AIAA Team Space Transportation Design Competition

Triton Atmosphere and Geyser Orbital Surveyor

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Executive Summary

With the advent of the National Aeronautics and Space Administration's (NASA) Space Launch System (SLS), capabilities of space exploration are soon to be pushed beyond their current limits. With its completion, the SLS will allow for payloads with larger masses and volumes to be launched on higher energy trajectories than previously possible. This proposal presents a mission that makes use of these new possibilities to pursue exploration into the outer reaches of the solar system.

The mission discussed in this proposal is the Triton Atmosphere and Geyser Orbital Surveyor (TAGOS). The concept of this mission is obtained from Jet Propulsion Laboratory's "Project Haukadalur" request for proposal (RFP). The RFP presents the need for an investigation of Neptune's largest satellite, Triton. Throughout the history of space travel, Voyager 2 has been the only craft to visit Neptune. As Voyager 2 performed its Neptune flyby, it captured images of erupting geysers on the surface of Triton. This raised many questions regarding the composition of the geyser exhaust and the atmospheric properties of Triton. To discover the answers to these questions, Jet Propulsion Laboratory (JPL) has requested the retrieval of data regarding Triton's atmosphere, surface, and geysers found on the Triton.

The goal of the TAGOS mission is to utilize the capabilities of NASA's SLS by launching a spacecraft to Triton to fulfill the requirements set forth by JPL's RFP. The mission includes geological mapping as well the analysis of its atmosphere and geyser plumes. This proposal details all aspects of the TAGOS mission, including requirements, mission architecture, science operations, trajectories, and vehicle subsystem design, as well as management components such as scheduling and budgeting. Key trade studies on selected mission components are included to justify all design decisions. All decisions were made to ensure compliance with all RFP requirements while minimizing the risks associated with the mission.

The TAGOS mission is made possible by NASA's SLS. Due to the Neptune arrival date in the year 2035 required by the RFP, a high energy trajectory is required to meet the deadline. In addition to this, many challenges must be overcome such as communication with Earth over such great distance and dealing with the extremely low temperatures experienced while in the Neptunian system. These challenges all require increased mass budgets leading to numerous engineering decisions. This mission will make full use of the SLS capabilities while providing valuable information on Neptune's largest satellite, Triton.

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List of Symbols and Notation

Symbol or Acronym	Definition
ACS	Attitude Control System
ASRG	Advanced Stirling Radioisotope Generator
BWG	Beam-Waveguide
C&DH	Command and Data Handling
C3	Characteristic Energy
CCSDS	Consultative Committee for Space Data Systems
ConOps	Concept of Operations
COPV	Composite Overwrapped Pressure Vessels
dB	Decibel
deg or °	Degree
DM	Dual Mode
DRM	Data Return Module
DSN	Deep Space Network
E_b/N_0	Energy per bit to noise power spectral density ratio
EIRP	Equivalent Isotropically Radiated Power
EOL	End of Life
Eq	Equation
Ex	Example
g_0	Standard Gravity due to Acceleration on Earth
HAIL MARY	Helio-Assisted Intel Launch for Mission Asset RecoverY
He	Helium
HGA	High Gain Antenna
HiPAT	High Performance Liquid Apogee Thruster
Hz	Hertz
IMU	Inertial Measurement Unit
IR	Infrared
I_{sp}	Specific Impulse
J	Joule
JPL	Jet Propulsion Laboratory
km/s	Kilometers per second
ksps	Kilosymbols per second
L	Liter
LEO	Low Earth Orbit
LGA	Low Gain Antenna
LV	Launch Vehicle
m	Meter
MIPS	Million Instructions per second
MLI	Multi-Layer Insulation
MLI	Multi-Layer Insulation
MMRTG	Multi-Mission Radioisotope Thermoelectric Generator
N	Newton
N_2H_4	Hydrazine
NASA	National Aeronautics and Space Administration
NICM	NASA Instrument Cost Model
N-s	Newton second
NTO	Nitrogen Tetroxide
Org.	Organization
Pa	Pascal
PAF	Payload Attachment Fitting
pg.	Page

PMD	Propellant Management Devices
RFP	Request for Proposal
RHU	Radioisotope Heater Unit
ROM	Rough Order of Magnitude
RTG	Radioisotope Thermoelectric Generator
RTG	Radioisotope Thermoelectric Generator
s	Seconds
SLS	Space Launch System
SOI	Sphere of Influence
SSPA	Solid State power amplifiers
SSR	Solid State Recorder
STDN	Spacecraft Tracking and Data Acquisition Network
TAGOS	Triton Atmosphere and Geyser Orbital Surveyor
TDRSS	Tracking and Data Relay Satellite System
Telecom	Telecommunication
TIS	Triton Imaging System
TOF	Time of Flight
TRL	Technology Readiness Level
TT&C	Telemetry, Tracking and Command
TWTA	Traveling-Wave Tube Amplifier
USCM8	Unmanned Spacecraft Cost Model Version 8.0
$V_{\infty,L}$	V_∞ at Earth launch.
Vol	Volume
vs.	versus
W	Watt
WBS	Work Breakdown Structure
ΔV	Change in velocity
ΔV_A	Change in velocity at Neptune arrival
ρ	Density

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1 Mission Overview

The TAGOS mission will investigate geysers on the surface of Triton, the largest moon of Neptune. The TAGOS mission will leverage NASA's SLS being developed to launch a large spacecraft. The mission will attempt to gather high resolution data of these elements of Triton: the surface, geyser exhaust, and atmosphere. On the surface, geological (mineral, surface history, and thermal) mapping and the surface composition data is needed. For the geyser exhaust, data is required of the composition, particle size, and particle volume density in three areas: in a plume, in an eruption cloud, and on the surface. In the atmosphere, data on the composition, temperature, pressure, and density of the atmosphere at different altitudes is needed. This data is to be sent back to Earth via the Deep Space Network (DSN). The mission will use an orbiter to get the data of Triton's surface. The mission will also use a probe to collect the data of Triton's atmosphere and another probe to collect the data for the geyser exhaust. The mission will also attempt to do a secondary data return phase to travel closer to Earth to transfer even higher resolution data.

1.1 Requirements

The TAGOS mission's requirements come from both the JPL and AIAA RFPs. The JPL RFP contains requirements regarding the exploration of Triton, especially of the geysers, the timeline for the mission, and the cost of the mission. Appendix A contains the full RFP from JPL. The AIAA RFP contains broad requirements for a space mission beyond Earth and Lunar orbits and also requires that no more than two SLS Block 1B launches shall be used for the mission.

Table 1-1 Mission Requirements

Requirement Number	RFP Para. Number	Description
1	5 (JPL)	Provide geological mapping of geyser zone area with 10 m resolution (Goal: Full surface mapping at 1 m resolution)
2	6 (JPL)	Determine composition of surface in area not covered by geyser precipitation
3	7 (JPL)	Determine the composition, particle size, and particle volume density of the entrained solid material released by Triton's geysers in a plum, in an eruption cloud, and on the surface
4	8 (JPL)	Determine composition of geyser driving exhaust
5	9 (JPL)	Determine composition, temperature, pressure and density of Triton's atmosphere from 20 km above its thermopause to the surface at 100 m intervals (Goal: from 100 km above surface at 10 m intervals)
6	10 (JPL)	Shall arrive at Triton by December 2035, complete operations by December 2039 and deliver data before December 2040
7	11 (JPL)	Spacecraft must be able to sustain cruise science operations prior to arrival
8	12 (JPL)	Project must cost less than $5 Billion
9	19 (AIAA)	Project shall utilize up to two SLS Block 1B launches

1.2 Architecture Development

There were two functional architectures considered by the team to fulfill the requirements set forth by the AIAA and JPL RFPs. Every requirement given by the RFPs affected the architecture designs and throughout the process the architectures went through continuous alterations as more data was obtained through both research and calculations. Ultimately the team decided to go with the Stryker-1 architecture for this mission.

In addition to the chosen Stryker-1 architecture, an additional architecture (Stryker-2) was considered. Both architectures contained many similarities but differed primarily in the arrival at the Neptunian system. While Stryker-1 utilizes its propulsion system to enter into an orbit around Neptune, Stryker-2 utilized an aerocapture maneuver. To accomplish this, the Stryker-2 orbiter was designed using the NASA Hypersonic Inflatable Atmospheric Decelerator (HIAD). This is a deployable heat shield that would allow the Stryker-2 to withstand the extreme temperatures of the maneuver. The use of the aerocapture maneuver greatly reduced the overall mass of the orbiter due to not requiring a large amount of propellant for arrival. However, upon further analysis, it was found that both architectures were within the mass limit to launch on the trajectory to Neptune. It was decided that the aerocapture maneuver would only add unnecessary risk and the TRL of the HIAD was too low to rely on. For these reasons, Stryker-1 was selected for the TAGOS mission.

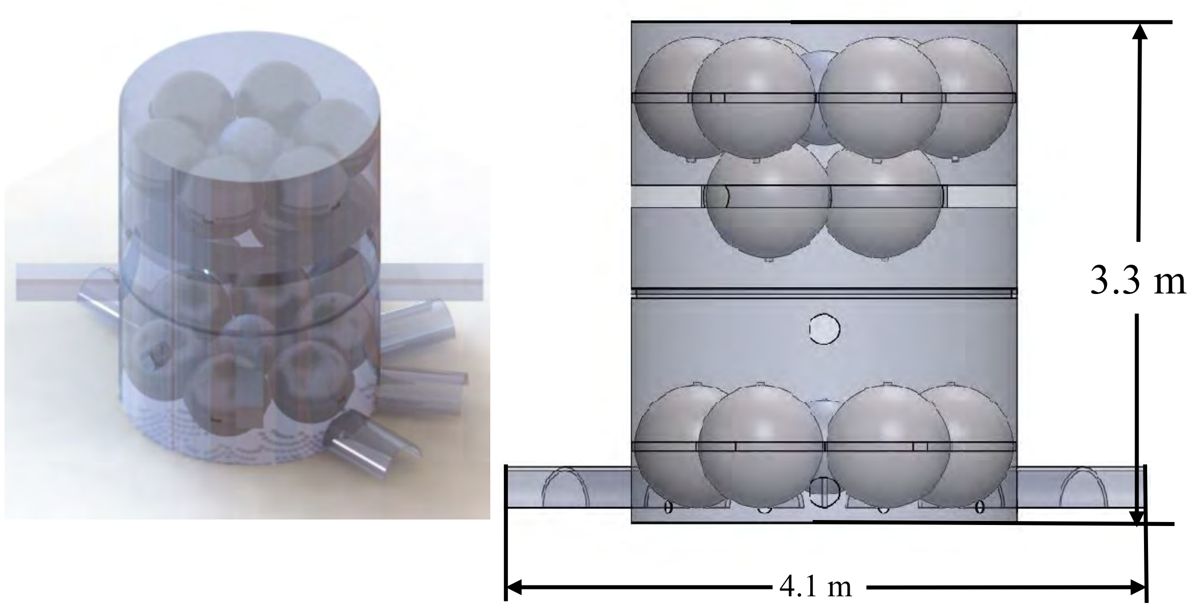


Figure 1-1 HCP Layout for Interior Fuel Tanks on Stryker-1 Orbiter

The Stryker-1 architecture is an orbiter which carries a separable Atmosphere Probe and Geyser Probe to fulfill all the scientific requirements. A Centaur upper stage launch vehicle is included in the design to satisfy the time requirement set forth by the JPL RFP; however, this greatly reduces the space left in the SLS payload fairing for the Stryker-1 orbiter. The orbiter was designed to have interior fuel tanks small enough to utilize hexagonal close-packed (HCP) lattices shown in Figure 1-1 above. Smaller tanks and HCP packing both decrease the total height of the Stryker-1 orbiter but increase the total weight of the propulsion system. The smaller tanks and HCP packing were chosen because it was decided the added mass to the orbiter was worth the smaller total height.

