taking massive quantities of data, due to the distance between Earth and our spacecraft, and the possible size of our spacecraft, the transmission rate of such data is going to be severely limited.

Our mission architecture requires two antennas, for high gain and low gain transmission and receiving. Received commands will be pre-processed on an uplink card, and telemetry to be transmitted will be formatted by a downlink formatter, which will use a Reed-Solomon (255,223) encoding to increase reliability in the transmitted signal. These commands and telemetry will be processed by our radiation hardened on board processor, the 12 MHz Mongoose V.

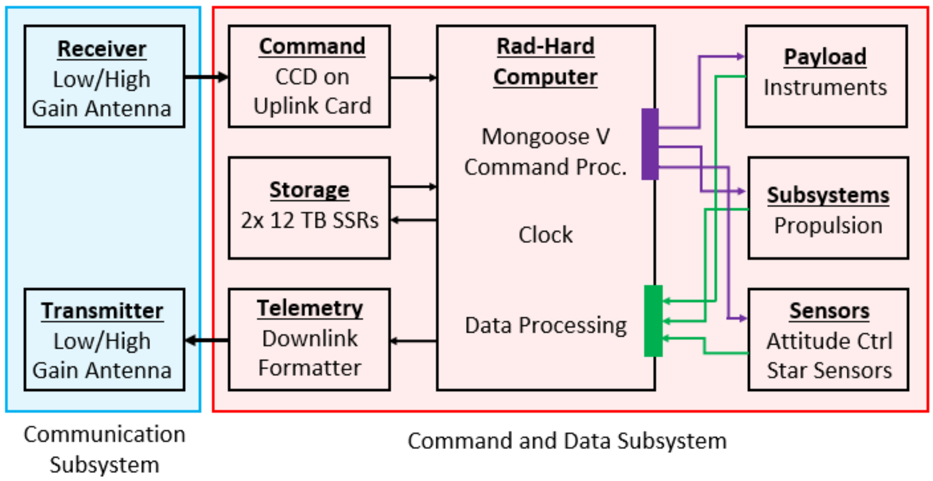


Figure 5.3-1 *Data Processing Flow*

Due to the vast difference in transmission distance, the low gain communication links between the lander and the orbiter will have much higher performance than the high gain link between the orbiter and the Earth. However, this data will need to be buffered on the orbiter before it can be transmitted to the DSN. This will increase the data storage requirements of the orbiter, which will have two 12 TB solid state recorders.

While all our instrumentation will contribute to the overall data storage requirements of the orbiter, the clear majority of raw data that will be collected by the orbiter will be from its visible imagine cameras, which will be performing a full mapping of Triton's surface as Triton revolves underneath the orbiter's polar orbit. This surface mapping imagery shall be collected by the HRCC, UVS and VIS at a rate of 10.08 megabits per second. Every other instrument shall be commanded to point at specific targets of interest at specific times, however, the visible camera shall be pointed towards the surface as often as possible to maximize image collection of the surface. With continuous coverage, it will take two months to collect a complete image of the surface. However, to limit the amount of data that must be stored, significant pre-processing must take place to shrink the size of the data collected. Many portions of the surface will be passed over multiple times due to the intersection of subsequent orbits. These sections shall be checked for image quality and then cut and spliced together to form on continuous image of the moon's surface. This will shrink the volume of imagery that must be transmitted to the DSN. Furthermore, our data volume shall go through a lossless compression algorithm to shrink its size even more. With the similarity in color pallets of the features on Triton, and the signal noise found in images from New Horizons, we're expecting a compression ratio of at least 60%. Table 5.3-1 displays the component breakdown of this subsystem.

An important aspect of the command and data handling system is the fault protection. Fault protection implements software coding to detect, assess, and mitigate subsystem faults such as a valve malfunctioning. An estimated three million lines of code have been incorporated into the subsystem and an additional two million lines of code was incorporated solely for fault protection.

Table 5.3-1 *Command and Data Handling Component Breakdown*

Component	Mass (kg)	Power (W)
C&DH Processor Card	0.15	5.00
Solid State Recorder (SSR) Card	7.10	17.0
Instrument Interface Card	0.07	2.50
Critical Command Decoder (CCD) on Uplink Card	0.10	2.50
Downlink Formatter on Downlink Card	0.10	0.50
Total:	**7.52**	**27.5**

5.4 Telecommunications

The requirement for the telecommunications system on the orbiter states that all data necessary to fulfill the RFP requirements must be done so by the end of the mission in December of 2040. This allows five full years to transmit instrument data from the orbit of Triton to earth's DSN. The orbiter will be allowed to transmit data to the DSN for 8 hours per Earth day, for the entire duration of the mission. The orbiter will implore the use of the X-band frequency and with transmit data at a frequency of 8.42 GHz using a 3.0-meter parabolic HGA shown in Figure 5.4-1. The largest challenge for the transmission of data is the enormous distance between Triton and Earth.

Figure 5.4-1 *3.0-meter Parabolic HGA*

With 50 W of power and the spacecraft located 4.7 billion km from Earth, the antenna will have a maximum and minimum downlink rate to the DSN of 13 kbps and 6.5 kbps, respectively. Using Reed-Solomon turbo coding the spacecraft's telecommunications system will have the capability to transmit 15/7.5 ksps. Over the five years the spacecraft will orbit Triton, this translates to a maximum of 683 Gbits and a minimum of 341 Gbits of science data that can be transmitted from the spacecraft to the DSN. Due to the vast distance the signal must travel, the ACS system must be able to keep a pointing accuracy of 0.10 degrees to maintain constant contact with the DSN. The HGA will be mounted to the spacecraft bus opposite from the spacecraft payload so that the payload can take data while the HGA transmits are specified location in orbit. A low gain antenna is also placed atop the HGA, however, will only be used for close communication with at beginning of the mission and if there is a need for emergency uplink at any point.

To communicate with the Triton lander, an Electra UHF radio (shown in Figure 5.4-2) will be included to the telecommunications system of the orbiter, as well as an Electra-Lite UHF radio on the lander.[67] These radios allow for high data transfer rates at minimal power consumption between the spacecraft and the lander. The Electra requires 67W to transmit and 30W to receive, however due to the nature of the landers objective, the Electra on the orbiter will almost exclusively be receiving. Due to its power consumption, main communications to the DSN will be halted for 1.51 hours while the lander transmits all of its data to the orbiter as it passes over. A helical antenna (Figure 5.4-3) on the lander will be used to transmit from the Electra-Lite at 390-450 MHz to its parent radio on the orbiter.

Figure 5.4-2 *Electra UHF Radio*

Figure 5.4-3 *Helical UHF Antenna*

The helical antenna provides a 180-degree field of view. Using 40W, the Electra-lite can transmit between 2 and 2048 kbps to the orbiter. During the 1.51 hour fly over of the orbiter, the lander will transmit between 8.2 Mbits and 8.4 Gbits of data back to the orbiting vehicle. The orbiter will act as a relay and transmit the data obtained by the lander instrument to the DSN. This will allow for all RFP required data to be transmitted to earth before December of 2040. Table 5.4-1 shows the breakdown of the mass and power for the telecommunications subsystem.

Table 5.4-1 *Mass and Power Statement for TES Telecommunications Subsystem*[68][69]

Equipment	Mass (kg)	Power (W)
Electra Radio & UHF Antenna	**7**	10
Telecom Panel	20	0
Antenna (High Gain, Low Gain)	30	0
SDST	10	2
SSPA (x2)	3	50
Total	**70**	**62**
Total (Xmit to DSN)		**52**
Total (Rx from Lander)		**37**

5.5 Power Subsystem

The power subsystem is the heart of the spacecraft, essentially keeping all other systems alive and running during the course of the mission. This is an integral part of the system, therefore, the choice of power supply was critical in the design process. Primary batteries, while effective, would be too heavy and unpractically large for a mission length of over 18 years. Fuel cells were also explored as an option, however the size and weight associated with the system and tanks for the fuel were also too large under preliminary estimates to be a viable option. Solar cells are generally the best option for spacecraft power, but only within a certain distance from the Sun. As we get further from the Sun, the solar flux received by it decreases drastically, and in order to receive enough power at Triton from solar cells, the solar array would be approximately 1.46×10^8 square meters. This size is ridiculously large and heavy, therefore solar cells are not a feasible option. Nuclear reactors were also considered as an option, but no practically tested nuclear power sources have been created for satellites. The combination of R&D and lack of support from the public would make this a poor choice of power supply for the mission. The only final option we had for our power supply was using radioisotope thermoelectric generators (RTGs).

RTGs have been in use by satellites since the beginning of human exploration; Apollo 12 on the moon in 1969, Voyager 1's deep space mission in 1977, and even the Curiosity Rover on Mars' surface in 2011 all utilized RTGs. In total, the US has had 26 mission using 45 RTGs total and they have never been the cause of an accident. This makes the RTGs a very reliable source of power. RTGs have been used for very similar missions, have a reasonable mass-to-power output, a high reliability, and are perfect for long missions in deep space. There are many types of RTGs that could be used for this mission. One option is the use of General Purpose Heat Source (GPHS) RTGs, which have very high power outputs. However, these types of RTGs are currently out of production and, economically speaking, to restart production for a single mission would be unrealistic. Multihundred-Watt (MHW) RTGs were used on Voyager 1 and 2, but they too are out of production. The next generation of RTGs, known as Advanced Stirling radioisotope generators (ASRGs) boast a higher system efficiency of 23%, a higher electrical power output, a smaller size and a mass compared to previous RTGs. They also keep 85% of their power after 14 years.[72] ASRGs, however, have had many setbacks and will not be ready until 2028. [10] Although this RTG would be ideal, they are unavailable for Project TES. The last RTG analyzed was the Multi-Mission RTG (MMRTG). Alice Caponiti, the DoE's Director of Space and Defense Power Systems, stated that there are two MMRTGs still available for launch by December 31, 2021, and new RTGs can be produced within 5 years.[70] Given this is an RTG still within pour timeline, and it gives reasonable efficiencies and power outputs, it was chosen for Project TES. An example of an MMRTG can be seen in Figure 5.5-1 below.

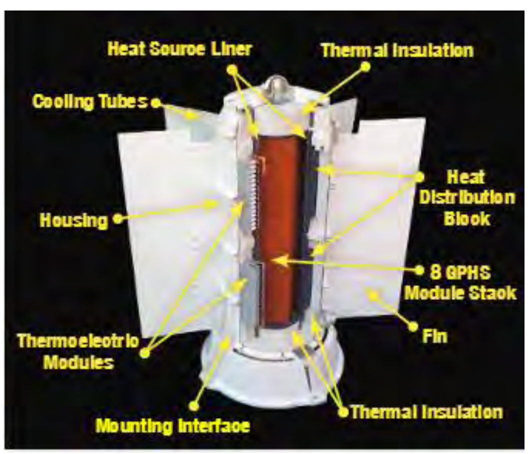


Figure 5.5-1 *MMRTG with Various Components* [71]

From the power budget in Section 3.2 and lifetime assessment in Section 10.0, which will be discussed later, four MMRTGs were chosen to act as the spacecraft's primary power source. Per NASA, each MMRTG contains 3.52 kilograms of Pu-238, the main power source, which provides 2,000 W of thermal power and 110 W of electrical power at beginning of life. Each unit has a mass of 43.6 kg.[71] As mentioned before, 2 MMRTGs are currently available for future missions, both of which Project TES will utilize. The other two MMRTGs must be manufactured. In a Report to Congress given in June 2010 titled *Start-Up Plan for Pu-238 production for Radioisotope Power Systems*, 1.5 kilograms of Pu-238 would be produced and allocated each year for scientific purposes. Given each RTG requires 3.52 kilograms and 5 years to manufacture, the other two will be available by the launch date of 2022.

RTGs are the main power supply, however there is a multitude of support systems that are associated with the main power subsystem. These support units are generalized into three main categories Power Distribution Units (PDUs), Power Regulatory Units (PRUs), and cabling. New Horizons' spacecraft block diagram can be seen as an example in Figure 5.5-2 which will be used as a model to design TES's satellite power subsystem. A fully redundant PDU complete with temperature sensors will be utilized within the power subsystem which will be able to distribute power to the various other systems within the spacecraft. External thermal shunts and a shunt PDU will also be implemented to control excess power from the RTG. The shunt PDU will control the amount of power needed to be rejected in the form of heat or the amount of power to recover by the rejected heat depending on the specific mode the spacecraft will be in. The wiring will be used to connect the various electrical components to the PDU and MMRTG unit. Project TES's power block diagram can be seen in Figure 5.5-3.