Data rates to send scientific data from Triton back to Earth were also found to be a challenge. Calculations showed that when considering the extreme range from Triton to Earth, the resulting data rates were unacceptably slow, and that it would take up to 40 years to send all the data back while in orbit around Triton. To solve this issue, it was decided the Stryker-1 orbiter would start a journey back towards the Sun at the end of its Triton mission to move closer to Earth and hence decrease the time required to transfer all the scientific data. The Stryker-1 orbiter is designed to have two sections as shown in Figure 1-2 below; the bottom section will have all the equipment needed throughout the entire mission while top section will have all the equipment that stops serving a purpose once the mission at Triton is completed.

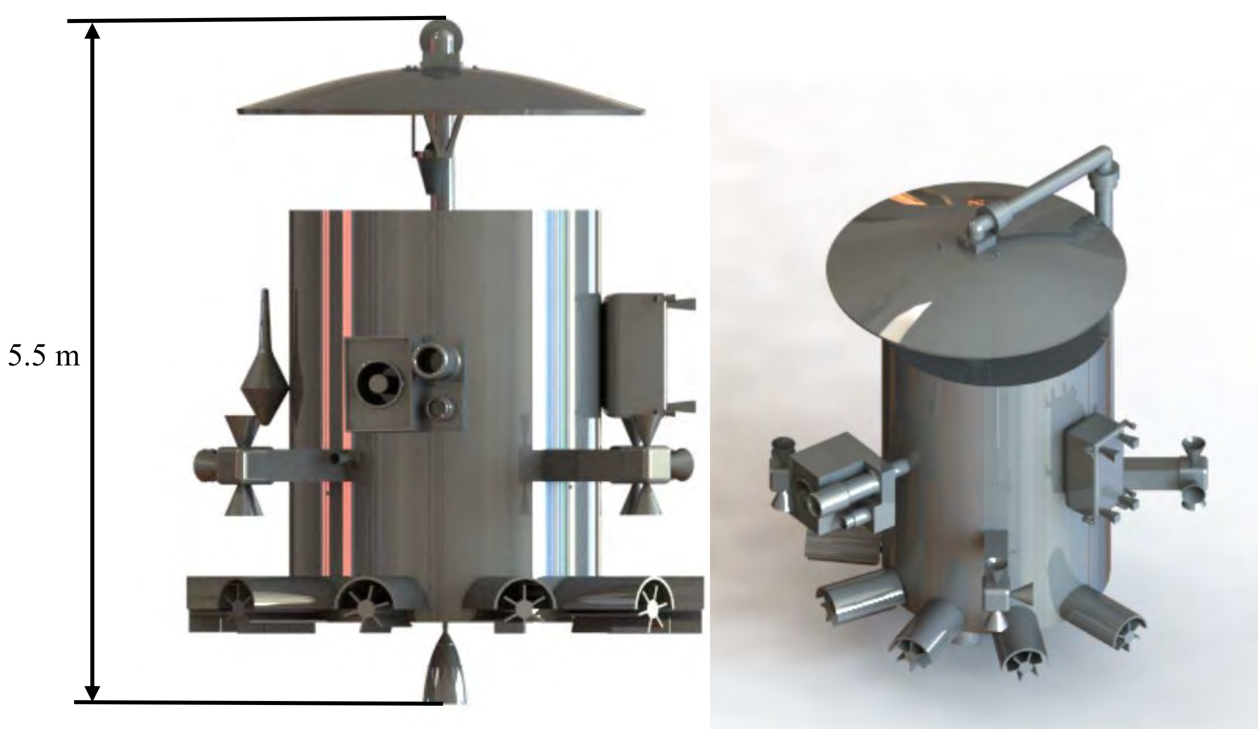


Figure 1-2 Assembled Architecture of Stryker-1 Orbiter with Probes Attached

To improve communication with Earth as well as effectively decrease the volume of Stryker-1 during launch, the Stryker-1 antenna is mounted on an extendable boom. This boom is attached with two gimbals at the point of contact with the orbiter, with an additional two at the opposite end where the antenna is located. This allows for the antenna to be retracted during launch, saving space in the payload fairing. After launch, the gimbals on both ends of the boom allow for the antenna to be articulated to continuously point in the direction of Earth.

In order to power all the equipment to complete the mission the team chose to use six Multi-Mission Radioisotope Thermoelectric Generators (MMRTGs). These were decided to be placed on the bottom half of the orbiter towards the bottom because we needed them throughout the mission. This also affected the ACS thruster placements to be in such a way that the exhaust of the thrusters would not come in contact with the MMRTGs. To achieve this the ACS thrusters were placed a small distance away from the body of the orbiter as shown in Figure 1-2 above.

Overall the Styker-1 architecture equipment was specifically chosen to include as many proven technologies as possible in order to decrease the risk involved with completing the mission. This was done by utilizing equipment with high TRL levels that would decrease the possibility of failure per part within the design.

2 Science

2.1 Science Overview

The scientific data requirements for this mission were split into three major categories: mapping requirements, atmospheric sampling, and geyser zone surveying. Because of the needs to obtain atmospheric data in a geyser-free zone as well as geyser plume data, two different probes were added to the Stryker-1 orbiter. The breakdown of data gathering is listed in Table 2- below. This table displays all scientific requirements as well as the spacecraft elements that fulfill them.

Table 2-1 Science Requirement Breakdown

Requirement	Orbiter	Atmosphere Probe	Geyser Probe
Provide geological mapping of geyser zone area with 10 m resolution (Goal: Full surface mapping at 1 m resolution)	✓		
Determine composition of surface in area not covered by geyser precipitation	✓		
Determine the composition, particle size, and particle volume density of the entrained solid material released by Triton's geysers in a plume, in an eruption cloud, and on the surface			✓
Determine composition of geyser driving exhaust			✓
Determine composition, temperature, pressure and density of Triton's atmosphere from 20 km above its thermopause to the surface at 100 m intervals (Goal: from 100 km above surface at 10 m intervals)		✓	

The Stryker-1 orbiter is responsible for the geological mapping the surface as it orbits Triton. After approximately 2 years when the surface of Triton has been fully mapped by the orbiter, the first probe to deploy is the Atmosphere Probe. This probe is launched from the orbiter and rapidly descends through the atmosphere of Triton obtaining data for density, temperature, and pressure. This data is transmitted to the orbiter before a hard landing on the surface of Triton. After the atmospheric data has been retrieved, this information is used to plan the descent of the Geyser Probe. This probe features four gimballed thrusters used to control its descent as it passes through a geyser plume, collecting samples of the solid materials from the geyser. After the probe has landed on the surface of Triton, it will remain active for 24 hours obtaining more samples and transmitting the data to the orbiter.

2.2 Vehicle Instrumentation

2.2.1 Orbiter Instrumentation

The objective of the Stryker-1 orbiter is to complete the geological mapping of Triton's surface. This will be accomplished by the Triton Imaging System (TIS). The TIS consists of two separate instruments that will obtain visual, composition, and infra-red mapping data as Stryker-1 orbits Triton.

The first instrument utilized on the orbiter is the Main Space Science Systems ECAM-C50 narrow angle camera. This instrument is responsible for the visual mapping of Triton. A narrow angle camera was chosen for a smaller field of view cone, meaning less area is captured allowing for a higher resolution. The selected camera has a resolution of 5 megapixels and a field of view (FOV) of 0.7 degrees. These values were used determine the required orbit height to obtain the goal resolution. These calculations were based on the curved planet surface area as shown in Figure 2-1 below. It was determined that at an orbital altitude of 90 km, a total area of 4.96 km^2 would be captured. With a total pixel count of 5,151,600, this gives a resolution of 0.96 m/pixel, meeting the surface mapping goal.

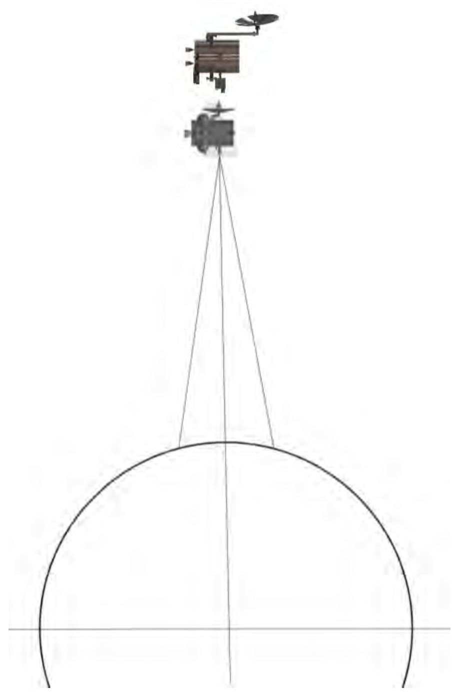


Figure 2-1 Visual Instrument Mapping Diagram

The second instrument on the orbiter is the Visible and Infrared Mapping Spectrometer (VIMS). This instrument analyzes the different wavelengths of incoming light to determine the composition of the source. This will be used to analyze the composition of the surface of Triton as well as the atmosphere. The reason for choosing this instrument as opposed to other options is that it is remote sensing, meaning that the Stryker-1 orbiter can obtain all composition data without having to physically be in contact with any materials. Due to these instruments not being commercially available, estimates for mass and power were made using the Cassini spacecraft as a reference.

To accurately image the surface of Triton, the TIS must face the area to be imaged while the orbit progresses for up to 3 seconds while the picture is being taken. Because during design it was determined that it would require too much power and mass for the Attitude Control System (ACS) to accurately reposition the entire orbiter for each image without blurring, the addition of a scanning platform was deemed necessary. This platform, referred to as the Visual Instrument Platform (VIP), moves independently of the orbiter using two NEA Electronics G3[5] gimbals. These gimbals allow the VIP to move freely on two axes and provide a pointing accuracy of 0.0075 degrees, with a top speed of 1.5 degrees per second. This accuracy and slew rate reduces the error associated with pointing the TIS. The mass and power breakdown for the orbiter instrumentation is listed in Table 2-2 below.

Table 2-2 Orbiter Instrumentation

Instrument	Mass (kg)	Power Consumption (W)
ECAM-C50	0.256	2.5
VIMS	33	24
G3[5] Gimbal (2)	3.6	54.8

2.2.2 Atmosphere Probe Instrumentation

The Atmosphere Probe features three separate instruments for sampling the atmosphere of Triton. The first of these is the Omega CY670 cryogenic temperature diode. This temperature sensor is accurate within temperature ranges of 1.4 K to 500 K. This capability allows the Atmosphere Probe to accurately measure the ambient temperature of Triton's atmosphere which is currently stated to be around 38 K. This sensor has its highest accuracy values in the lower temperature range as shown in Table 2- below. From this table, it is shown that the atmosphere of Triton will fall into the highest accuracy range for this sensor

Table 2-3 Temperature Diode Accuracy

Temperature (K)	Accuracy (mK)
1.4	±12
10	±12
77	±22
300	±32
500	±50

The second instrument on the Atmosphere Probe is the Omega ACC797 accelerometer, whose purpose is to measure the acceleration of the probe in the axis of its descent as it falls through Triton's atmosphere. The data obtained from this will then be used with the known values for acceleration due to gravity at triton to calculate the drag force acting on the probe. With the known geometry properties of the probe, this data can then be used to find the density of Triton's atmosphere. Due to Triton's maximum acceleration from gravity being only 0.781 m/s^2, or 0.0797 g_0, a high peak acceleration value for the sensor was not required. Because of this accuracy and mass were prioritized in the selection of the sensor.

The final instrument on the Atmosphere Probe is the LP 1400 pressure transducer. This is included to determine the pressure gradient of Triton's atmosphere. Due to Triton's low atmospheric pressure of around 14 microbars, a sensor that is capable of reading very small pressures was required. This sensor was selected because it has a pressure range of 10.3 to 68.9 microbars. The Atmosphere Probe instrumentation is summarized in Table 2- below.

Table 2-4 Atmosphere Probe Instrumentation

Instrument	Mass (kg)	Power Consumption (W)
CY670 Temperature Diode	.25	2.5
LP 1400 Pressure Transducer	1	0
Omega ACC797 Accelerometer	0.135	1

2.2.3 Geyser Probe Instrumentation

The two instruments featured on the Geyser Probe are the mass spectrometer and dust detection sensor. These are both direct-sensing instruments meaning they will have to be in direct contact with the geyser exhaust in order to analyze it. The purpose of the mass spectrometer is to determine the composition of the particles that pass through it. In addition to this, the dust detection sensor will analyze the particle size and density. Together these instruments fulfill the geyser data requirements. Table 2- below summarizes the Geyser Probe instrumentation. Similar to the VIMS, .the mass and power consumption rates for these instruments were estimated based on the Cassini instruments as a reference.