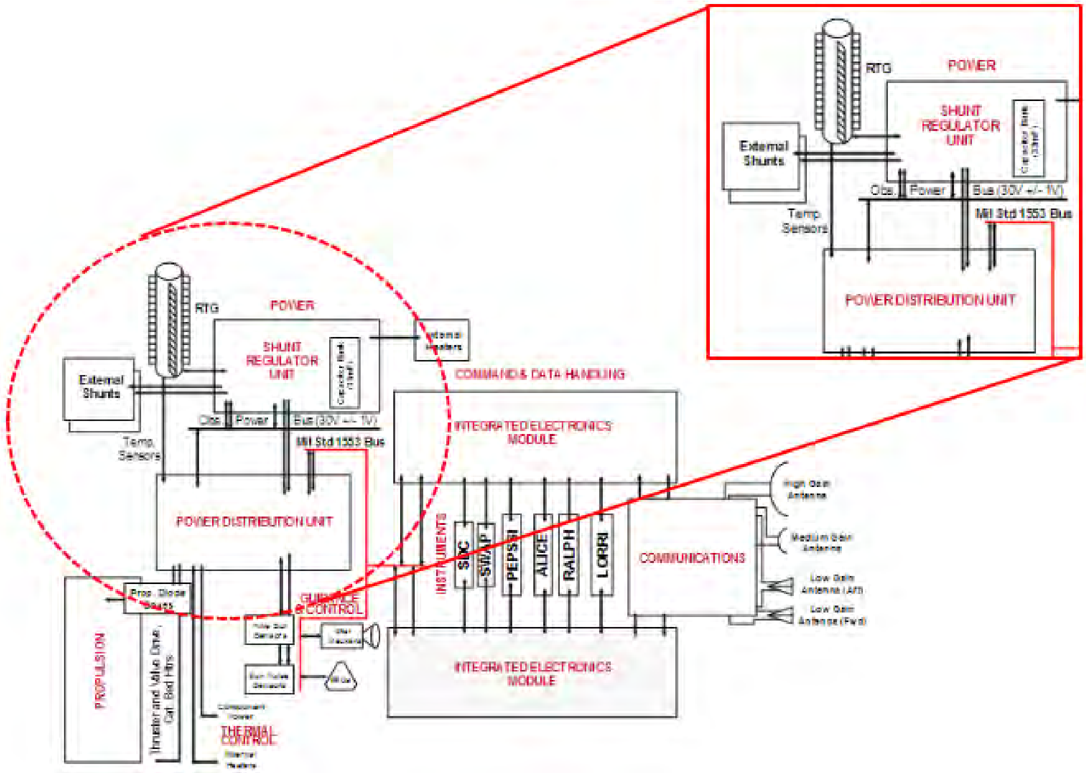


Figure 5.5-2 *Reference New Horizons Spacecraft Block Diagram* [73]

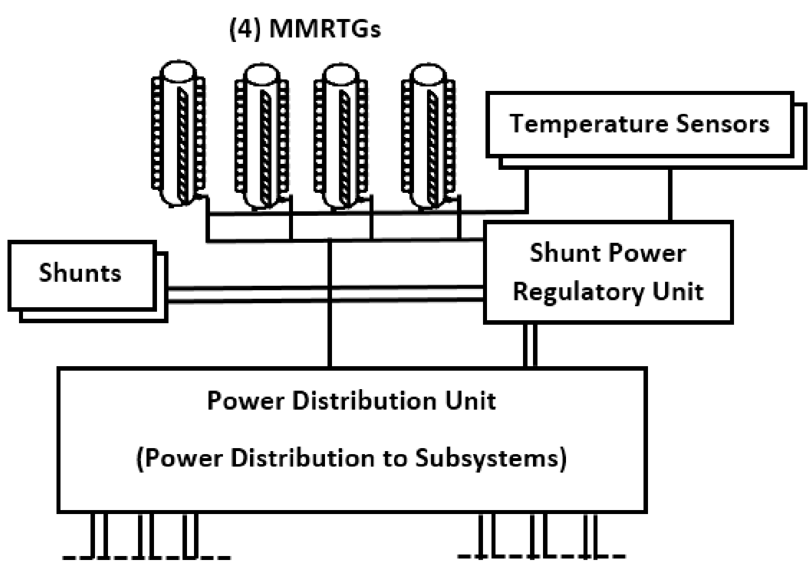


Figure 5.5-3 *Project TES's Satellite Power Block Diagram*

The compilation of the various spacecraft power subsystems can be seen within Table 5.6-1. There will be no batteries within the spacecraft because the MMRTGs power output is consistent and reliable enough that simple capacitors can be used as redundancy in the PDU. The PDU, PRU, mounting hardware, and cabling are estimated values using Brown's past mission estimation correlations.[3] Estimations are used to accurately determine the mass and power since previously used equipment can not necessarily be applied to this specific mission and therefore much be made custom to the spacecraft needs.

Table 5.5-1 *Power Subsystem Mass and Power Statement*

Item	Mass (kg)	Power (W)
Batteries	0.00	0.00
(4) MMRTGs with Attachment Hardware	174.40	0.00
Power Regulatory Unit, Power Distribution unit, Shunts	36.12	45.00
Electrical Cabling	11.21	14.00
Total Power Subsystem	**221.72**	**59.00**

5.6 Thermal Control Subsystem

Thermal control of the spacecraft is difficult to analyze because there are many variables that effect the temperature of an object. In space, there are only two forms of heat transfer: radiation and conduction. Conduction occurs within two materials that are close enough to one another that they are physically 'touching' and a transfer of energy occurs which either raises or lowers the temperature. This heat flux will only occur on the spacecraft itself for the individual components, interactions, and subsystems that will be in direct contact with one another. Radiation is the transfer of energy through waves or particles, this can occur in a vacuum or through a physical medium such as Earth's atmosphere. Due to the laws of physics, the only real way for a spacecraft to remove the heat or energy is to radiate it away from itself. Heat generation can be internal such as the through electronics or Pu-238 within the RTG, but can also occur with radiation from the Sun or other planets.

The first step in understanding any thermal control situation is understanding the surrounding temperatures without the added thermal control to accurately size the system to either raise or lower the temperature. Figure 5.5-1 shows the maximum and minimum temperature limits the spacecraft will experience leaving Earth. The power dissipation includes the amount of electronics that will be power on during this segment which is highly dependent on the mode the spacecraft will be in. The lower temperature limits show if the spacecraft is in complete shade around

Earth and the higher temperature limits show if the spacecraft is in direct sunlight, therefore the temperature range the spacecraft can possibly be in departing from Earth is shown. Figure 5.5-2 shows the temperature limits during transit when the spacecraft is within its transfer mode. As the distance from the sun increases the temperature of the spacecraft will decrease which is to be expected. Figure 5.5-3 shows the temperature limits at Triton which are also within the specified limits however at Triton the sun has less influence of the spacecraft therefore the lower and higher temperature range is much shorter than at Earth.

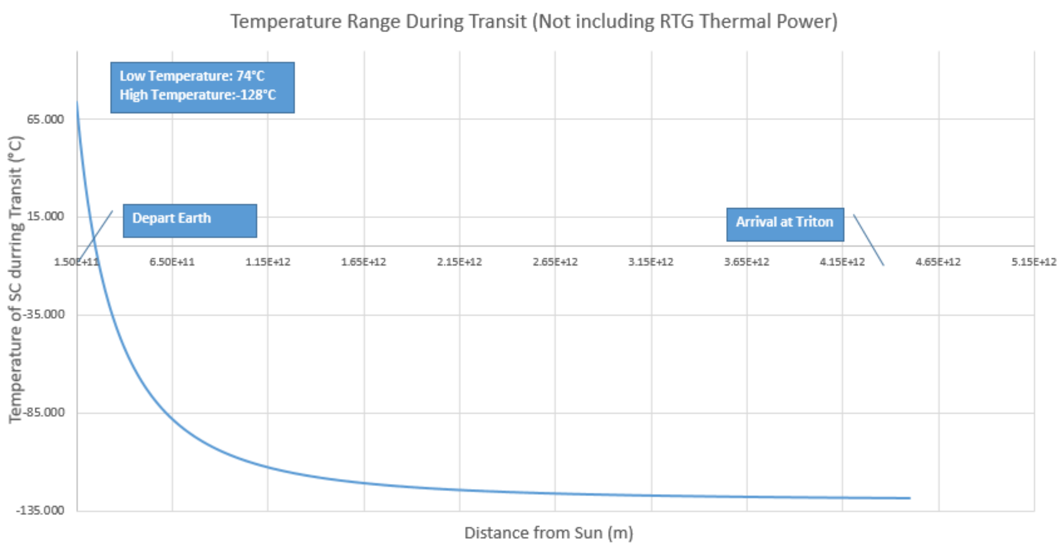


Figure 5.6-1 *Temperature Limits at Earth*

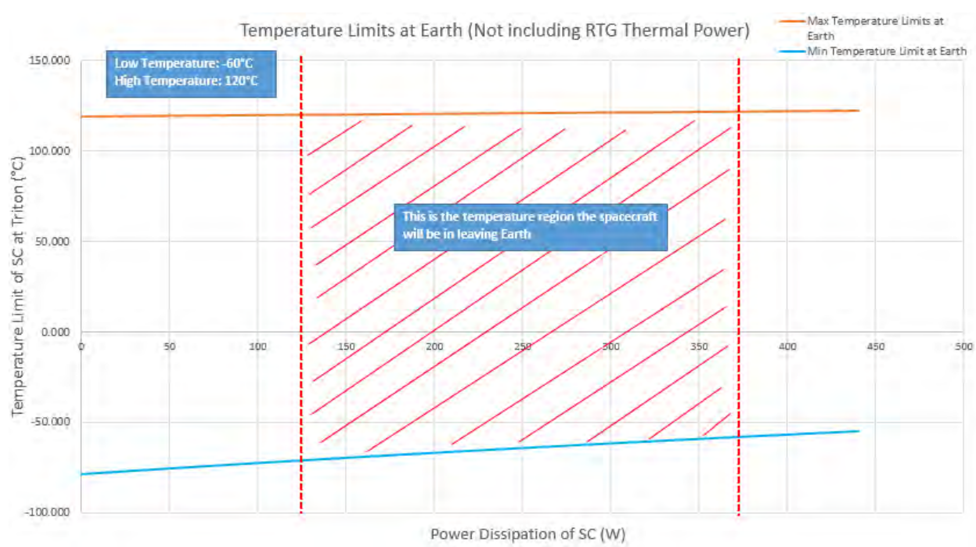


Figure 5.6-2 *Temperature Range During Transit*

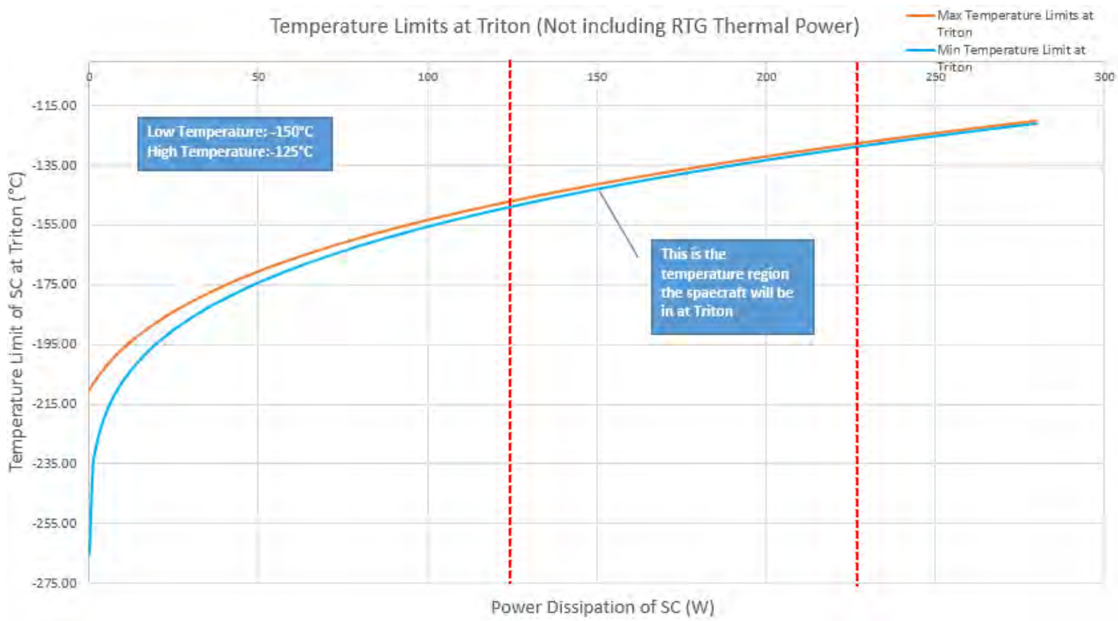


Figure 5.6-3 *Temperature range at Triton*

At Earth, radiators are required to keep the temperature down into the temperature limits of the spacecraft. During transit and arrival at Neptune, power is required to raise the temperature of the spacecraft to operating limits. Limits of the various spacecraft subsystems are shown in Table 5.5-1 below.

Table 5.6-1 *Temperature Limits of Spacecraft Subsystems*

Subsystems	Lower Limit (°C)	Upper Limit (°C)
Electronics and Payload Sensors	0	40
Hydrazine Fuel	7	35
Oxygen Oxidizer	-11.2	21.2
Infrared Detector Sensors	-200	-80
Structures	-46	65
RTGs	-46	200

There are two types of thermal control systems: active and passive. For this unmanned spacecraft, a passive thermal system will be utilized because it is lighter, requires less power, and is cheaper.[72] This passive thermal control will utilize passive copper heat pipes, radiators, louvers, thermal switches, RHUs, and multi-layer insulation. A schematic of these components are shown in Figure 5.5-4.

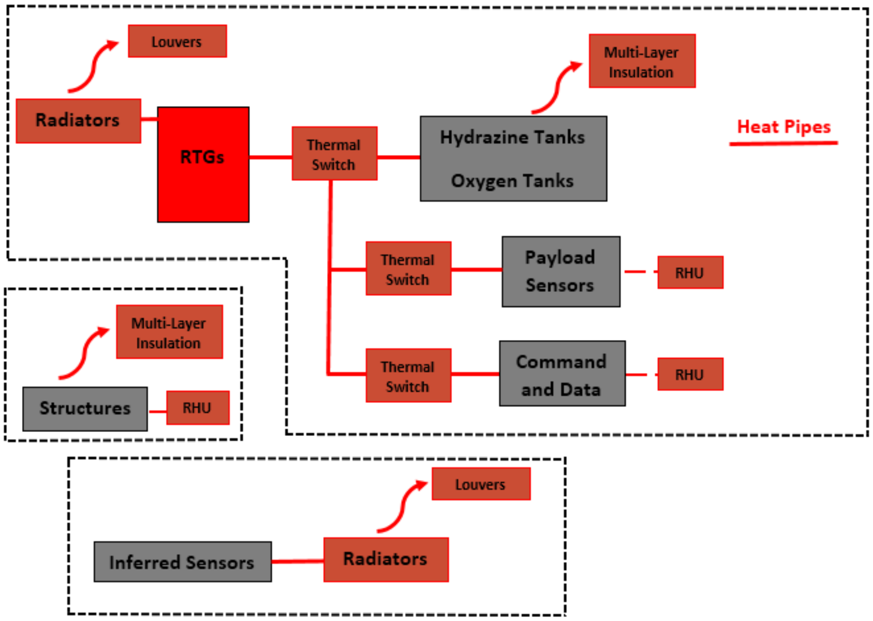


Figure 5.6-4 *Passive Thermal System for Spacecraft*

The passive copper heat pipes have a diameter of 1.27 cm and are used to transfer heat away from the RTG to the various subsystem in the spacecraft. This is ideal because the RTGs will output 8,000W of thermal power at BOL and this will be utilized instead of heaters to save weight and cost. The thermal switches are made from Honeywell and will be implemented at key locations of thermal conduction paths; they will respond to temperature sensors and will open or close the conduction paths to allow heat flux to flow or become disrupted from the heat pipes. The multi-layer insulation will be composed of 12 alternative layers of a light weight film known as Kapton with internal layering of Mylar support material with 5-mil Chromium vapor deposited coating that will provide an emissivity of 0.24 and absorptivity of 0.57.[74] This will be used to insulate key parts such as the structure and fuel tanks by allowing less radiation heat to escape from those areas. 16 RHUs will also be implemented to insure a steady heat flux to the structure of the spacecraft as well as the key electrical subsystems. Radiators will be used to release excess heat from the spacecraft using principles of radiation control. These radiators will be passively controlled with louvers which will control rate of heat flux with bimetallic springs that actuate the louvers' blades on the radiators at specific temperatures. Below is the mass statement for the passive thermal control system including the power and mass calculations per the spacecraft's needs. To meet the lowest upper limit that the spacecraft can handle the largest radiator surface area is sized at 62 square meters insuring the spacecraft does not over heat at Earth. Past Earth the radiators will mostly be inactive and will not be radiating any excess heat. At Triton and during transit, it was found that the RTGs will sustain the spacecraft temperature lower limits through conduction of the heat pipes and thermal switches and temperature sensors. Table 5.5-2 shows the thermal control component's mass and power final calculations.

Table 5.6-2 *Thermal Control Components Mass and Power*

Component	Mass (kg)	Power (W)
Heat Pipes	9.99	0
2 Radiators	7.88	0
3 Thermal Switches	0.06	1
16 RHUs	0.64	0
Multi-Layers Insulation	1.49	0
3 Temperature Sensors	0.09	0.5
Totals	**20.15**	**1.5**

5.7 Structural Elements

The main body of our spacecraft will be a hexagonal structure to incorporate all the payloads to be facing the target and the placement of the subsystems to be equally distributed. The hexagonal structure is constructed with 7075-Aluminum that have properties that can withstand the shear and axial loads of all the payload instruments and subsystem mountings on the spacecraft. An aluminum construction will allow for a higher technical readiness level as it will be feasible to construct the body frame.

In designing the body frame, the extreme load conditions were taken into consideration. The extreme load condition of the spacecraft is the launch loads which is found through the SLS Mission Planner Guide shown on Table 5.7-1. From the SLS launch vehicle, the maximum launch load is 3.5g's. Based on that acceleration, the launch loads were determined by the mass of the spacecraft multiplied by the acceleration. This force was distributed of the bottom section of the payload where the payload attach fitting is connected.

Table 5.7-1 *Missions Planner Guide Launch Loads* [31]

	Lift off	Transonic	Max Q*Alpha	Max G, Boost	Max G, Core
Axial Acceleration, g	2.75	2.00	2.50	3.25	3.50
Lateral Acceleration, g	0.75	0.75	0.50	0.30	0.25

In the initial structure analysis, the main body frame showed a high deflection shown on Figure 5.7-1, where the propulsion system was mounted. This resulted in a very high risk for the structure for failure in the propulsion system thus a revised design was created.

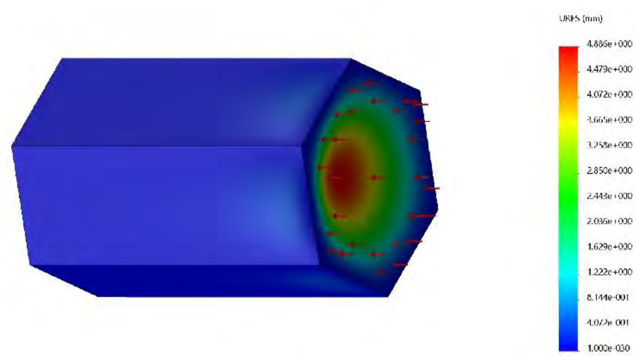


Figure 5.7-1 *Payload Bus FEA Launch Load Analysis*

For the second revision, a beam was used going along the top and bottom part of the structure to add support to the base. This reduced the deflection from the base and created a well distributed stress along the structure. However, the beams were prone to buckling. Thus, the final revision placed a plate at the middle of the structure to address the stability to the columns. The structural analysis of the payload bus is shown below on Figure 5.7-2. The extra holes on the payload bus were created to reduce the mass of the structure.

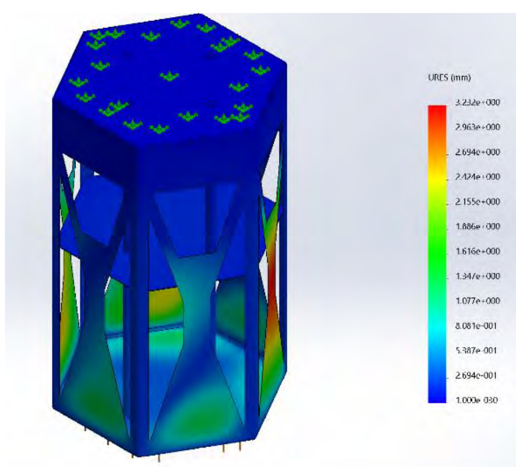


Figure 5.7-2 *Payload Bus FEA Final Revision Launch Load Analysis*

6 Lander Overview

6.1 Determination of Landing Site

The second phase to the science mission involves the Vespucci lander. Vespucci shall detach from TES and soft land on Triton's surface to conduct surface composition analysis and take high resolution surface photographs. The TES Orbiter will spend just over a year mapping Triton's surface to determine an ideal landing site. The most beneficial landing site was determined to be in the geyser region near the plumes because Vespucci will be able to retrieve data of Triton's surface as well as geyser precipitate composition. This data alone would be difficult to differentiate what is surface material and what is geyser precipitate material, but, in conjunction with TES's geyser plume data, the two can be separated and identified. The HRCC and VIS onboard the TES Orbiter will be used to map Triton's geyser region to a ground resolution of 7 meters per pixel.

Vespucci will face some obstacles during its descent. Although the HRCC and the VIS provide a detailed ground resolution, they may not accurately identify smaller geological features that may complicate the landing. This could cause Vespucci to land on an uneven surface and possibly topple over. Another obstacle is that Triton's surface may be softer than expected, which could lead to Vespucci falling through a hollow surface. These obstacles may delay the landing phase until sufficient data is retrieved by TES.

6.2 Subsystems

One of the crucial part of our missions is the ground science operations. Our mission designed a lander that will take data and transmit data to the main spacecraft. The lander subsystems will have all the subsystems that the main spacecraft contain which are the following: Power, Structures, Thermal Control, Propulsion, Attitude Control System, Command and Data Handling, and Telecommunication.

Unlike the main spacecraft, the lander power system will be utilizing Lithium Ion Cells that will power the entire vehicle. At peak operations, the landers maximum power output is 100 W. Using the maximum power output to design the power system, the lander will require 14 cells in series and as redundancy, it will have another line of that in parallel. The total mass of the Lithium Ion cells in the power system is approximately 30.24 kg using the Lithium Ion specification shown below in Figure 6.2-1. The entire power system will be a total of 44.41 kg with 7 kg for the power control electronics and another 7 kg for the cabling weight.

	VES 140	VES 180	VL 48E	VL51ES	VES 16	VL 6P
Guaranteed capacity (Ah)	39	50	48	51	4.5	6.6
Mean voltage at C/1.5	3.6	3.6	3.6	3.6	3.6	3.6
End of charge voltage (V)	4.1	4.1	4.1	4.1	4.1	4.1
Energy (Wh)	140	180	170	180	16	22
Specific energy (Wh/kg)	126	165	150	170	155	65
Height (mm)	250	250	250	222	60	143
Diameter (mm)	53	53	54	54	33	38.2
Weight (kg)	1.13	1.11	1.13	1.08	0.155	0.34
Power capability current pulses A						225

Figure 6.2-1 *Saft Lithium Ion Cell Specifications*[18]

For the thermal control system, a lumped mass analysis was performed. The extreme conditions for the lifetime of the lander will occur for the extreme hot during take-off, and extreme cold when the spacecraft landed on Triton. Based on the lumped analysis, the lander will be in sufficient temperatures for the extreme hot condition with a multilayer insulation. However, at Triton the lander will be below its acceptable operational and nonoperational lower limits. Therefore, the lander will require external heaters to keep the temperature of the lander to its acceptable temperature limits.

For the Command and Data Handling system, the total amount of data the scientific payload will take is at least 16 GB. The lander will have an onboard storage of 16 GB of data storage. The onboard processing unit will also take control of the spacecraft and receive all its commands from the main spacecraft and send its status information to the main spacecraft.