Table 2-5 Geyser Probe Instrumentation

Instrument	Mass (kg)	Power Consumption (W)
Mass Spectrometer	8	24
Dust Detection Sensor	3	5

3 Concept of Operations

Due to the large C3 needed to depart Earth for the TAGOS mission, the LV that will be used is NASA's SLS Block 1B with an additional Centaur upper stage to provide the needed C3 of $144\ km^2/s^2$ to begin the interplanetary trajectory to Triton. The trajectory will be utilizing a Saturn gravity assist which will provide enough energy to get the spacecraft to Neptune within the JPL RFP required arrival date. Once in orbit around Neptune, the Styker-1 orbiter will conduct gravity assists using Triton, as shown in section 3.2.5, to change the elliptical orbit of the orbiter to be similar to Triton's near circular orbit. This will lower the amount of ΔV needed to get into an orbit around Triton. Once in Triton's orbit, sciences operations will begin and once completed disposal operations will commence.

3.1 Launch Vehicle

The launch vehicle for the TAGOS mission is NASA's SLS Block 1B Cargo which is a requirement of AIAA RFP. The estimated time of completion for the SLS Block 1B Cargo is the year 2022. This means that Stryker-1 and the LV will be in development during the same time. This does increase the risk for the TAGOS mission not meeting JPL's RFP arrival date of December 2035 if the SLS completion date is pushed back. The capabilities of interest of the SLS for this mission are shown in Table 3-1 below and an artist rendering of the SLS is shown in Figure 3-1.

Table 3-1 NASA SLS Block 1B Launch Capabilities, Launch Date, and Cost

NASA SLS Block 1B	
Payload Capability to LEO	105 metric Tons
Payload Capability to Required C3	1 metric Tons
Payload Fairing Diameter	8.4 m
Payload Fairing Height	19.1 m
Launch Cost	$870,000 k
Estimated Year Completion Date	2022

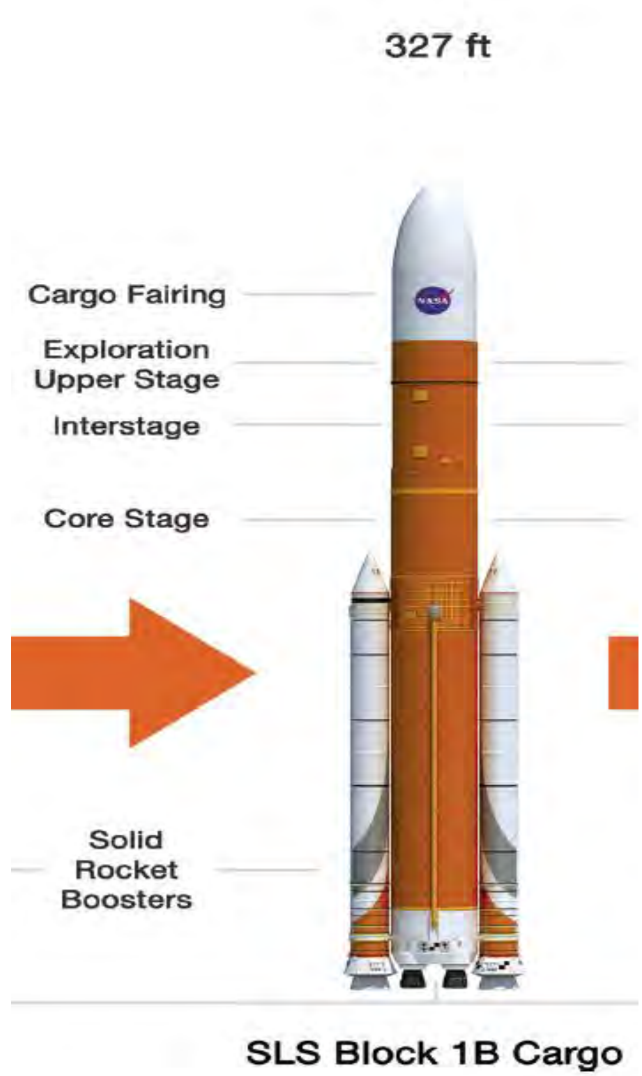


Figure 3-1 NASA's SLS Artistic Rendering. Courtesy: NASA

3.1.1 Launch Vehicle Alternative

Due to the C3, estimated payload mass and required arrival date stated by the JPL RFP there is no alternative launch vehicle that will be able to carry Styker-1 to meet the requirements. If construction is delayed then a later launch window will be used and the JPL requirement would not be fulfilled, however, AIAA RFP will still be due to no required arrival date.

3.1.2 Centaur Upper Stage

Due the predicted mass of Stryker-1, the addition of a Centaur upper stage is needed for the Earth departure V_∞ of 12 km/s. The addition of the Centaur allows up to 7,400 kg to be launched with the required C3. The Centaur was chosen due to its proven reliability and having the highest ΔV capability. Figure 3-2 below shows V_∞ vs. SC mass where the horizontal red line represents the required departure V_∞ and the blue line represents the change in V_∞ with change in mass.

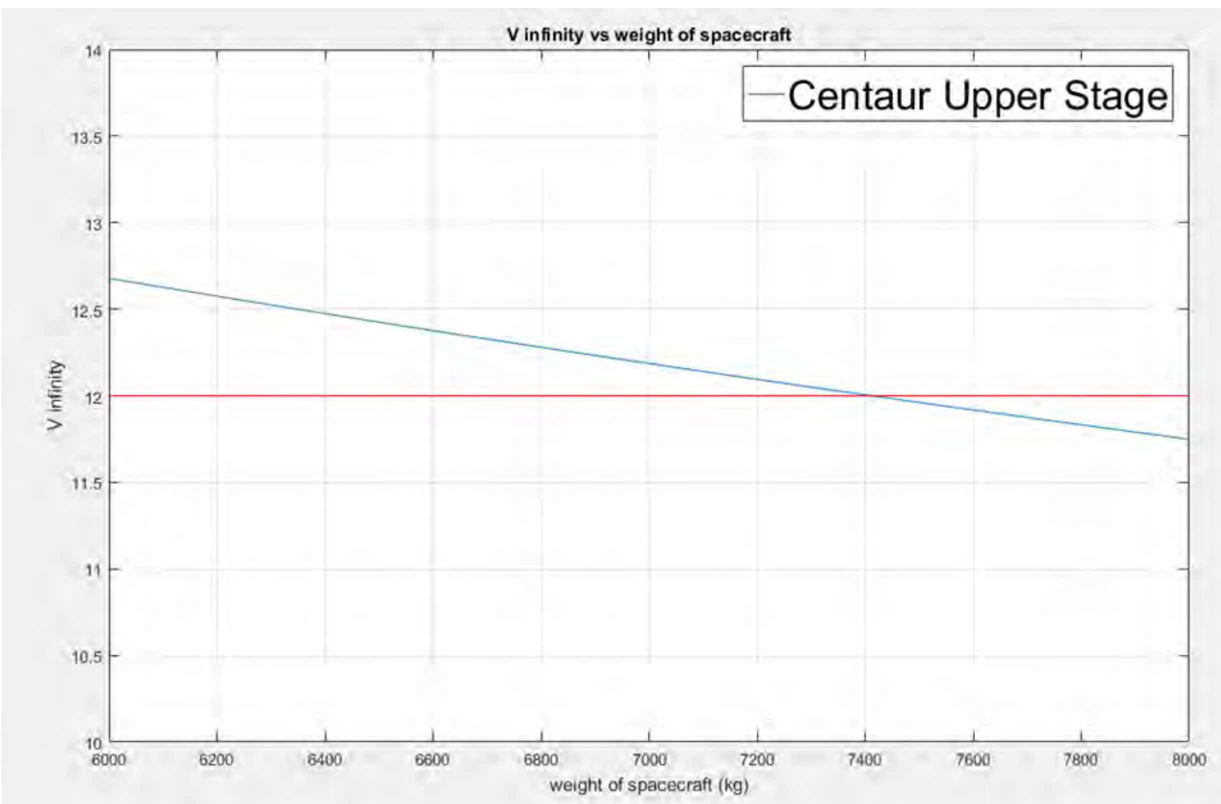


Figure 3-2 V_∞ vs. SC Mass. With a V_∞ = 12 km/s the Centaur can carry a launch mass of 7,400 kg

3.2 Trajectory and Orbits

3.2.1 Trajectory

Trajectories to Neptune typically require high launch energies, and long time-of-flights (TOF) even with gravity assist so with the assumptions described in section 3.2.2 a trajectory was found for the TAGOS mission to Triton that meets the required arrival date in JPL's RFP of December 2035. Alternative trajectories to Neptune were determined by Hughes [1]; however, the arrival date to Triton would put it at most nine years past the JPL RFP requirement so those trajectories weren't chosen. The alternative trajectories are shown in section 3.2.6 as possible trajectories for the AIAA RFP.

The chosen trajectory launches on April 12, 2022 with a C3 of 144 m^2/s^2. The spacecraft will utilize a gravity assist at Saturn on February 23, 2025. The total TOF from Earth to Saturn is approximately 1,048 days. After the flyby, Stryker-1 will travel the last phase of the interplanetary trajectory to Neptune which will arrive on April 2, 2035 which will be a TOF of 3,690 days after the Saturn flyby. In total the interplanetary trajectory will take 12.98 years. A visual representation of the concept of operations (ConOps) for the trajectory to Neptune is shown in Figure 3- below.

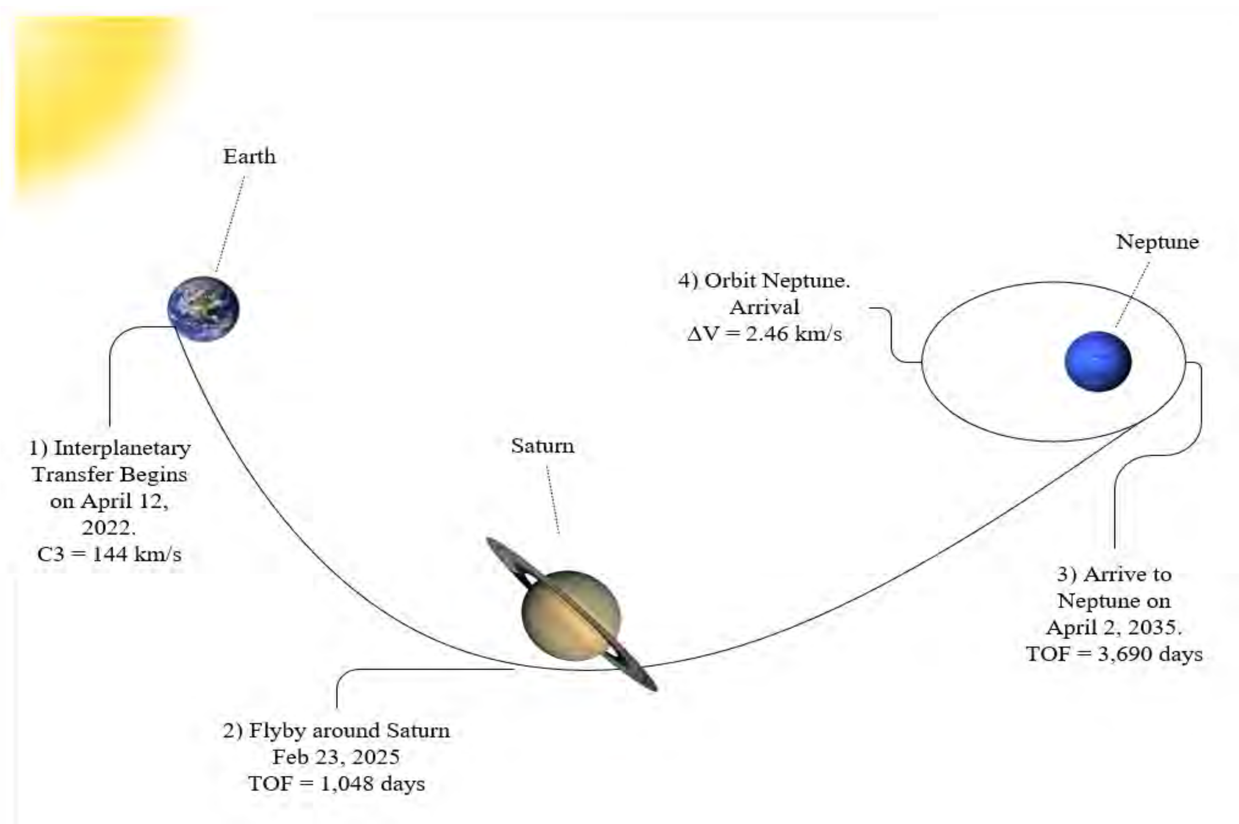


Figure 3-3 Interplanetary Trajectory from Earth to Saturn Flyby to Neptune. Total TOF 12.98 years.

3.2.2 *Assumptions*

Trade studies for the interplanetary trajectory were conducted by Hughes [1] in which the Earth-Saturn-Neptune trajectory was found. In determining the optimal trajectory to Neptune, certain constraints were imposed to reduce the number of trajectories in the study which is shown in Table 3-2 below.

Table 3-2 Trajectory Search Constraints

Parameter	Value
Minimum flyby altitude	50k km (Saturn)
Maximum TOF	15 years
Maximum launch V_{∞}	17 km/s (1 flyby)
Maximum maneuver ΔV	3 km/s

3.2.3 *Earth Departure Launch Windows*

Due to the possibility of launch delays, a departure timespan was determined by solving Lambert's problem. Since the position vectors of the Earth and Neptune can be determined using an ephemeris, a MATLAB program was created using modified code created by Curtis [2]. Figure 3-4 below is the Porkchop plot created of the desired launch date at Earth with the Saturn gravity assist arrival date.

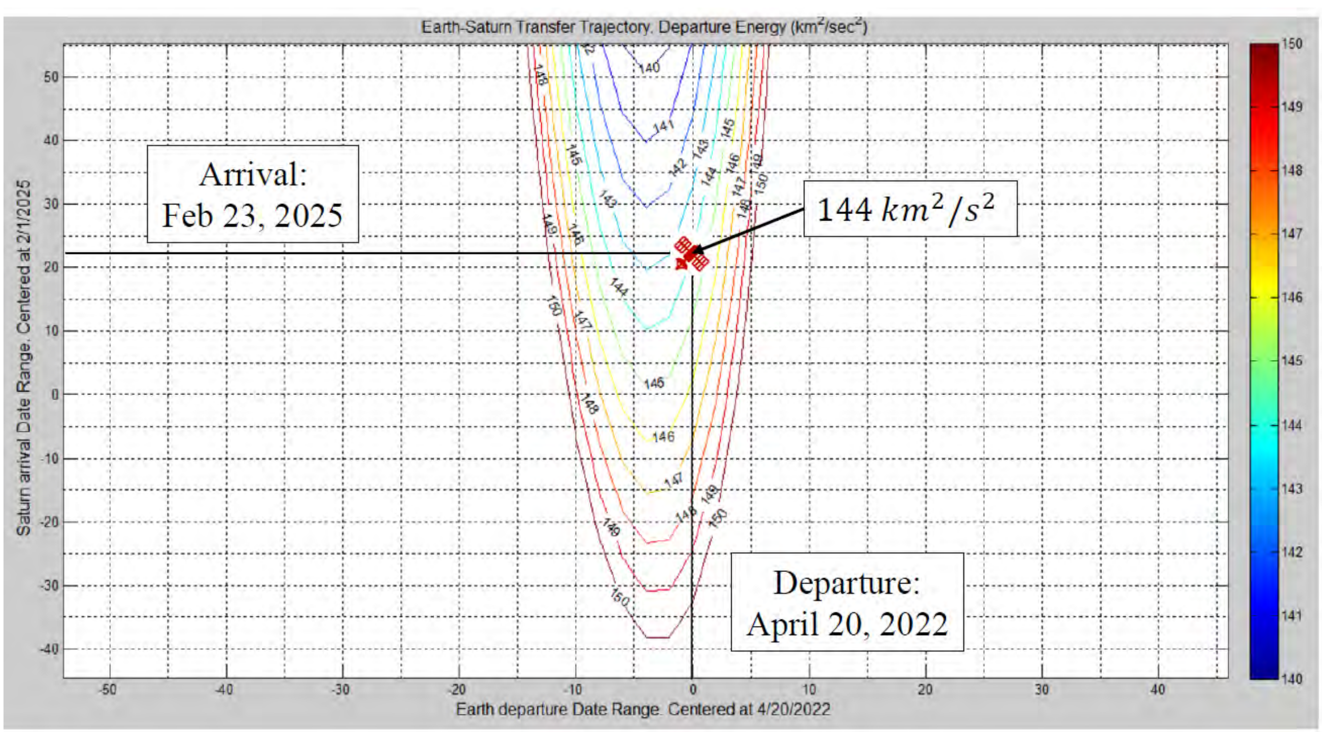


Figure 3-4 Earth Departure Window. Each colored line represents a change in C3

Figure 3-5 below shows the change in C3 if launch date is pushed back by 0, +5, +10, +15, +20 past the April 20, 2022 launch date. A relatively small change in C3 occurs at +5 days past planned launch date, however after nine days, the C3 begins to increase substantially causing the amount of launch mass to decrease assuming the desired arrival to Saturn stays the same.

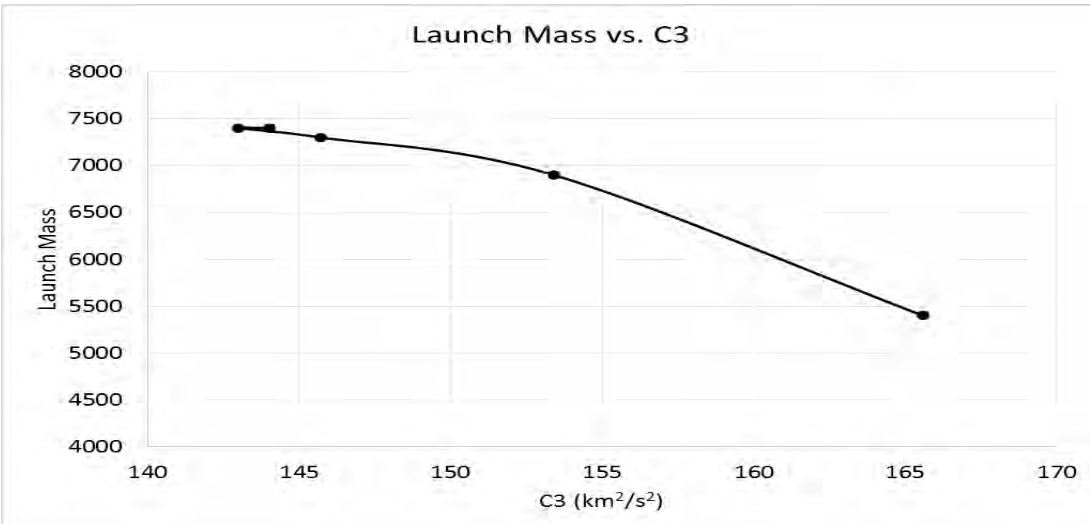


Figure 3-5 Change in C3 and Launch Mass Due to Launch Delay

3.2.4 Neptune Arrival

Hughes [1] found upon arrival to Neptune, a ΔV of 2.46 km/s will be provided by Stryker-1 to capture in an elliptical orbit with a period of 100 days, semi-major axis of 95 Neptune radii (approximately 2.352 million km) and periapsis of 2.5 Neptune radii (approx. 61,910 km).

An inclination change is expected once Stryker-1 gets into orbit to become coplanar with Triton. A large capture orbit period will lower the amount of ΔV needed for this maneuver. In Figure 3- below from Hughes [1] is a graph of the ΔV needed depending on the inclination change.

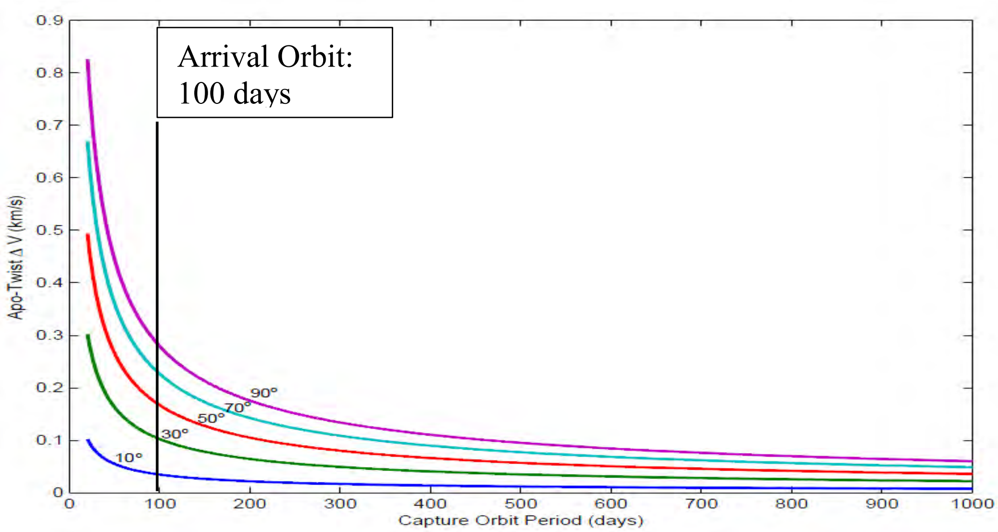


Figure 3-6 Apo-Twist ΔV (km/s) required for various inclination changes.

3.2.5 Neptune-To-Triton Transfer

Stryker-1 will utilize a Triton gravity assist to decrease the energy of the orbit around Neptune to reach a near circular orbit, as shown in Figure 3- below.

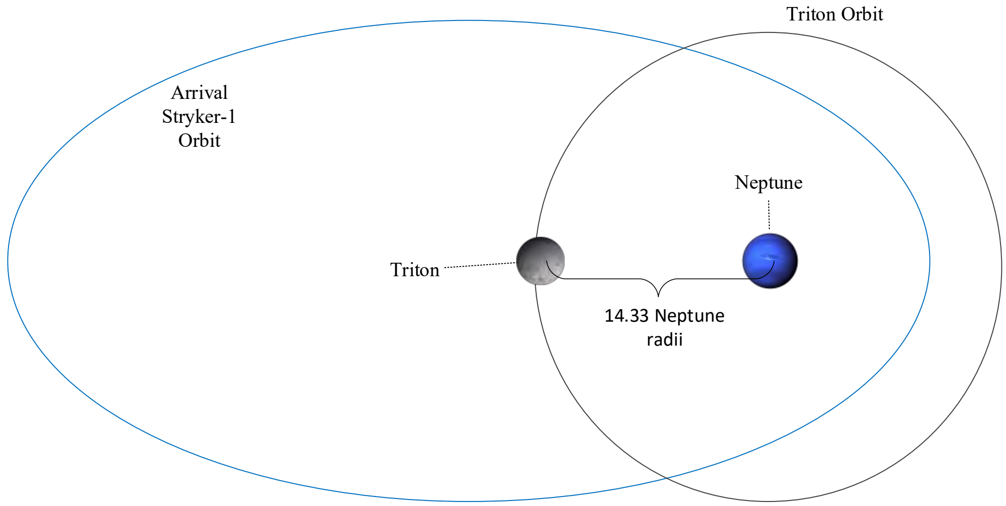


Figure 3-7 Triton and Stryker-1 orbit around Neptune

To accomplish this, while in orbit around Neptune, gravity assist at Triton will be used to decrease the energy of Styker-1 to get into an orbit around Neptune that is similar to Triton's. These maneuvers will decrease the ΔV needed to get into a Triton polar orbit which will range from $0.25\ km/s\ to\ 0.5\ \ km/s$. Figure 3-8 below shows possible maneuvers. After achieving the desired orbit, Stryker-1 will transfer into a polar orbit around Triton with an altitude of 90 km in December 2035.