One of the crucial subsystem for the lander is the propulsion subsystem. Descending from 800 km above the surface of Triton, the propulsion system will be heavily reliant on the descent of the spacecraft. Calculating the descent velocity of 1150 m/s from 800 km, the total propellant for a safe landing on the surface of Triton is 48 kg. The total subsystemmass which includes the propellant tank, the engine, the valves, and other miscellaneous components will be 31 kg.

To control the descent of the lander, the attitude control system will be controlling the descent of the spacecraft. The attitude control system will be using thrusters to control the 3 axis rotations of the lander. The thrusters on Vespucci are the same as the one used on the TES.

For the structures subsystem, the lander will experience a total energy of 0.1 J of energy during landing. The amount of energy the lander will be experiencing at landing is negligible, our analysis for the lander's structure is based on the propulsive burns during the descent stage of the lander. The propulsion system loads the lander with 440 N of force. Thus, a circular structure made with 7075-Aluminum will be used. Thus, the total subsystemmass of the structure will be 50 kg. The total dry spacecraft mass is 110.8 kg. The lander mass statement is shown below as Table 6.3-1.

Table 6.2-1 *Lander Mass Statement*

Subsystem	Budget (kg)	Current (kg)
Power	15	29.29
Structures	21	21
Thermal Control	2	2
Propulsion	10	31
Command and Data	5	5
Attitude Control ACS	7	19
Telecom	5	5
Cabling	6	0
Payload	9	9
Budget Total	**80.00**	**120.87**
Dry Mass Margin	24.00	
On-Orbit Dry	**104.00**	**120.87**
Solid Propellant	30.00	63.2
Cold Gas Propellant	8.00	0.45
On-Orbit Wet	**142.00**	**184.50**

7 Project Management Plan and Schedule

7.1 Project Management Plan

In conjunction with the JPL Project Haukadalur RFP, the AIAA Space Launch System RFP served as the basis for our undergraduate senior design project. Similar to industry, the RFP provided set specific requirements and deadlines for the proposal and design. To meet said requirements and deadlines, a project management plan was implemented. The plan would incorporate the capabilities of the team and the deadlines for the RFP into a design schedule, which includes conceptual design, preliminary design, and detailed design. Conceptual design focused more on mission architectures, while preliminary and detailed design focused more on spacecraft design.

To effectively manage the workload of the project, a work breakdown structure was created to define the elements that were involved in spacecraft design from conceptual design to end of mission life shown below in Figure 7.1-1. Each design subsystem was assigned a lead based on their knowledge and experience prior. A deputy was also assigned under the lead to provide back-up and review completed work. One team member was given the role of systems engineer to implement proper measures that would prevent integration, manufacturing, or disposal issues. Such measures include reviewing the progress of each member during the weekly meetings and identifying any problems that may occur with other subsystems or further down the design timeline; communicating between subsystems to make sure future work is on the same page; thoroughly identifying potential project risks and introducing mitigation measures; and continuously checking back with the RFP to make sure all work is satisfying the requirements.

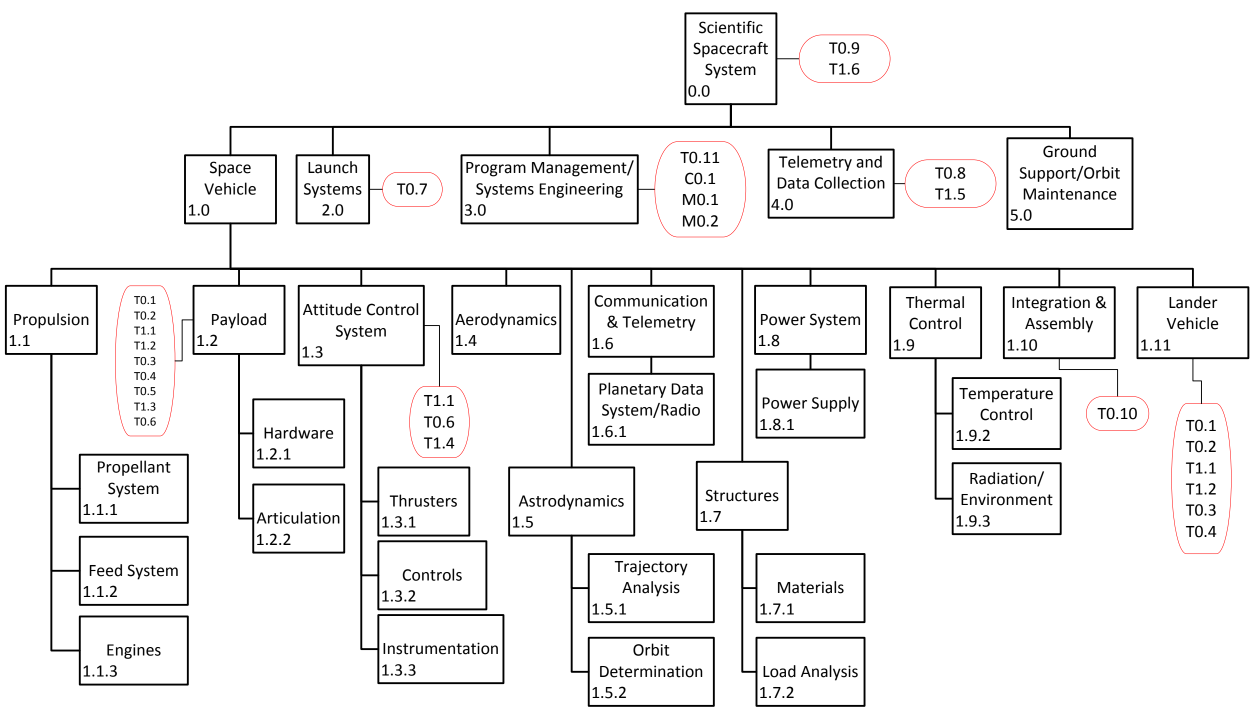


Figure 7.1-1 *Work Breakdown Structure with RFP Requirements Interface*

The proposal was broken up into development phases and placed into specific school quarters; conceptual design in fall quarter, preliminary design in winter quarter, and detailed design in spring quarter. Weekly meetings, both in class and at least one on the weekend, were chosen so that the team could communicate progress of individual work, present any issues that they had with the team to solve together, and make team decisions on important choices such as determining what payload items would be most efficient to the mission. Progress reviews were also incorporated into the weekly meetings to make sure everyone is staying on track with the schedule so that other team members aren't waiting on their work or the team isn't bombarded with catching up as the project report neared.

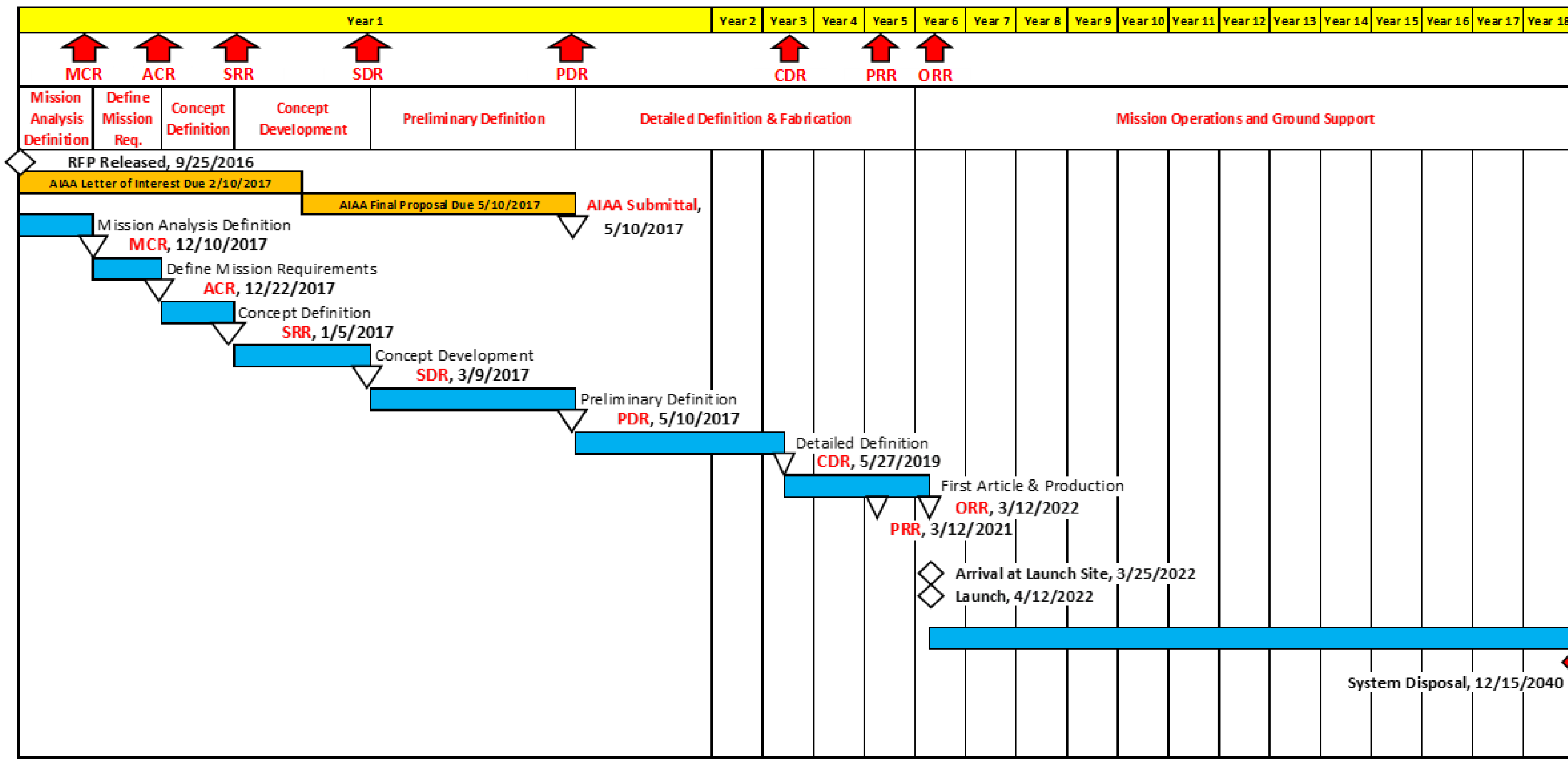
During winter quarter, the team was given the opportunity to present in front of faculty and industry from Northrop Grumman and Jet Propulsion Laboratories. The audience consisted of employees in the following fields: systems, structures, manufacturing, materials, communications and telemetry, and payload. Following the presentations, the audience graciously provided feedback on the designs. The presentations proved to be invaluable because it allowed the team to re-evaluate subsystem designs and find alternatives.

7.2 Project Schedule

The program management plan incorporates the conceptual, preliminary, and detailed design. The conceptual design has multiple reviews including mission concept review, alternative concept review, systems requirements review, and system design review. The preliminary design only includes the preliminary design review and the detailed design only includes the critical design review. The conceptual design incorporates more reviews because of the wider scope it must cover to ultimately down-select to one design, whereas the preliminary and detailed designs only focus on a single design.

The program reviews are defined by the large red arrows at the top to show how many reviews are in play during the mission program as well as the white downward facing triangles after each task to show which review is for which task. As seen in the schedule in Figure 7.2.1, the conceptual and preliminary design sections of the program are shorter than typically seen in industry because of our senior design course limited these tasks to a mere 3 months per. As for the remaining program tasks, due to strict trajectory limits, the detailed design was allocated 2 years, and manufacturing and testing was allocated 2 years.

Figure 7.2.1 Program Life Cycle Schedule incorporates everything from the RFP release to Mission Disposal



8 Assembly, Test, and Launch Operations

The ATLO process shall consist of a series of completing engineering data, manufacturing components and subsystems, testing components, integrating into a whole spacecraft, and preparing for launch. This process is projected to last approximately four years, in order to ensure adequate manufacturing and testing of all components prior to launching. Each step is briefly described in Table 8.1; the process is modeled after the third edition of "Space Mission Analysis and Design" (Chapter 12, Table 12-3).

Table 8.0-1 *Steps of Assembly, Testing, and Launch Operations*.[75]

Step	Description	Length of time
Prepare Engineering Data	Complete schematics with dimension, specifications, materials needed, processing methods	1-6 months
Manufacture Components	-Manufacturing planning -Procurement and testing -Assembly -Acceptance testing	1-6 months (parallel with engineering data prep) -Electronic parts: 3-18 mos. -Assembly: 1-3 mos.
Qualify Components	Functional testing and environmental exposure	1-6 months, depending on component and environment
Integrate, Prelaunch Test Spacecraft	Mechanical assembly, environment exposure, construction of spacecraft	6-18 months

Engineering data shall consist of schematics which include specifications of components/subsystems, as well as procedures on how to manufacture and integrate them. Each component shall have its own schematic, and various schematics will combine in order to show how they are to be manufactured, tested, and integrated with each other. Furthermore, the engineering data will show mounting and encapsulation procedures.

The manufacturing processes will involve raw materials from third party vendors, and different clean rooms shall be utilized to manufacture different categories of components (mechanical, inertial instruments, electronic, optical, etc.). While the mechanical manufacturing (plating, chemical treatment, composite manufacture adhesive bonding, etc.) will not require any controlled cleanliness, all other categories will require clean rooms ranging from Class 100 to 10,000. This stage (along with testing) is where quality assurance will be used to ensure all components are in compliance with engineering data. All failed pieces and anomalies will be kept in order to document repeated failures and rectify any design weaknesses.

Testing of components for this mission will include functional tests where components must be able to withstand adverse vibration, shock, and thermal conditions. Vibrations testing will simulate launch vehicle acoustics and engine rumble by giving random-signal frequencies ranging from 20 to 2,000 Hz. Shock tests will simulate payload fairing separation and spacecraft separation bolts, while thermal tests shall simulate the conditions the spacecraft will face. For example, any component that contributes to the lander must be able to function under temperatures as low as 35 K (-238°C). Functional tests for each subsystem will be conducted, such as static launch load tests, pattern tests for antennas, and closed-loop tests for attitude control systems.

The components shall be transported to the launch site for final spacecraft assembly. At the launch site, the spacecraft shall be inspected, propulsion systems tested, propellant loaded, mating spacecraft to the SLS Block 1B, and monitoring until the vehicle is prepared to launch.

9 Technology Readiness Level

Many of our components had to be addressed for TRL concerns. Subsystems including payload, propulsion, structure, attitude control, thermal, and power were analyzed to determine how much new technology was being implemented, and thus, how high the overall risk for our entire spacecraft system was. We used NASA's outlined definitions for our criteria in selecting the correct TRL for each component.[29] Almost all items in our subsystems were deemed to be a 9 on the TRL scale, meaning they were proven, ready to use, and reliable. Only three of our technologies have a TRL of 7, since they are still developing prototypes. Specifically, these were our composite tanks for propulsion (carbon fiber), our composite magnetometer boom, and our eMMRTGs. Our TRLs for all our major components are listed below in Table 9.0-1.

Table 9.0-1 *Tech Readiness Levels*

Subsystem	Technology	Technology Readiness Level
Payload	Science Payloads (collective)	9
Propulsion	Composite Tanks	7
Propulsion	Thrusters (AMBR)	9
Propulsion	Valves	9
Structure	Magnetometer Boom	7
ACS	Thrusters (MR-111C)	9
ACS	RWAs	9
ACS	Torque Rods	9
ACS	Sun Sensor	9
ACS	Star Trackers	9
ACS	IMUs	9
Thermal	Radiator	9
Thermal	Louvers	9
Thermal	RHUs	9
Power	MMRTG	9
Power	eMMRTG	7
Power	PDU	9

10 Mission Lifetime Assessment

As talked about in previous sections, the mission lifetime is rather long and encompasses almost two decades. Phase-1 of the mission from Earth to a Saturn flyby will last approximately 2 years and 326 days, Phase 2 from Saturn to Neptune 10 years and 130 days, Phase 3 from Neptune to Triton arrival 65 days, Phase 4 Triton arrival to end of operations 4 years and 106 days, and Phase 5 from end of operation to end of data delivery 1 year. In total, the mission Duration is 18 years and 262 days. Making this one of the longest sustained missions ever tackled by man. With mission length comes complications in sustainability, survivability, and operational assessment. The two main lifetime assessment criteria that were analyzed were the power subsystems as well as the propellant usage by the propulsion subsystem. As discussed within Section 5.5 (Power), four MMRTGs will be utilized to power the spacecraft. Previous MMRTGs utilized are: SNAP19 (40 W) used on Pioneer and Viking, MHW RTG (160 W) used on Voyager, and GPHA RTG (300 W) used on Galileo, Cassini, Ulysses, and New Horizons. These missions show that RTGs are a flight proven system that can be used for planetary and deep space travel especially since an RTG has never been reported as a failure in any space mission. However, RTGs are not perfect and the power output will degrade over time known as "Degradation". The causes associated with this power decline over time include fuel decay due to the radioactive decay of Pu238, thermoelectric degradation from bonds which increase resistance over time, and much more. These effects are not negligible and over time can vastly effect the power output on MMRTGs. It was found through extensive testing of real world RTGs that the fuel decay is 0.8%/year in which power drops by 1.1 % per year due to this fuel decay alone.[70] It was also found that thermoelectric degradation adds another 1.8% per year to fuel decay effects.[70] This means the total MMRTG power reductions are 2.9%/year due to fuel decay and TE degradation.[70] Figure 10.1-1 shows actual performance data from beginning of life of the power output effects over a 10 year period.

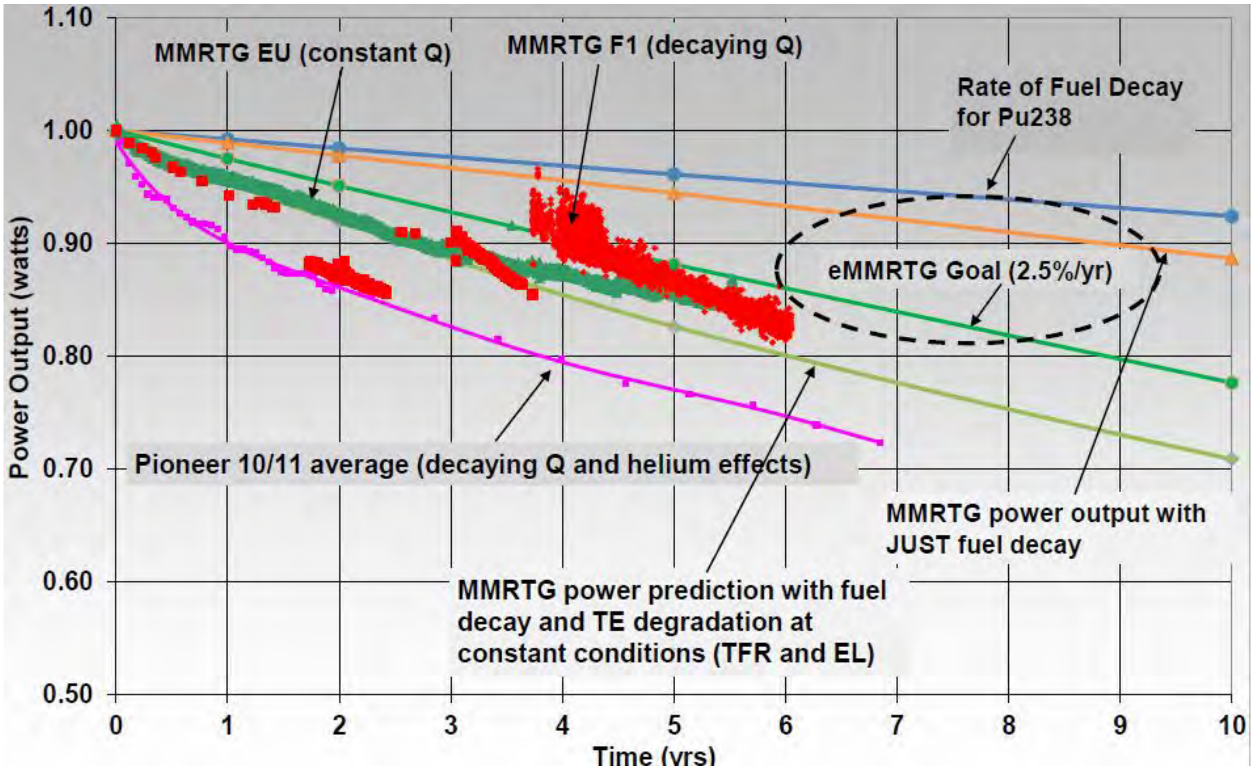


Figure 10.0-1 *Actual Performance Data Normalized to BOL* [1]

Next Generation MMRTGs are known as eMMRTGs and are expected to have a slightly better efficiency running at 2.5% per year. [70] The spacecraft will be utilizing two MMRTGs and two eMMRTGs due to the power supply restrictions that it may face at EOL. MMRTGs provide approximately 2,000 W of thermal power and 110 W of electrical power at BOL.[71] Since four are currently being used on the spacecraft, this brings the power supply at BOL to 440 W. Using the linear relation found by RTG degradation shown above, calculations were made to estimate the spacecraft's power from BOL to EOL. The results of these calculations can be seen in Figure 10.0-2 for transit and Figure 10.0-3 for operations.

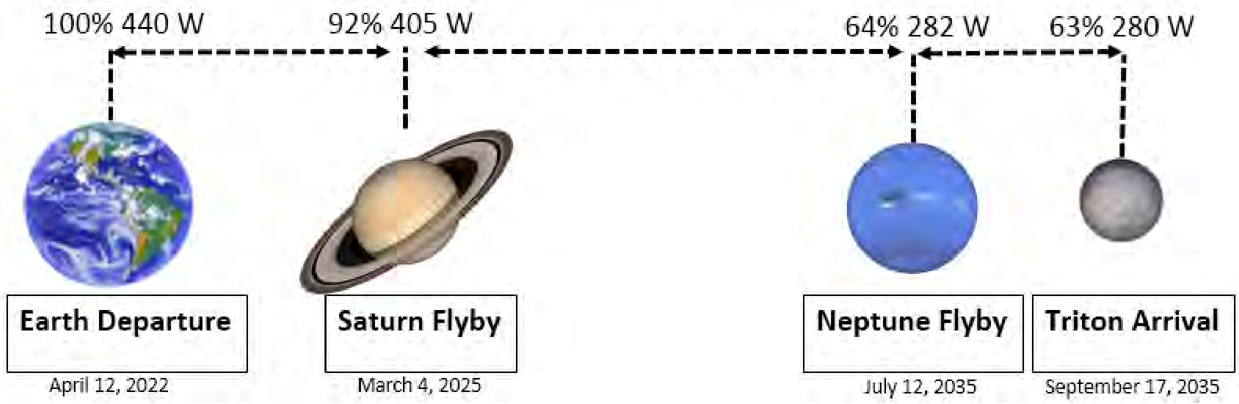


Figure 10.0-2 *Spacecraft Transit Power Degradation from BOL to EOL*

Figure 10.0-3 *Spacecraft Operations Power Degradation from BOL to EOL*

From the figures above, it can be seen that the power loss from the MMRTGs are large and at EOL the power output is under 50% of the original power output. This is an important aspect of the lifetime assessment and will play an important factor in power allocation and modes during spacecraft operations.

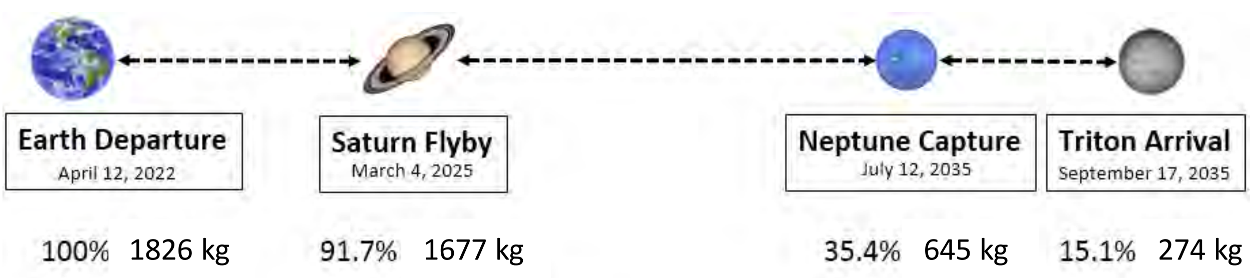
The spacecraft has different propulsion modes that allocate specific amounts of fuel per mode. Each mode has been discussed in detail in Section 5.4 (Propulsion), however these modes include: Corse Corrections Transit, ACS Transit, Neptune capture, Triton Capture, Station Keeping at Triton, ACS at Triton, and Contingency. Corse corrections are small burns necessary to stay on course during long distances, these are mainly to compensate for either propulsion inconsistencies or lift-off phase corrections. The ACS for transit will maintain the correct pointing either for the telecommunications or the thrusters for course correction. Upon arrival, Neptune requires a large burn in order to capture into its orbit, as well as Triton Capture within its orbit. Once at Triton, station keeping will be necessary to keep the spacecraft within its correct orbit; environmental disturbances cause the spacecraft to change orbits constantly which was discussed in greater detail in Section 5.1 (Attitude Determination and Control System).