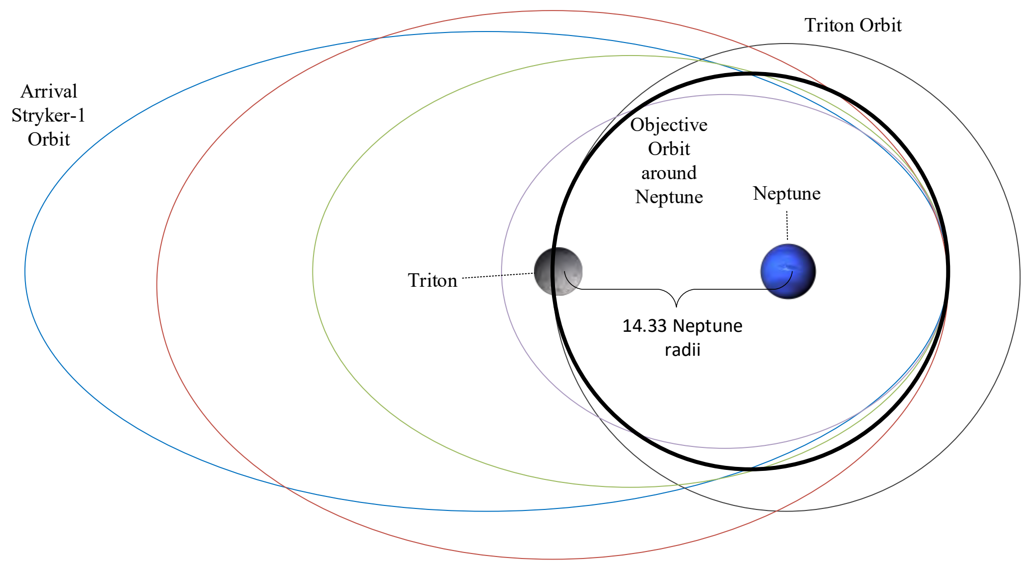


Figure 3-8 Blue is arrival orbit around Neptune and thick black line is the desired orbit to transfer to Triton's orbit.

3.2.6 Alternative Interplanetary Trajectories to Neptune

Table 3- below shows alternative trajectories to Neptune. These trajectories were not chosen due to the arrival date at Neptune which is past the JPL RFP arrival date requirement. However, Table 3- is included because the AIAA RFP does not state a required arrival date, meaning these trajectories can be utilized if the SLS completion is delayed. The alternative trajectories come from Hughes [1] which were considered because of the low C3 required.

Table 3-3 Alternative Departure Trajectories.

Path	Launch Date	Arrival Date	$V_{\infty,L}$ (km/s)	ΔV_A (km/s)	TOF (years)	Maneuver ΔV (km/s)	Total ΔV (km/s)
VEJN	9/20/2029	1/2044	4.50	3.27	14.30	2.23 (PF)	**5.5**
VVEJN	1/24/2028	12/2042	7.00	4.30	14.90	-	**4.3**

VEJN is a Venus-Earth-Jupiter flyby and VVEJN is a Venus-Venus-Earth-Jupiter flyby. A $2.23\ km/s$ powered flyby is needed for the VEJN trajectory while Stryker-1 is doing a gravity assist around Earth.

3.3 Science Operations and Data Retrieval

After orbital insertion and plane change into a polar orbit at Triton, the attitude control system (ACS) will confirm the positioning of the Stryker-1 orbiter with its star trackers and inertial measurement units (IMU). Then the orbiter will switch to a nadir pointing mode, with the ACS using its thrusters to maneuver the orbiter into position.

The orbiter will be imaging Triton with the Triton Imaging System. A polar orbit around Triton is utilized to ensure full coverage of the moon can be obtained. Simulations of the orbiter and the TIS have shown that the imaging will take approximately 2 years to cover the entire surface of Triton, considering the TIS's FOV and the orbiter's altitude. Figure 3-9 below shows a preliminary simulation.

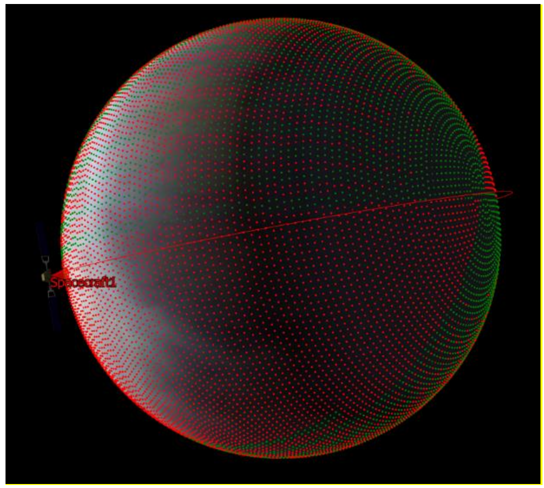


Figure 3-9 Screenshot of FreeFlyer Simulation with TIS FOV

While the orbiter is imaging Triton, the high gain antenna (HGA) will continuously track Earth. This will allow for a constant link via the DSN for commands and telemetry. During the imaging, full mapping data will be stored on a digital tape recorder (DTR), and a compressed version will be stored on a solid state recorder (SSR). The data stored on the SSR will be sent back to Earth from the orbiter with the available bandwidth while imaging is occurring.

The orbiter will process the full imaging data on board to look for a potential landing site for the Atmosphere Probe and the Geyser Probe. The Atmosphere Probe needs a landing trajectory 75 km upwind or 40 km crosswind of any geysers and outside the periphery of the geyser zone to gather atmospheric data that meets the requirements set in JPL's RFP. The Geyser Probe needs a landing trajectory that travels through a geyser plume and eruption cloud, and lands on the surface in the geyser zone. When a potential landing site for either probe has been found while imaging, the full imaging data will be stored on the orbiter on the SSR.

After the full surface has been imaged, the potential landing sites will be ranked for each probe, and then the top ranked sites will be sent back by the orbiter via the DSN to Earth for ground control crews to further examine.

After a landing site for the Atmosphere Probe has been identified, the orbiter will perform an orbital maneuver to alter its longitude of the ascending node, so that its orbit is over the landing zone of the Atmosphere Probe. The orbiter will then reposition its HGA to nadir pointing so that it can communicate with the probe during its descent. Once the orbiter has the correct orbital position and antenna position it will drop the probe. The probe will use its instruments to gather atmospheric data every 100 m during its 480 second descent. The probe will be constantly sending scientific data and telemetry to the orbiter during its descent. The orbiter will continue tracking the Atmosphere Probe with its HGA until its hard landing on Triton's surface. The orbiter will then reposition its HGA to relay the data collected to the ground crew on Earth via the DSN.

The ground crew on Earth will then examine the data to better understand the atmosphere of Triton. They will also then edit the parameters of the descent software for the Geyser Probe's soft landing algorithm. These edited parameters will then be sent to the orbiter via the DSN so that the software on the Geyser Probe can be updated.

After a landing site has then been identified for the Geyser Probe, and the software has been updated for the Geyser Probe, the orbiter will then adjust its orbital position again to now pass over the landing zone for the Geyser Probe. The orbiter will then position its HGA to nadir pointing in preparation communicate with the probe during its descent and after landing. The TIS will then image the landing zone as soon as it comes into view so that it can confirm that the geyser is active. If confirmed that the geyser is active in the landing zone, the probe will be dropped into the landing zone. The probe will use its mass spectrometer and dust detection system to gather data on the geyser exhaust composition and particle size during its descent. The probe will be constantly sending scientific data and telemetry to the orbiter during its descent. After 460 seconds of free fall, thrusters will fire for approximately 43 seconds to perform a soft landing. Once landed the probe will collect soil samples and send the data back to the orbiter. Once the probe sends a "Low Power, Shutting Off" signal to the orbiter via telemetry, the orbiter will reposition itself to point its HGA at Earth. The orbiter will then relay the data collected to the ground crew on Earth via the DSN.

The orbiter will attempt to maintain an orbital position so that it can maintain communications with Earth via the DSN during its entire polar orbit of Triton. Small orbital maneuvers will be needed to alter the longitude of ascending node so that the ascending node will be perpendicular to Earth. The HGA will track the Earth to maintain communications. The orbiter will then finish transmitting all the remaining mapping data on the SSR.

3.4 HAIL MARY: Secondary Data Return Phase

Due to limited telecommunication capabilities hindered by Triton's distance from Earth, Stryker-1 is only capable of returning the required 10 meter per pixel geyser zone mapping data, atmospheric data, and geyser zone samples within the mission time frame. Although this does fulfill mission requirements, a large amount of potential data including the goal of full surface mapping at 1 meter per pixel resolution is unable to be recovered. Because of this, an additional post-mission phase, the Helio-Assisted Intel Launch for Mission Asset RecoverY (HAIL MARY) is the proposed solution. The purpose of this mission phase is to return a large amount of extra data by sending a segment of the Stryker-1 orbiter on a trajectory towards Earth and transmitting data as it approaches.

The HAIL MARY phase begins in August 2041 by separating the data return module (DRM) from the remainder of the orbiter. This is done to reduce the amount of propellant required to escape from Neptune's SOI. The separated orbiter components include the empty propellant tanks, the science instrumentation and scanning platform, and all structural components above the separation plane. The remaining components on the DRM include the power supply, propulsion system, attitude control system, command and data system, thermal control, telecommunication system, and the remaining fuel required for the escape from Neptune and course corrections. The mass statement for the data return module is shown in Table 3- below.

Table 3-4 Data Return Module Mass Statement

Subsystem	Mass (kg)
Structure	100
Power	350
Thermal	11
Command and Data	47
Telecommunication	156
Propulsion	170
Attitude Control	14
Propellant	422
Total	**1,270**

After the DRM has been separated, a ΔV of 1.52 km/s is provided by the propulsion system. This, along with aid from Triton's retrograde orbit velocity will allow the DRM to escape Neptune's SOI in the radial direction toward the Sun. From that point on, the data return module will begin to be pulled by the Sun's gravity alone, setting it on a trajectory inwards through the solar system.

Error! Reference source not found. is a plot of the DRM's distance from the Sun vs. time as it approaches Earth. This plot was used along with planetary orbit data to correctly determine the launch window that would arrive at Earth's orbital radius with Earth in the correct position. This was also used to address the risk of crashing into another planet by ensuring that the trajectory would not cause the data return module to enter another planet's SOI.

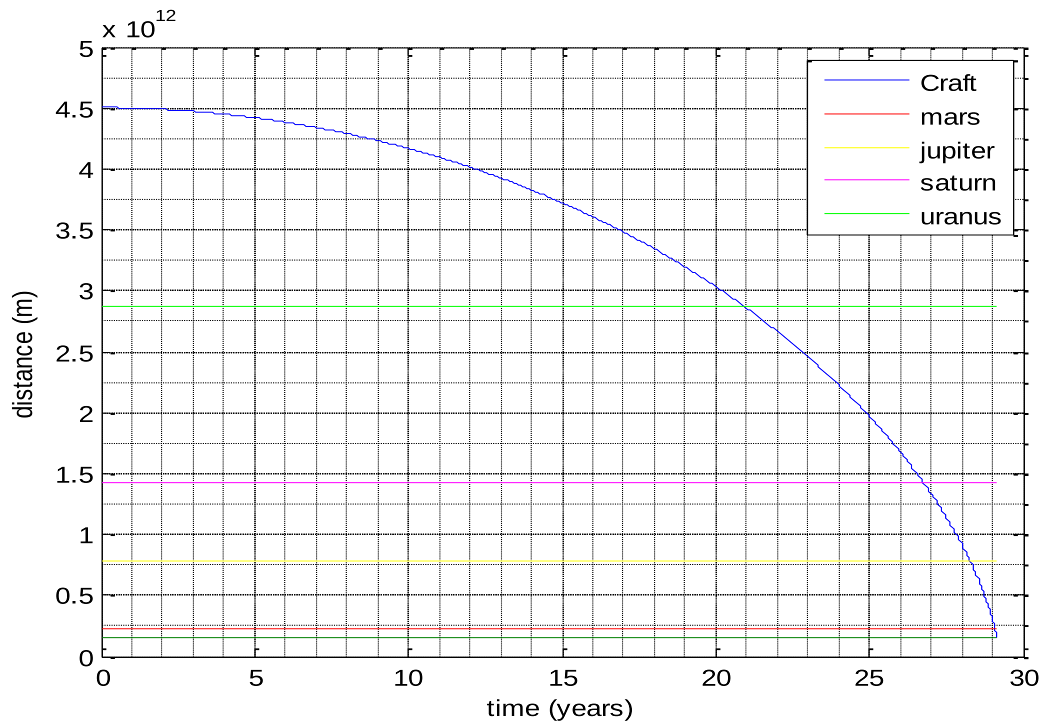


Figure 3-10 DRM Distance from Sun vs Time Plot

Figure 3-10 depicts that a total duration of 29.17 years is required to return to Earth. During the first 26.75 years of this, the data return module will remain in cruise mode not transmitting any data back to Earth, only requiring once a week communication to determine necessary course corrections. This continues until the data return module reaches 8.5 AU from Earth, at which desired data rates can be achieved. At this point, regular communication with Earth with recommence for 12 hours a day. This allows for a symbol rate of 75.16 ksps. Data transmission will continue with increasing rates until the data return module passes Earth. The total additional data capable of being transmitted through this is 17.4 terabytes, including full surface mapping at 10 m resolution with additional 1 m resolution imaging. After Earth is passed, all communication with the data return module will end and it will continue on its trajectory into the Sun.