The ACS will also be necessary to keep the correct pointing accuracy needed for the payload science operations and Telecommunications. Finally, a small contingency was also added within the propellant budget for Risk mitigation and a possible opportunity for mission extension; found in Section 13.0 (Risk Mitigation and Opportunity Management). Using Brown's estimation techniques propellant masses were found for every propulsive mode as shown in Table 10.0-1 [31].

Table 10.0-1 *Spacecraft Propulsion Allocations During Mission*

Course Corrections (kg)	ACS Transit (kg)	Neptune Capture (kg)	Triton Capture (kg)	Station Keeping (kg)	ACS at Triton (kg)	Contingency (kg)	Total (kg)
84	63.89	1032	371	154	20.46	100	1826

Using these values, it was possible to determine the exact amount of propellant that would be allocated during each phase of the mission. Figures 10.0-4 and 10.0-5 show the remaining propellant at each phase of the mission.



Figures 10.0-4 *Mission Lifetime Assessment of Propellant for Transit*

Figures 10.0-5 *Mission Lifetime Assessment of Propellant for Operations*

From this analysis, it can be seen that propellant is allocated for each phase of the mission and at EOL a small 5% contingency will remain within the tanks which will insure there will be adequate propellant within the tanks to complete each phase of the mission in its entirety, guard against any risk or phenomena which might require extra

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unplanned propellant usage, and possibly to extend the mission into the next Phase-6. With the necessary propellant and power outlined in the above section, it can be seen that the spacecraft has the necessary resources of power and propellant plus contingency to complete the mission efficiently and with full confidence in success.

11 Planetary Protection Protocols

In order to study Triton (or any solar system body) as safely as possible, planetary protection will be taken into high consideration for how the studies will be conducted. According to NASA's standards for planetary protection, there are five mission categories of protection—Category I requiring no documentation, and Category V requiring documentation that shows no destructive impact upon returning to Earth. The mission categories give the common planetary targets, as well as common mission types (flybys, orbiters, landers, etc.). Table 11.0-1 showing this data can be seen below.

Table 11.0-1 *Mission Categories (courtesy of NASA).*[4]

Planetary Targets/Locations	Mission Type	Mission Category
Undifferentiated, metamorphosed asteroids; Io; others TBD.	Flyby, Orbiter, Lander	I
Venus; Earth's Moon; Comets; non-Category I Asteroids; Jupiter; Jovian Satellites (except Io and Europa); Saturn; Saturnian Satellites (except Titan and Enceladus); Uranus; Uranian Satellites; Neptune; Neptunian Satellites (except Triton); Kuiper-Belt Objects (< 1/2 the size of Pluto); others TBD.	Flyby, Orbiter, Lander	II
Icy satellites, where there is a remote potential for contamination of the liquid-water environments, such as Ganymede (Jupiter); Titan (Saturn); Triton, Pluto and Charon (Neptune); others TBD.	Flyby, Orbiter, Lander	II*
Mars; Europa; Enceladus; others TBD (Categories IVa-c are for Mars).	Flyby, Orbiter	III
	Lander, Probe	IV(a-c)
Venus, Moon; others TBD: "unrestricted Earth return"	unrestricted Earth-Return	V (unrestricted)
Mars; Europa; Enceladus; others TBD: "restricted Earth return"	restricted Earth-Return	V (restricted)

According to this table, and in accordance with previous deep space missions such as Voyager, Cassini, and New Horizons, the two categories that stand out the most as candidates for how documentation will be provided are Category II and Category IV. Category II documentation is used for flybys, orbiters, and landers; this category encompasses most icy satellites where foreign spacecraft carry only a remote chance of contamination. For this category, very little documentation is required—a short plan of pre-launch and post-launch analyses is usually sufficient. It should also be noted that every deep space mission so far has been labeled as a Category II mission.

Category IV documentation, however, involves landers and probes; the only deep space moons that required this category are Europa and Enceladus, where scientific opinion has determined that contamination could compromise future missions.[4]

For the purposes of this mission, a Category IV approach will be taken. While Triton is considered an icy satellite, there is possible tidal activity occurring beneath its ice layers. The chemical composition of the moon will be analyzed, and there is a small possibility it can carry elements that support some form of life here. Therefore, there is enough scientific opinion and interest to keep Triton protected. Furthermore, for the pure sake of safety more documentation here will be preferred over less. The specific protocols involved with Category IV shall include a trajectory bias analysis, use of clean rooms, sterilization of the assembled spacecraft, and analyses for contamination and bioburdens.[4]

12 Cost Estimation Models

The RFP states that the mission must be completed with a budget of 5 billion in 2017 US dollars. The largest factors and volatility effecting the cost of this project are: the relatively small size of the spacecraft, the relatively long time for which the spacecraft must be supported, as well as the use of the new Space Launch System. Cost estimating relationships (CERs) were used in order to predict the potential cost and budget for this mission, and these relationships were provided and developed by NASA using data from previous missions to create a model to estimate cost. The models we used to predict project budget were the NICM, in addition to the USCM8.[76] The cost is broken down by each of the necessary subsystems required for a successful mission. The NICM uses the mass, power, and design life of the instruments to predict their individual costs. These are summed to show total instrument cost. The lander, its instruments, and costs have been estimated separately and then added to total cost of the mission. The rest of the components on the spacecraft use the USCM8 and the relationships are based on the dry spacecraft bus weight. All values have been scaled to show their cost in 2017 US dollars. Tables 12.0-1 shows the instrument costs for the orbiting and the landing spacecraft respectively.

Table 12.0-1 *TES Orbiter and Vespucci Lander Instrument Cost Estimations*

Hardware Costs (NICM Model)			
WBS Element	***Cost Drivers Relations***	***Cost Driver Values***	***CER MPE (FY 2010 $K)***
HRCC	M = Total Instrument Mass (kg)	2.5	14,608
	P = Max Instrument Power (W)	4	
	DL = Design Life (months)	300	
VIS	M = Total Instrument Mass (kg)	8	20,721
	P = Max Instrument Power (W)	15	
	DL = Design Life (months)	300	
UVS	M = Total Instrument Mass (kg)	5.5	12,088
	P = Max Instrument Power (W)	6	
	DL = Design Life (months)	300	
LIS	M = Total Instrument Mass (kg)	2	6,700
	P = Max Instrument Power (W)	5	
	DL = Design Life (months)	240	
RAT	M = Total Instrument Mass (kg)	4.0	3,908
	P = Max Instrument Power (W)	7.5	
	DR = Data Rate (kbps)	0.8	
	T = Instrument Tech Readiness Level (TRL 4-9)	9	
Triple-S	M = Total Instrument Mass (kg)	3.0	3,230
	P = Max Instrument Power (W)	5	
	DR = Data Rate (kbps)	1.08	
	T = Instrument Tech Readiness Level (TRL 4-9)	9	
MAGIC	M = Total Instrument Mass (kg)	3.0	6,627
	P = Max Instrument Power (W)	3	
	DL = Design Life (months)	300	
TDA	M = Total Instrument Mass (kg)	1.0	7,004
	P = Max Instrument Power (W)	3	
	DL = Design Life (months)	300	
LDA	M = Total Instrument Mass (kg)	1.5	7,602
	P = Max Instrument Power (W)	4	
	DL = Design Life (months)	240	
Cost Estimation Model Reference Year	2010	Total Instrument Cost (in 2010)	**27,470**
		Total Instrument Cost (in 2017)	**34,130**

Table 12.0-2 shows the cost of each of the separate subsystems for the landing vehicle. Table 12.0-3 to Table 12.0-6 details the cost of the software development, launch vehicle, ground operation, and DSN tracking 10% has been added to each mission category to account for the addition of the lander as part of the spacecraft system.

Table 12.0-2 *TES Orbiter and Vespucci Lander Spacecraft Cost Estimations*

Design, Manufacturing, Testing NON-RECURRING Cost (USCM8 Model)			
CER Category	***Cost Drivers Relations***	***Cost Driver Values***	***CER (FY 2010 $K)***
SC Bus	X1 = SC Mass (kg)	964	106,008
Structure & Thermal Control	X1 = Structure + Thermal Mass (kg)	212	25,203
Attitude Determination &	X1 = ADCS Mass (kg)	92	29,808
Power	X1 = EPS Mass (kg)	224	14,403
Propulsion (Reaction Control)	X1 = Total RCS Tank Volume (cc)	1335600	18,706
Telemetry, Tracking, & Command (TT&C)	Y = Average TT&C Cost (no statistical CER currently)	0	26,916
Communications Payload	X1 = Communications Subsystem Mass (kg)	124	76,632
Integration, Assembly, & Test (of bus & payload into SC) (IA&T)	X1 = SC Bus + Payload Nonrecurring Cost ($K)	106036	20,677
Program Level (other than Comm)	X1 = Space Vehicle + IA&T Nonrecurring Cost ($K)	297782	123,282
Aerospace Ground Equipment	X1 = SC Bus Nonrecurring Cost ($K) X2 = 0 (for Comm. Sats), X2 = 1 (for Non-Comm)	106008	37,124
Cost Estimation Model Reference (Year)	2010	Total Non-Recurring Cost (in 2010)	**478,759**
		Total Non-Recurring Cost (in 2017)	**594,842**

Table 12.0-3 *Software Development Cost Estimation*

Software Development Costs			
CER Category	***Cost Driver Relations***	***Cost Driver Values***	***CER (FY 2008 $K)***
Software Development	X1 = Lines of Code	5,000,000	100,000
Cost Estimation Model Reference (Year)	2008	Total Software Develop Cost (in 2008)	**100,000**
		Total Software Develop Cost (in 2017)	**132,197**

Table 12.0-4 *Launch Vehicle and Launch Operation Cost Estimation*

Launch Costs			
CER Category	***Cost Driver Relations***	***Cost Driver Values***	***CER (FY 2016 $K)***
Launch Vehicle	Y = Cost of Saturn V Production in 2016 ($K)	710,000	710,000
Launch Operations	Y = Cost of Saturn V Operations in 2016 ($K)	520,000	520,000
Cost Estimation Model Reference (Year)	2016	Total Launch Cost (in 2016)	**1,230,000**
		Total Launch Cost (in 2017)	**1,268,745**

Table 12.0-5 *Ground Operations Cost Estimation*

Ground Operation Costs			
CER Category	***Cost Driver Relations***	***Cost Driver Values***	***CER (FY 2008 $K)***
Ground Operations	X1 = Quantity of Personnel	50	328,500
	X2 = Rate per Personnel ($/day)	1,000	
	X3 = Duration of Operations (Years)	18	
Cost Estimation Model Ref. (Year)	2008	Total Ground Operations Cost (in 2008)	**328,500**
		Total Ground Operations Cost (in 2017)	**434,269**

Table 12.0-6 *Ground Tracking and DSN Usage Cost Estimation*

Ground Tracking/Telecomm Costs			
CER Category	***Cost Driver Relations***	***Cost Driver Values***	***CER (FY 2008 $K)***
DSN Hourly Usage	X1 = Hourly DSN Rate ($/hr for 2009)	1057	911,787
	X2 = Aperture Weighting	4	
	X3 = Number of Station Contacts	3	
	X4 = Number of Hours per Use	8	
	X5 = Number of Days for Mission Duration	22464	
Cost Estimation Model Ref. (Year)	2009	Total Ground Tracking Cost (in 2009)	**911,787**
		Total Ground Tracking Cost (in 2017)	**1,168,550**

Table 12.0-7 *Total Cost Estimation*

Triton Mission	
Cost Category	***Cost (in B$ 2017)***
Instruments	0.03
Spacecraft Development	0.59
Software	0.13
Launch Vehicle	1.27
Ground Operations	0.43
DSN Operations	1.17
Total	**3.633**

The cost of the SLS was based on NASA estimations as the project has not yet been completed. This allows room for fluctuation and error within the estimations of other systems within the spacecraft. The total estimated cost of the project is $3.63 billion, under the budgeted $5B given by NASA JPL in the RFP. When comparing to similar projects: Cassini cost $3.26B in 1997, New Horizons cost $700M in 2016, and Juno cost $1.1B in 2011.[72][73][74]

13 Risk Mitigation and Opportunity Management

13.1 Risk Mitigation

Project TES is a technically complex mission and with mission complexity comes uncertainty. In this section, multiple risks are identified that could potentially have a severe negative effect on our system.