The HAIL MARY phase provides additional value to the mission by greatly increasing the amount of data that can be returned to Earth. While it does add on over 29 years to total mission duration, the majority of this time does not require constant communication with the data return module meaning the cost is minimized for this segment. All risks associated with performing this procedure will not affect completion of mission requirements since it occurs after all required data has been returned.

3.5 Disposal

For the disposal of the equipment from the mission the team first referred to the planetary protection guidelines given by NASA [3]. Under these guidelines, Neptune and Triton both fall under Category II missions which are allowed flybys, orbiters and landers. Therefore, the plan for disposal is to leave the Geyser Probe and Atmosphere Probe on Triton and send the separated portion of the Stryker-1 orbiter to Neptune after it detaches from the data return module. The data return module of the Stryker-1 obiter will be disposed of by the final part of the mission where it is sent into the Sun.

4 Vehicle

Multiple requirements from the RFP require measurements to be made at different atmospheric intervals as well as within geyser exhaust. The orbiter alone will not be able to fulfill those requirements, thus an Atmosphere Probe and Geyser Probe were made in order to be able to conduct those experiments and be able to send the information back to the orbiter. The Atmosphere Probe will be able to analyze the density, temperature, and pressure at 100 meter intervals, while the Geyser Probe will be able to sample data in the geyser plume, eruption cloud, and surface of Triton. It will also be able to analyze the composition, particle size, and density at Triton.

4.1 Orbiter

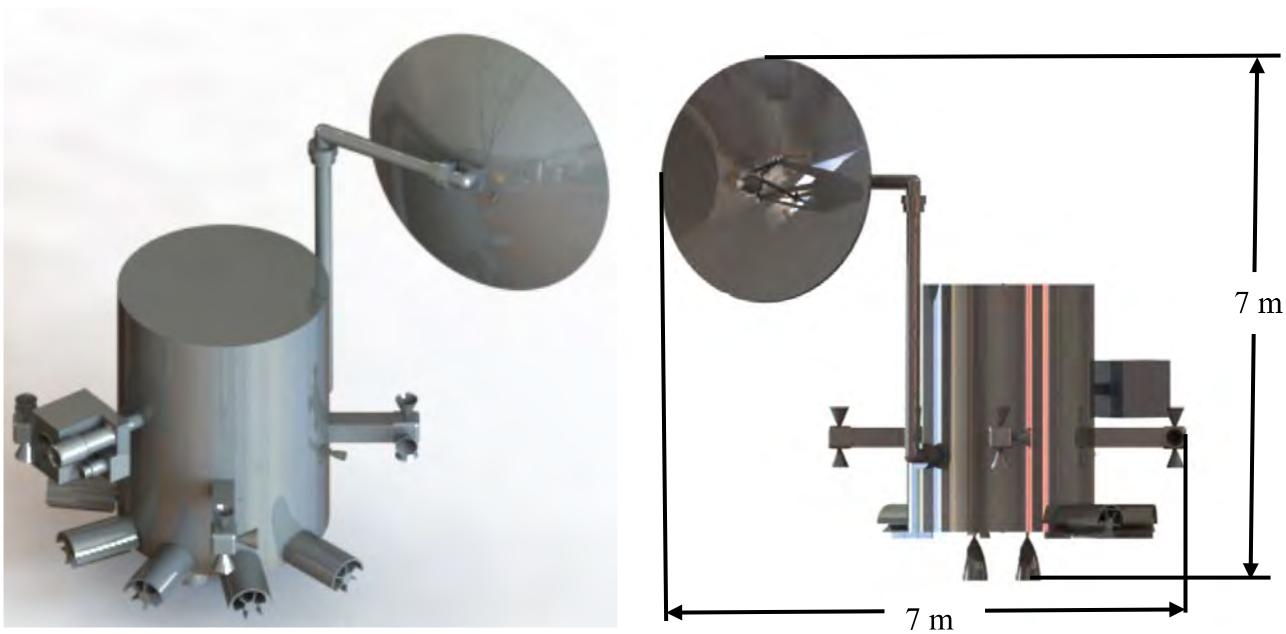


Figure 4-1 Stryker-1 orbiter CAD Model

A model of the Stryker-1 orbiter is shown in Figure 4-1 above. This design was chosen by down-selecting 2 original designs. The design consists of a cylindrical structure with thermal protection, propulsion system, telecommunication system, command and data system, power system, and an attitude control system. The Stryker-1 orbiter mass statement is shown in

Table 4-1 below. These systems are detailed in the following sections.

Table 4-1 Orbiter Mass Table

Subsystem	Mass (kg)
Structure	202
Thermal Control	23
Attitude Control	14
Power	380
C&DS	47
Comm.	156
Propulsion	427
Payload	37
Total	**1,286**

4.1.1 Structure

Stryker-1 orbiter structure was largely inspired by spacecraft like Voyager and Cassini with influence coming from *Spacecraft Structure* by Jacob Job Wijker [4]. The orbiter structure must be designed as to withstand the acceleration estimates as provided from the SLS Mission Planner's Guide table below.

Table 4-2 Estimated Axial and Lateral Loads Experienced during Launch

Acceleration Direction	Lift Off	Transonic	Max Q x α	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

The main body consists of a titanium frame for both the bottom and top portions of the orbiter. Each half of the orbiter consists of four semi-cylindrical titanium frames to support the major loads induced by launch, max Q, and separation. The titanium frame design is shown in Figure 4-2 below Each of these frames is attached to the other using a strap and 12 bolts. For splitting the two halves of the spacecraft, the top and bottom are mounted together by explosive bolts and guillotine cable cutters for the electronics.

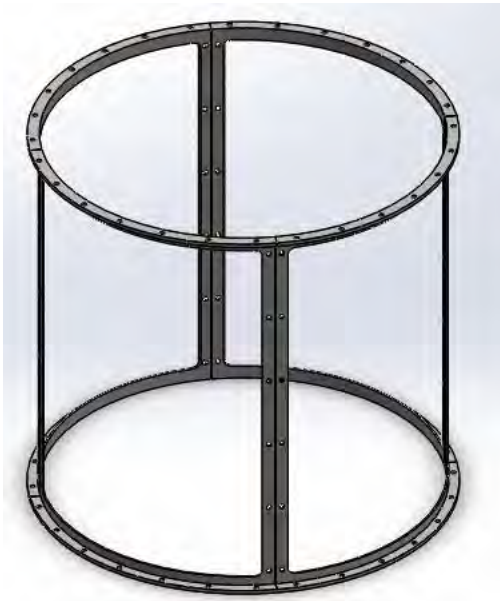


Figure 4-2 Stryker-1 Titanium Internal Frame

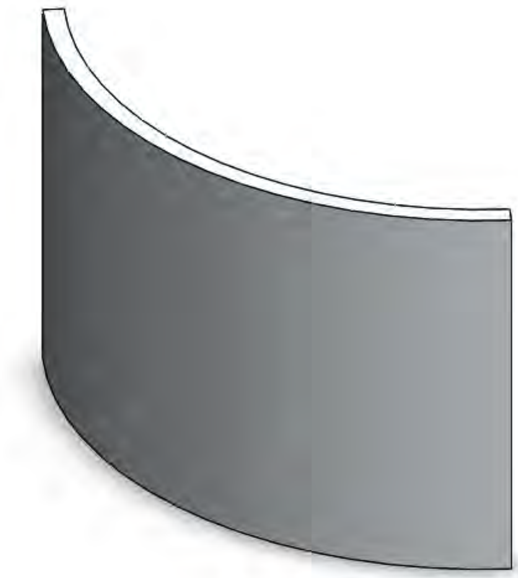


Figure 4-3 Stryker-1 Curved Aluminum Honeycomb Wall Panels

To protect the internals and reduce overall mass, rectangular composite honeycomb curved panels are used and mounted to the exterior of the titanium frame. These panels are shown in Figure 4-3 above. The honeycomb core of the structure will be made of an Aluminum alloy while the face sheets are composite. 8 total composite honeycomb panels will be necessary to cover the entirety of the exterior shell. Portions of the panels will be cut where necessary to allow for science instrumentation and the deployable antenna beam to protrude from the spacecraft.

Honeycomb core, composite, and other material properties are listed in

Table 4-3 and

Table 4-4 below. The tables list many possible options for when a detailed design phase is reached.

Table 4-3 Common Aluminum Honeycomb Core Properties

	d_c cell	Density ρ	Compressive Strength E_c	Shear Modulus (ksi)		Shear Strength (psi)	
Type of Honeycomb Core	**(in)**	**(lb/ft^3)**	**(psi)**	**G_L**	**G_T**	**τ_L**	**τ_T**
1/4 - 5056 - .002p	0.252	4.31	465.57	67.01	26.98	324.88	190.00
3/8 - 5056 - .0007p	0.378	1.00	34.81	14.94	8.99	44.96	24.66
1/4 - 5056 - .0015p	0.252	3.37	314.73	50.04	22.05	230.61	130.53
1/4 - 5056 - .0007p	0.252	1.62	79.77	20.02	12.04	78.32	37.71
3/16 - 5056 - .002p	0.189	5.68	735.34	93.98	35.97	480.08	279.92

Table 4-4 Common Metal and Non-Metal Fiber Properties

Metal Properties				
Material	**Density ρ (lb/ft^3)**	**σ_ν (ksi)**	**σ_ψ (ksi)**	**E (ksi)**
Aluminum				
2014-T6	0.1748	64.00	56.00	10.44
2024-T36	0.1729	70.00	60.00	10.44
6061-T6	0.1692	42.00	35.00	9.72
7075-T6	0.1748	75.90	65.00	10.30
Titanium				
Ti6Al-4V	0.277	160.00	144.90	15.95
Non-Metal Fiber Properties				
Material	**Density ρ (lb/ft^3)**	**σ (ksi)**		**E (Msi)**
E-Glass Fiber	159.18	459.78		10.49
S-Glass Fiber	156.0549	599.01		12.00
E-Glass in Epoxy	121.0986	200.15		7.50
S-Glass in Epoxy	121.0986	300.23		7.50
Aramid Fiber	105.4931	499.66		19.99
Aramid Fiber in Epoxy	87.39	279.92		12.00
HM Graphite Fiber	118.6017	300.23		54.99
HT Graphite Fiber	110.4869	349.54		34.95
HM Graphite in Epoxy	100.50	134.89		30.02
HT Graphite in Epoxy	93.633	204.50		22.05

4.1.2 Thermal

One of the goals for this mission is to be able to keep the orbiter within operational temperatures in order to have a fully functional orbiter in deep space. Table 4-5 below demonstrates the requirements for the orbiter thermal control system. It can be concluded from the table that the biggest factor will be to keep the orbiter and its instrumentation within operational temperature range.

Table 4-5 Orbiter Thermal Requirements

Requirement #	Description
4.1.2.1-1	Keep spacecraft within operational temperatures when around Earth's atmosphere, cruise, and arrival to Neptune
4.1.2.1-3	Keep subsystems and its payload in operating temperatures
4.1.2.1-4	Thermally isolate propulsion components from vehicle structure

Although the instrumentation has been selected in order to complete the mission requirements, the instrumentation has to be within operational temperatures ranges. Table 4-6 below demonstrates each instrument along with subsystem components that require thermal protection with their respective temperature limits.

Table 4-6 Instrumentation Temperature Limits

	Operational Temperature Range (°C)	Non-Operational Temperature Range (°C)
Imaging System	-200 to -80	-200 to -60
Visible Mapping Spectrometer	-153 to -133	-153 to -113
Hydrazine Tanks	-15	-15

Once the temperature limits are obtained, different methods can be considered in order to maintain the correct temperature. The goal of the thermal control system is to thermally insulate the orbiter from the environment, resulting in minimal heat loss. Also, since the orbiter will be approximately 2.8 billion miles away from the Sun, it will be assumed that there will be minimal heat flux from the Sun, meaning all incoming heat flux is from Neptune. Values for the heat radiated and the infrared flux from Neptune were calculated**.** These values are shown in Table 4-7 below.

Table 4-7 Radiation Transmitted from Neptune and the Sun

	W/m^2
Heat radiated from Neptune	1.52
IR Flux from Neptune	0.52

Table 4- shows that the orbiter will receive minimal radiation from Neptune, thus stating that the orbiter will experience temperature approaching 0 Kelvin and making the focus of the thermal control system to have provide additional heating and insulation.

Multi-layer insulation is necessary to protect the orbiter from incoming micrometeorites and debris from the thrusters, as well as keeping the instrumentation from freezing. MLI is also lightweight, which will help keep the orbiter from going over its mass limit. It will be necessary to use inner and outer MLI blankets. Table 4-8 demonstrates what material was selected for the inner and outer MLI layers as well as material properties.

Table 4-8 MLI Characteristics

	Inner-Layer Material	Outer-Layer Material
Material	Goldized Kapton	Coated & Backed Kapton
Number of Layers	20	10
Density, g/cm^3	0.0011	0.028
Thickness, mm	0.0076	0.013
Temp. Range, °C	-250 to 288	-73 to 65
Absorptivity, α	0.31	0.41
Emissivity, ε	0.5	0.40

From the data in Table 4-8 it was calculated that the MLI will be able to help keep the orbiter within operating temperatures throughout the mission. Although MLI will thermally insulate the orbiter, it will not be enough to keep the orbiter at the required temperature. The thermal wattage from the radioisotope thermoelectric generators (RTG) is used to provide additional heating to the orbiter. With the MLI at 30 layers it is calculated that the effective emissivity near the center will be 0.00235 and 0.0588 at the seams. From these emissivity values along with the required temperature ranges, the necessary thermal wattage required for heating the orbiter was calculated. Table 4-9 shows the results obtained for when instruments are operational and non-operational.

Table 4-9 Maximum and Minimum Thermal Wattage Calculated

	W
Maximum Q	603
Minimum Q	342
Total thermal wattage for 6 RTGs	12,000

Since the total thermal wattage is 12,000 thermal watts and the maximum thermal wattage needed is 603 W, it can be concluded that the RTGs are capable of providing enough thermal wattage to keep the orbiter within operational temperatures.

Once the thermal wattage necessary is known, the next task is to be able to transfer the excess heat from the RTGs to the rest of the orbiter. The constant-conductance heat pipe was selected along with the Groove Wicks design. This design was selected since it is inexpensive, has a high TRL and is able to perform consistently. It is also capable of being implemented as a flat plate, which allows for lower volume requirements. The heat plates are comprised of aluminum and utilize ammonia as the fluids. These were selected to minimize the mass of the system.

The high and low-gain antenna will experience large temperature changes and will need to be thermally insulated to prevent warping of the dish causing degradation of the signal. The concave side will be painted white, and the other side will be covered in MLI to keep it within an operational temperature range. If these methods will not be effective, an alternative can be to use composite material reflectors. These have a very low coefficient of thermal expansion and can withstand very large temperature swings.

As stated in Table 4-5, goal 4.1.2.1-4 states that the propulsion system should be thermally isolated from the vehicle structure. The vehicle's propulsion system will be using hydrazine, which has an operating temperature minimum of 15 °C. One way to help keep the propulsion system at operating temperatures is to cover the structure with MLI. This will help the heat emitted to be kept within the structure. A propellant lines heater will also be used. The heater power density will be varied along the line for temperature variations. The MLI will also cover the thruster valves and injectors, which will minimize heat loss when non-operational. Thermostatically controlled heaters will also be implemented to keep the valves and injectors within acceptable temperature limits when it is not operating. A heater will be necessary due to the heat lost by radiation from the non-insulated nozzle. A heat shield will also be implemented which will help protect the orbiter against thruster radiant heat and rocket plumes. The heat shield will be made of titanium and will help resist against high temperatures and will have low emissivity Gilmore **[5]**. Table 4-10 demonstrates the mass summary for the thermal control system. Masses for the thruster heater components are included in the propulsion system line items.

Table 4-10 Thermal Control System Mass Summary

	Mass (kg)
MLI	3.48
Heat Pipes	5.87
Thermostat	0.13
Foam Insulation	2.58
Heaters	2.56
Radiator	2.81
Paint	5.36
Total	**22.79**

4.1.3 Propulsion

The design process of the propulsion system for the Styker-1 orbiter began with defining the requirements for the propulsion system in order to ensure all mission needs are met. The first major requirement is the propulsion system has to fulfill the ΔV budget, shown in Table 4-, which notably includes a large Neptune arrival burn of 2.4 km/s. The second major requirement is the propulsion system has to accommodate the needs of the ACS since small impulse thrusters will be used for 3-axis attitude control.

Table 4-11 ΔV Budget for Entire Mission in km/s

ΔV Budget (km/s)	
Neptune Capture	2.4
Apoposeidon-twist maneuver	0.1
Triton Capture	0.4
Triton Plane Change	0
Stationkeeping	0.1
Main Mission Total	3.1
HAIL MARY Phase	1.5
Allotted Δv Budget	**4.6**

A trade study between the various types of propulsion system types was performed in order to select the most appropriate system for meeting mission requirements. Ultimately, a dual mode system utilizing liquid bipropellant and liquid monopropellant was selected in order to meet the mission requirements. A summary table of the trade study is listed in Table 4- below.

Table 4-12 Trade Study for Type of Propulsion System [6]

Requirement	Monopropellant	Bipropellant	Solid
I_{sp} (s)	165-260	300-450	<300
Impulse Range (N-s)	<45,000	>45,000	>45,000
Restart	Yes	Yes	No
Pulsing	Yes	No	No

Ion engines and cold gas are not applicable to mission because of thrusts requirements for the arrival burn and would not meet the required arrival date if the mission was altered to compensate for the low thrust provided by those types of systems. Solid motors are not appropriate due lack of restart capability which is a requirement for the mission since there are multiple burns during the mission. The advantage of a dual mode system is that it takes strengths of two types of systems to effectively meet multiple requirements that would have been other compromised if only single system was used. A liquid bipropellant system is most applicable to meeting the requirement of the larger burns in mission. A bipropellant system provides large thrusts, restartablility, and higher I_{sp} than monopropellants, which minimizes fuel mass requirements. However, bipropellants are not the most suitable solution for meeting the ACS requirements because of their larger impulse ranges. To compensate, a monopropellant system is utilized for its pulsing capabilities in order to meet the ACS's strict pointing requirements. Together a liquid bipropellant and monopropellant dual mode system utilizes the two types of systems' strengths in order to meet the two major requirements of the propulsion system without comprising on appointing accuracy or adding fuel mass due to a low I_{sp}.

The propellant that was selected for the dual mode propulsion system is hydrazine as the fuel and nitrogen tetroxide (NTO) as the oxidizer. Cryogenic fuels were ruled out because of the mission length being 18 years (48 including HAIL MARY phase), which would make storing the cryogenic propellant for that length undesirable due to difficulties storing cryogenics for long periods without boil-off losses. Hydrazine was chosen as the fuel due to its heritage as being used both as a bipropellant and monopropellant. NTO was chosen as the oxidizer as it is the standard oxidizer to be paired with hydrazine. Both hydrazine and NTO have simple storability requirements compared to cryogenic fuels. The combination of hydrazine and NTO for dual mode systems has proven heritage as this combination has been used in the Mars Global Surveyor and Juno missions [6] [7].

A trade study between several large thrusters was done in order to select the main engine for the Strker-1 orbiter, shown in Table 4-13 below. Aerojet's HiPAT DM was selected due to having the highest I_{sp} of the flight proven thrusters. The selected thruster is depicted in Figure 4-4. The Styker-1 orbiter implements 2 HiPAT DM thrusters one as the main thruster and the second as a back-up. The mission ConOps was created only considering the use of one thruster at a time, so the second thruster was added as a redundant system to minimize the risk of main thruster failure.

Table 4-13 Trade Study of N_2H_4/NTO Thrusters

Manufacture	Model	Thrust (N)	Mass (kg)	Diameter (m)	Height (m)	I_{sp} (s)	Heritage
Aerojet [8]	**HiPAT DM**	**445**	**5.44**	**0.362**	**0.727**	**329**	**Flight Proven**
Aerojet [8]	R-42 DM	890	7.3	0.381	0.7112	327	TRL 6
Aerojet [8]	AMBR	623	5.4	0.362	0.7257	333	TRL 6
Moog-ISP [9]	LEROS 1b	635	4.5	0.289	0.54	317	Flight Proven

Figure 4-4 Stryker-1 Orbiter's Main Thruster (Aerojet HiPAT DM Thruster) [8]

The Styker-1 orbiter's propulsion system will implement a pressure regulated system. A pressure regulated system was selected since bipropellant will be used and a consistent flow rate and inlet pressure is desired. A full blow-down system was not selected due to being typically more mass intensive when compared to a pressure regulated system. Helium was the selected pressurant as it is the most mass efficient choice of inert gases.

Propellant management devices (PMD) will be used over a diaphragm system for the fuel tank design. Despite a diaphragm system being generally simpler, it was not implemented in the fuel tank design due to diaphragm systems having typically lager masses over PMD based systems. A vane, sponge, trap combination will be used as the PMDs as they have extensive heritage in many tanks manufactured by Orbital ATK [10]. Composite overwrapped pressure vessels (COPV) will be used over solid metallic vessels. COPV offered a more mass efficient solution over a similar sized titanium tank. The propulsion system will utilize 7 hydrazine tanks, 7 NTO tanks, and 2 He pressurant tanks. A large number of small spherical tanks will be implemented in order to increase the packing efficiency of the propulsion system because volume was a design driver for this instance.

The propulsion system for the Styker-1 orbiter is shown in a line diagram below. The layout of the propulsion system started off by following similar dual mode system design of the Mars Global Surveyor found in Brown [6]. However, during the design process and mission constraints the propulsion evolved from a simple 2 pressurant tanks, 2 oxidizer tanks, 2 fuel tanks design to a 2 pressurant tanks, 7 oxidizer tanks, 7 fuel tanks design. This was done in order to save volumetric space by implementing hexagonal close packing. Additionally, the design adapted based on the addition of a secondary mission which later included a need for the orbiter to be able to segment itself, so the propulsion system was redesigned to have the ability for separate as well as encompassing the lager number of tanks. The line diagrams for each major iteration of the propulsion system are shown below as Figure 4-5 and Figure 4-6 below.