Technical risks are risks that are related to the cruise, operations, or communications of the overall mission and are organized by cause and effect as shown below:

- Technical Risk # T0.1: “If ineffective analysis functions are programmed due to requirements creep, then the adverse consequence to the system will be inaccurate data analyses."
- Technical Risk # T0.2: “If propulsion system damage occurs due to micrometeorite collisions, then the adverse consequence to the system will be delay or failure to reach Triton by 2035 or possibly failure in overall mission.”
- Technical Risk # T0.3: “If the thermal regulator malfunctions due to fatigued wiring from cyclic heating, then the adverse consequence to the system will be spacecraft overheating leading to complete meltdown.”
- Technical Risk # T0.4: “If the propulsion system contaminates the lenses on the payload cameras, then the adverse effects will be ineffective data collection."
- Technical Risk # T0.5: “If the MMRTGs overheat the spacecraft, then the adverse effects will be damage and possible failure to payload sensors.”
- Technical Risk # T0.6: “If the main star trackers are damaged during operation, then the adverse effects would be to use the redundant systems as back-up tracking.”
- Technical Risk # T0.7: “If the momentum wheels are damaged or experience loss of power, then the adverse effects will be having to use the secondary propulsion system to control the attitude of the spacecraft, thus resulting in less pointing accuracy, as well as a shortening of mission life.”
- Technical Risk # T0.8: “If the scheduled data transmission appointment is cancelled due to the DSN re-allocating to another satellite for an emergency response, then the adverse effects will be less data transferred by end of the RFP deadline."
- Technical Risk # T0.9: “If the command and data system is overwhelmed due to too much incoming data, then the adverse effects will be deleted or corrupted data, the inability to transfer data by end of RFP deadline, or useless data being transmitted.”

- Technical Risk # T0.10: “If MMRTG degradation is more than estimated, then the advers e effects will be the inability to power certain subsystems within the spacecraft during the same times.”
- Technical Risk # T0.11: “If the structure of the spacecraft bus is damaged or incapable of handling max loads during launch due to manufacturing flaws, then the adverse effects will be damage to other subsystems and possible failure of the overall mission.”

The technical risks are separated by the likelihood of the risk happening and the consequence to the system if the event occurs. A risk cube for the technical risks is shown in Figure 13.1-1. High and medium risks are the most dangerous, and are analyzed with risk mitigation waterfalls to assess ways to avoid them. Due to page limitations, an example of a risk mitigation waterfall is shown in Figure 13.2-2.

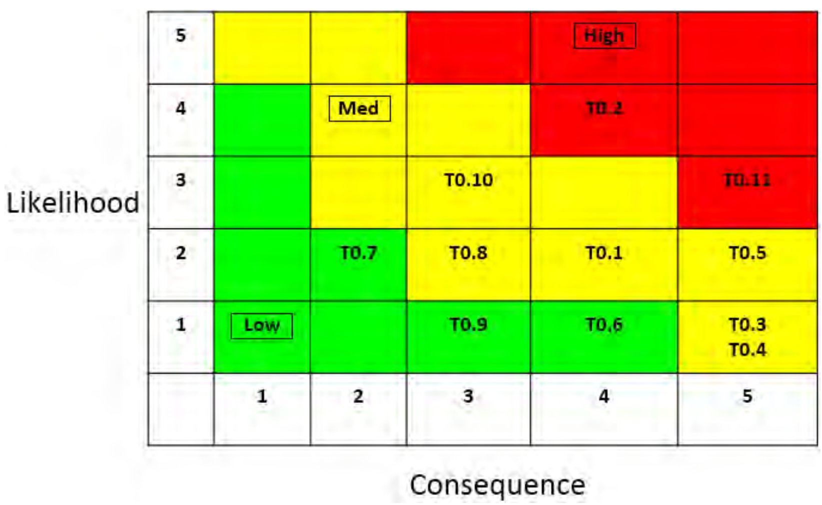

Figure 13.1-1 *Risk Mitigation Cube of Technical Risks*

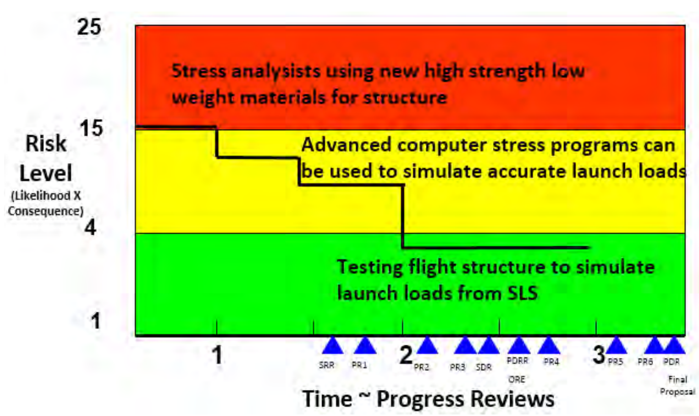

Figure 13.1-2 *Risk Mitigation Waterfall of Technical Risks*

Programmatic risks are risks that relate to the program, development, manufacturing, schedule, or cost that might affect mission program:

- Program Risk # P0.1: “If an SLS program delay occurs due to operational delays within the SLS supply chain, then the adverse consequence to the system will be a pushback of our launch date and potential Triton arrival date missed.”
- Program Risk # P0.2: “If the budget is exceeded due to unforeseen challenges, then the adverse consequence to the system will be a potential cancellation of the program.”
- Program Risk # P0.3: “If the program fails to meet the planetary protection protocols due to contamination during manufacture, then the adverse consequence to the system will be a delay in launch and potential missed launch date. “
- Program Risk # P0.4: “If the launch window for April 12, 2022 is missed due to longer development or manufacturing, then the adverse consequence to the system will be a delay in launch data that would push the start of the mission to May, 2024.”
- Program Risk # P0.5: “If bad launch weather prevents a launch on April 12, 2022, then the adverse consequence to the system will be a launch in the next 2 to 3 days after, and the margin of fuel in the spacecraft will be used to make up time during transit.”
- Program Risk # P0.6: “If a strike occurs at the manufacturing plant, then the adverse consequence to the system will be a project plan behind schedule."
- Program Risk # P0.7: “If the launch vehicle SLS is found to have problems at ULA, then the adverse consequence to the system will be a delay in launch until the problems are fixed.”
- Program Risk # P0.8: “If a delay in launch occurs, then the adverse consequence to the system will be a budget increase to compensate for the upset.”
- Program Risk # P0.9: “If an accident or unexpected event occurs during manufacturing of the satellite, then the adverse consequence to the system will be a budget increase as well as possibility of a delay in launch."
- Program Risk # P0.10: "If an accident occurs due to safety errors of hazardous materials such as the propellant, then the adverse effects will be fines and possible mitigation time being allocated to solve the problem."

The programmatic risk cube is shown below in Figure 13.1-3. Similar to the technical risks, the programmatic "Risk Mitigation Waterfall" is shown in Figure 13.1-4. Due to page limits, only one of the risk mitigation waterfalls is shown in Figure 13.1-4.

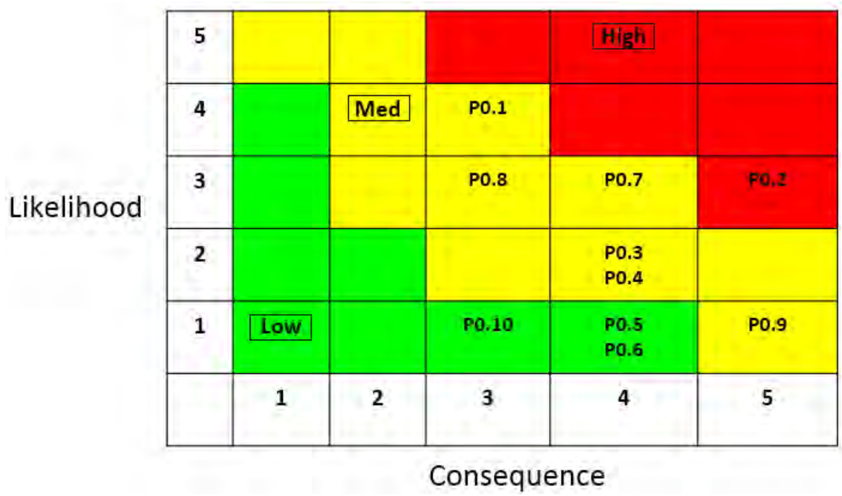

Figure 13.1-3 *Risk Mitigation Cube of Programmatic Risks*

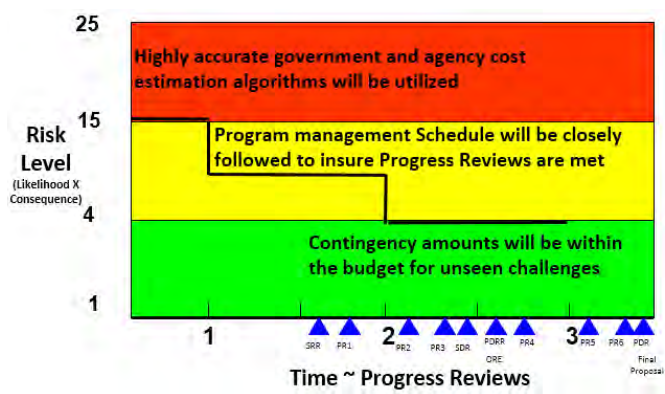

Figure 13.1-4 *Risk Mitigation Waterfall of Programmatic Risks*

13.2 Opportunity Management

Unlike risks, opportunities take advantage of certain circumstances that could arise during the development of our system. Instead of a catastrophe taking place from an unexpected event, a new and potential benefit can be created. In the same way a risk mitigation was performed, opportunity management seeks to identify opportunities from uncertain situations and to potentially increase the benefit from that opportunity with certain steps.

Technical opportunities relate to the cruise, operations, or communications of the overall mission and are organized by cause and effect as shown below:

- Technical Opportunity # T1.1: “If Plutonium-238 is acquired from US manufacturing firms, then the potential benefit to the system will be a safer, reliable, and long term fuel source for the spacecraft’s RTG”.
- Technical Opportunity # T1.2: “If a composite structure is implemented instead of traditional metal structures for the spacecraft bus, then the potential benefit to the system will be a lower cost and lower weight system”.
- Technical Opportunity #T1.3: “If an unexpected asteroid comes within observational distances to the spacecraft, the potential benefit to the system will be new and previously unknown data during the flyby”.
- Technical Opportunity #T1.4: “If an unexpected event occurs on the planet Saturn, the potential benefit to the system will be highly valuable data collected during the flyby”.
- Technical Opportunity #T1.5: “If there is leftover fuel and power reserves by end of operations, the potential benefit to the system will be mission extension into Phase-6”.
- Technical Opportunity #T1.6: “If the SLS launch vehicle provides more Delta-V then predicted, the potential benefit to the system will be possibly more fuel reserves by endo of the mission”.
- Technical Opportunity #T1.7: “If an ASRG has a major breakthrough and is available for the spacecraft by launch date, the potential benefit to the system will be a more efficient power supply and possible mass loss due to less MMRTGs”.
- Technical Opportunity #T1.8: “If SLS Block B is flight ready ahead of schedule, the potential benefit to the system will be less of a risk for budget and time scheduling”.
- Technical Opportunity #T1.9: “If a new fuel is created to use with the mission’s existing engines, the potential benefit to the system will be less mass for fuel and potential for more instrumentation on the payload”.

Technical opportunities are then separated by the likelihood of happening and the benefit to the system if the event occurs. Sorting opportunities in this way can show the low, medium, and high benefits of the system as shown by Figure 13.2-1 in the opportunity management cube. Low and medium benefits are then analyzed in an opportunity management fish ladder to increase the opportunity from a low benefit to a higher benefit of the system. Due to page limits, only one fish ladder is shown in Figure 13.2-2, but ideally more would be implemented in the system level design.

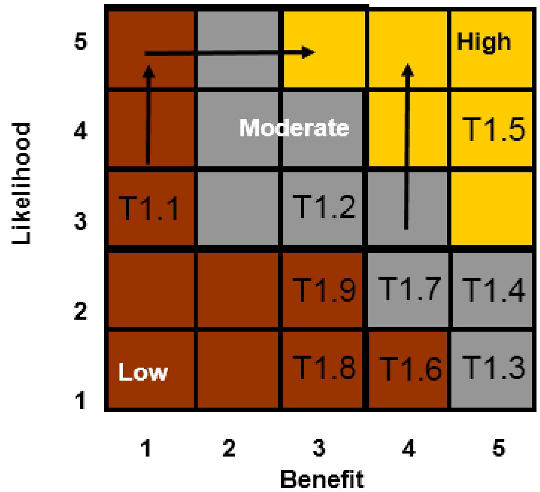

Figure 13.2-1 *Opportunity Management Cube for Technical Opportunities*

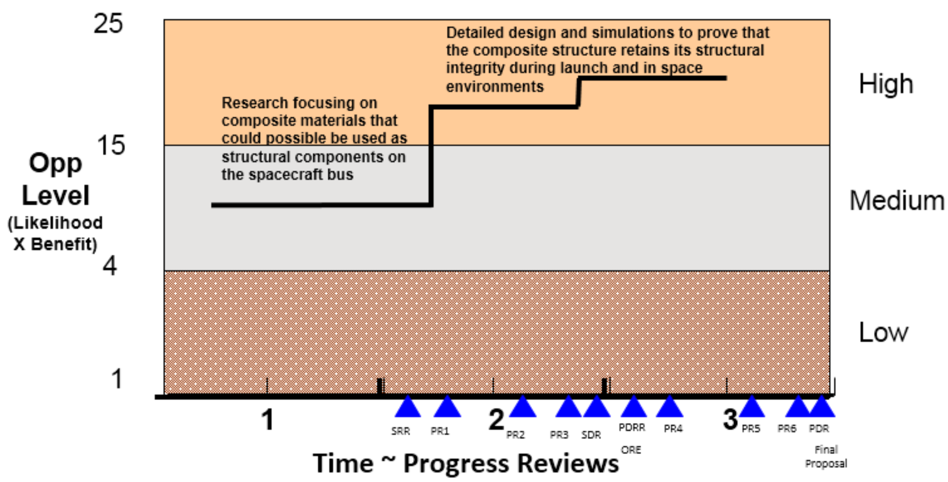

Figure 13.2-2 *Fish Ladder for Technical Opportunities*

Programmatic opportunities relate to the program, development, manufacturing, schedule, cost, or anything that might affect mission program and are organized by cause and effect as shown below:

- Programmatic Opportunity #P1.1: "If the project schedule is ahead of planned dates, the potential benefit to the system will be reduced cost of the overall project".
- Programmatic Opportunity #P1.2: "If the spacecraft cost EVMS is less than predicted, the potential benefit to the system will be a reduced cost for the customer or possible contingency budget".
- Programmatic Opportunity #P1.3: "If moral is high in the workforce, the potential benefit to the system will be a reduced cost and more time in the schedule".

- Programmatic Opportunity #P1.4: "If a new manufacturing process is made available for composite structures, the potential benefit to the system will be reduced cost in manufacturing spacecraft composite structures".
- Programmatic Opportunity #P1.5: "If a manufacturing process is outsourced to a more cost effective source, the potential benefit to the system will be a reduced cost for the program".
- Programmatic Opportunity #P1.6: "If a surge of new interns are hired to work on mundane work, the potential benefit to the system will be more time for better analysis of senior engineers".
- Programmatic Opportunity #P1.7: "If a more effective project management plan is implemented, the potential benefit to the system will be lower cost and time for the system development and manufacturing".
- Programmatic Opportunity #P1.8: "If the DSN can handle more than 8-hours a day for data communication, the potential benefit to the system will be more scientific data transmitted from Triton to Earth outside of the RFP guidelines".

Programmatic opportunities are then separated by the likelihood of happening and the benefit to the system if the event occurs. Sorting opportunities in this way can show the low, medium, and high benefits of the system as shown by Figure 13.2-3 in the opportunity management cube. Low and medium benefits are then analyzed in an opportunity management fish ladder to increase the opportunity from a low benefit to a higher benefit of the system. Due to page limits, only one Fish Ladder is shown in figure 13.2-4 but ideally more would be implemented in the system level design.

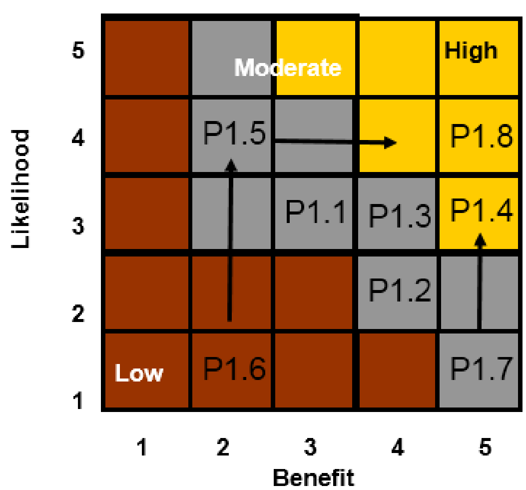

Figure 13.2-3 *Opportunity Management Cube for Programmatic Opportunities*

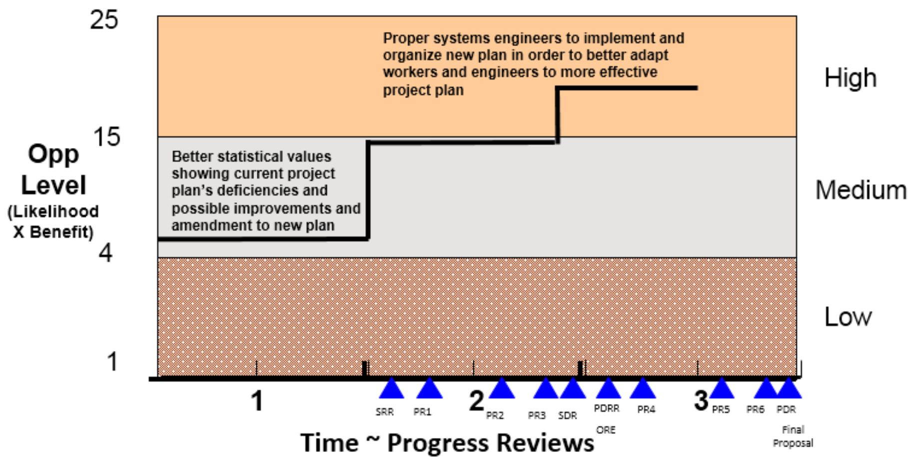

Figure 13.2-4 *Fish Ladder for Programmatic Opportunities*

14 Compliance Matrix

There are exactly 21 requirements listed in our compliance matrix that were obtained from our two RFPs from JPL and AIAA. Of the 21 requirements, 16 are primary requirements and 5 are derived requirements. A compliance matrix containing all technical requirements is shown in Table 14.0-1. A compliance matrix for all programmatic requirements is shown in Table 14.0-2. The requirements are specified as either required or optional, based on what was stated in the RFP. For all compliant requirements, a short description is given as to how the requirement is met. In addition, for technical requirements, the payload item which completes the requirement is given, and the method in which it does so is detailed. Lastly, the major discipline for each requirement is listed below, along with their respective requirement numbers which connect back to the WBS.

Table 14.0-1 *Technical Compliance Matrix*

RFP	Requirement Number	Requirement	WBS Number	Required/ Optional	Major Discipline	Completion Description	Compliance	Method
JPL	T0.1	Provide data for mineral, surface history, and thermal mapping of Triton's geyser zone with a resolution of 10 m	1.2	Required	Payload	Our instruments, including HRCC, VIS, Triple-S, and LIS, meet this requirement.	Compliant	Remote, Direct
JPL	T0.1.1	Full surface mapping at 1 m resolution	1.2	Optional	Payload	-	Non-Compliant	-
JPL	T0.2	Determine composition of the geyser zone surface in a clean area not covered by geyser precipitation	1.2	Required	Payload	Our instruments, including HRCC, VIS, LDA, Triple-S, and LIS, meet this requirement.	Compliant	Remote, Direct
JPL	T1.1	System shall be capable of differentiating between areas of the geyser zone that are covered or not covered by geyser precipitation	1.2	Required	Payload	Our instruments, including HRCC, VIS, LDA, Triple-S, and LIS, meet this requirement.	Compliant	Remote, Direct
JPL	T1.2	System shall be capable of analyzing surface soils	1.2	Required	Payload	Our instruments, including HRCC, VIS, LDA, Triple-S, and LIS, meet this requirement.	Compliant	Remote, Direct
JPL	T0.3	Determine the composition, particle size, and particle volume density of the solid material released by Triton's geysers	1.2	Required	Payload	Our instruments, including VIS, Triple-S, and LIS, meet this requirement.	Compliant	Remote, Direct
JPL	T0.4	Determine the composition of the geyser-driving exhaust and its pressure and volume density in the eruption plume	1.2	Required	Payload	Our instruments, including UVS and LDA, meet this requirement.	Compliant	Remote, Direct
JPL	T0.5	Determine the composition, temperature, pressure, and density of Triton's atmosphere from 20 km above its thermopause to the surface at 100 m intervals in altitude	1.2	Required	Payload	Our instruments, including VIS, UVS, TDA, and RAT, meet this requirement.	Compliant	Remote, Direct
JPL	T0.5.1	Map atmosphere 100 km above the surface at 10 m intervals	1.2	Optional	Payload	-	Non-Compliant	-
JPL	T1.3	System shall have the capability to analyze the composition of a trace atmosphere	1.2	Required	Payload	Our instruments, including VIS, UVS, TDA, and RAT, meet this requirement.	Compliant	Remote, Direct
JPL	T0.6	Atmospheric measurements shall be made 75 km upwind of any geysers in the region or 40 km crosswind and outside the periphery of the geyser region	1.2	Required	Payload	Our instruments, including VIS, UVS, TDA, and RAT, meet this requirement.	Compliant	Remote, Direct
JPL	T1.4	System shall be capable of orienting its instrumentation to avoid interference by geyser plumes	1.3.2	Required	Avionics	Our ACS system allows for accurate pointing.	Compliant	Remote
JPL	T0.9	System must be capable of performing science operations prior to arrival at Triton	0.0	Required	Payload	Our ground and science operations begin prior to Triton arrival.	Compliant	-

Figure 14.0-2 *Programmatic Compliance Matrix*

RFP	Requirement Number	Requirement	WBS Number	Required/ Optional	Major Discipline	Description	Compliance
JPL	T0.7	Arrive at Triton by December 2035, completion of operations before December 2039 and data delivery by December 2040	2.0	Required	Launch System	Our launch date of April 12, 2022 and our Saturn flyby trajectory allows us to arrive at Triton by December 2035.	Compliant
JPL	T0.8	All scientific data shall be delivered to the Planetary Data System	1.5	Required	Telecom	By using the DSN, we will deliver all data to the PDS.	Compliant
JPL	T1.5	System must be capable of interfacing with the Deep Space Network	1.5	Required	Telecom	Our telecommunications communicate with the DSN.	Compliant
JPL	T1.6	Spacecraft shall be capable of using its instruments in deep interplanetary space	0.0	Required	Spacecraft System	Ground and science operations will begin prior to Triton arrival.	Compliant
AIAA	T0.10	System must be capable of launching aboard a currently available launch vehicle including SLS	1.8	Required	Integration & Assembly	Our system uses SLS Block 1B as its launch vehicle.	Compliant
JPL	T0.11	System shall abide by NASA's Planetary Protection Protocols	3.0	Required	Systems Engineering	Both our spacecraft will neutralize propellant, not contaminate Triton, and abide by all Planetary Protection Protocols.	Compliant
JPL	C0.1	Project cost cap is $5 billion	3.0	Required	Program Management	Our system costs $3.63 billion total.	Compliant
JPL	M0.1	Conceptual Design Review is due the last half of March 2017	3.0	Required	Program Management	Conceptual Design Review was submitted March 2017.	Compliant
JPL/ AIAA	M0.2	Preliminary Design Review is due the last half/beginning of May 2017	3.0	Required	Program Management	Preliminary Design Review will be submitted May 2017.	Compliant

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