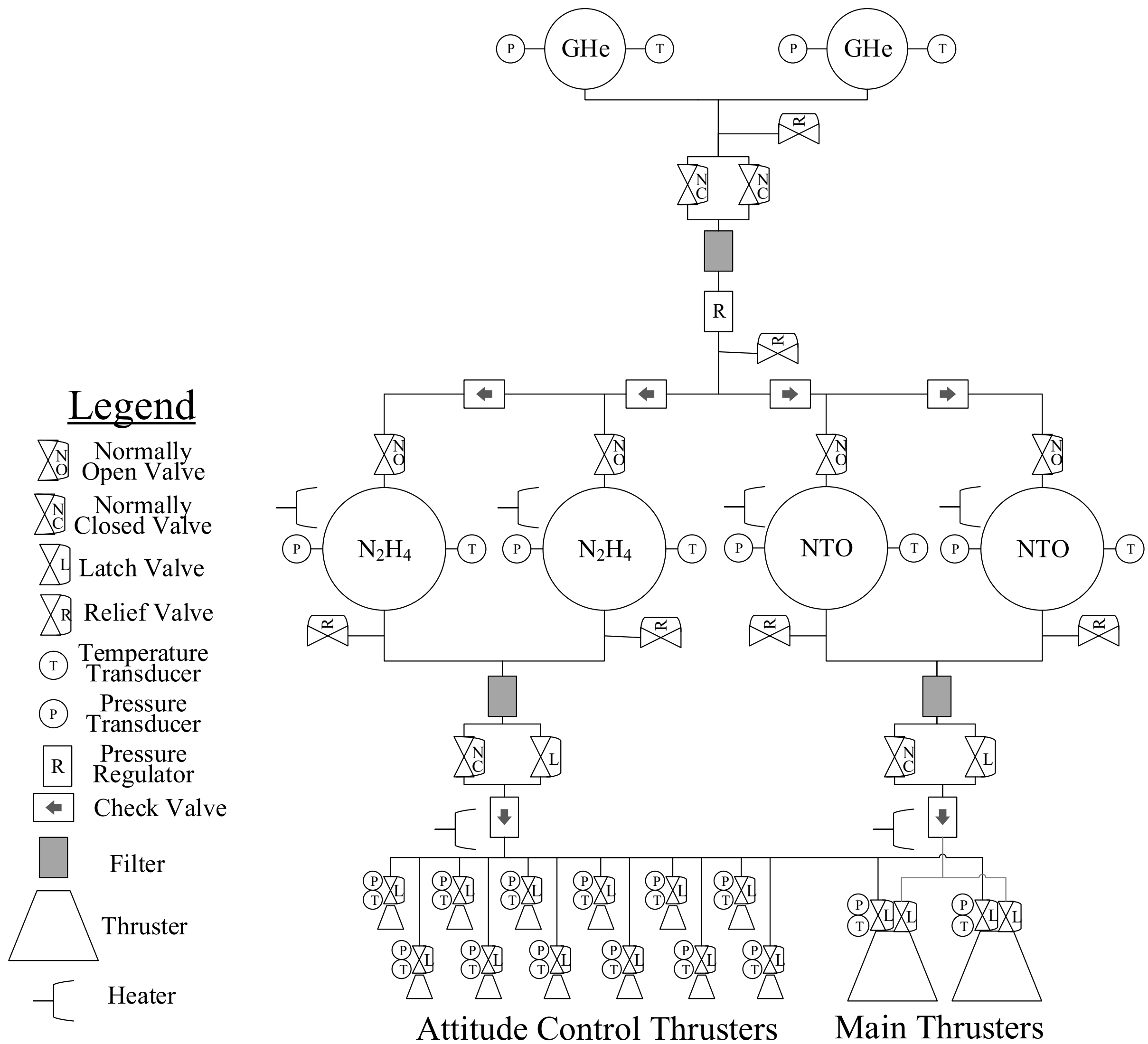


Figure 4-5 Initial Design of the Propulsion System for the Styker-1 Orbiter

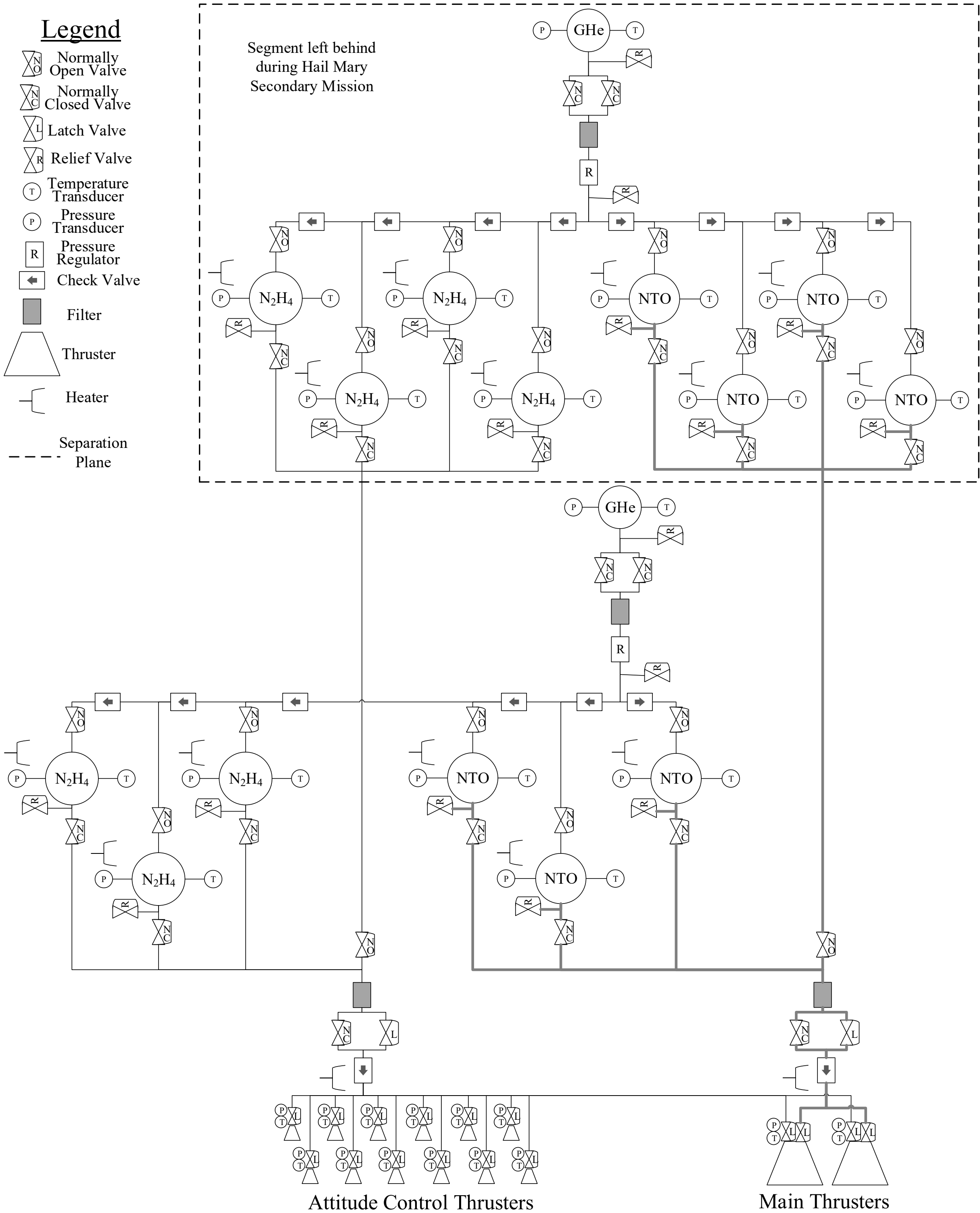


Figure 4-6 Final Design of the Propulsion System for the Styker-1 Orbiter

The mass of propellant of required for the HAIL MARY phase will be 422 kg. The propellant mass was calculated using the rocket equation with ideal condition and the segmented spacecraft empty mass, outlined in Table 4-14 below. The mass of bipropellant of required for the baseline mission will be to 3,755 kg. The bipropellant mass was calculated using the rocket equation with ideal conditions and additional 20% margin, outlined in Table 4-15 below.

Table 4-14 Calculation of Bipropellant Mass for Secondary Mission

Nitrogen Tetroxide & Hydrazine at 26.7°		
I_{sp}	323	s
ρ_{NTO}	1,450	kg/m^3
ρ_{Hyd}	1,021	kg/m^3
Mix ratio	1.42	
ΔV_{total}	1,520	m/s
On-orbit mass	686	kg
Propellant mass	422	kg
m_{NTO}	248	kg
m_{Hyd}	175	kg
Vol_{NTO}	0.171	m^3
Vol_{Hyd}	0.171	m^3

Table 4-15 Calculation of Bipropellant Mass for Primary Mission

Nitrogen Tetroxide & Hydrazine at 26.7°		
I_{sp}	323	s
ρ_{NTO}	1,450	kg/m^3
ρ_{Hyd}	1,021	kg/m^3
Mix ratio	1.42	
ΔV_{total}	3,100	m/s
On-orbit mass	1,885	kg
Propellant mass (includes 20% margin)	3,755	kg
m_{NTO}	2,203	kg
m_{Hyd}	1,552	kg
Vol_{NTO}	1.520	m^3
Vol_{Hyd}	1.520	m^3

The mass of the monopropellant required for attitude control throughout the mission's life to counteract max disturbance torques values will be 84.8 kg, calculated using the rocket equation with ideal conditions, outlined in Table 4-6 below.

Table 4-16 Calculation of Monopropellant Mass

Hydrazine Monoprop at 26.7° C		
I_{sp}	120	s
ρ_{Hyd}	1,021	kg/m^3
ΔV_{total}	52.6	m/s
On-orbit mass (dry)	1,885	kg
Propellant mass	86.2	kg
Vol_{Hyd}	0.084	m^3

The propellant tanks are sized to be 15.5 kg and 15.0 kg for each of the 7 hydrazine tanks and 7 NTO tanks, respectively. The calculations were done assuming COPV and PDMs will be used with an additional growth margin of 20%, outlined in Table 4- below.

Table 4-17 Sizing of the Propellant Tanks (7 of each)

Assumptions			Calculations Using SMAD Fig. 18-9 [7]		
Ullage	3%		Fuel Volume	282	L
Growth Margin	20%		Oxidizer Volume	267	L
Fuel Volume (N_2H_4)	0.229	m^3	Fuel Tank Mass (N_2H_4)	15.7	kg
Oxidizer Volume (NTO)	0.217	m^3	Oxidizer Tank Mass (NTO)	15.1	kg

The 2 pressurant tanks are sized to be 45.8 kg each. The calculations were done assuming COPV and PDMs will be used and the additional growth margin of 20% is included in the assumed total propellant volume, outlined in Table 4- below.

Table 4-18 Sizing of the Pressurant Tanks (2 of each)

Assumptions			Calculation Using SMAD §18.5.3 [7]		
Initial tank pressure	34.4	MPa	Mass of pressurant (He)	17.7	kg
Final pressure ≥	2.8	MPa	Total pressurant volume	0.363	m^3
ρ_{He}	48.68	kg/m^3	Pressurant tank volume	0.182	m^3
Temperature	293	K	COPV Tank Mass	47.0	kg
Gas constant	2,077	J/kg K			
Total propellant volume	3.842	m^3			

The final dry mass of the propulsion system was calculated to be 427 kg dry and 4,286 kg wet and will consume 117 W while the main thruster is in use. The final mass and power summary tables are below in Table 4-10 and Table 4-20.

Table 4-19 Mass Summary Table for the Styker-1 Orbiter's Prolusion System

Propulsion System Mass Summary (kg)	
Propellant (N_2H_4/ NTO)	3,841
Pressurant (He)	17.7
Fuel tanks (7)	110.0
Oxidizer tank (7)	106.0
Pressurant tank (2)	94.0
Main thruster (2)	10.9
Heaters (18)	9.0
Valves (59)	44.8
Filters (3)	0.9
Temperature Transducers (31)	3.1
Pressure Transducers (31)	9.3
Line and fittings	38.8
Total propulsion system mass (dry)	**427**
Total propulsion system mass (wet)	**4,286**

Table 4-20 Power Summary Table for the Styker-1 Orbiter's Prolusion System

Propulsion System Power Summary (W)	
Heaters (18)	18
Latch Valves (2)	53
Main thruster (1)	46
Total propulsion system Power	**117**

4.1.4 Telecommunication

The telecommunication (telecom) system of the Styker-1 orbiter is designed to establish and maintain a connection to the Deep Space Network (DSN) in order to transmit data and receive commands. Additionally, the Stryker-1 will receive data from the two probes in order to relay their data. The major requirement for the Stryker-1's telecom system is established by JPL's RFP, which requires all requested scientific data to be delivered to the customer by December 2040. The delivery date requirement establishes the data transfer rate as the design driver for the telecom system.

The DSN was selected as the receiving ground station due to the distance of the mission being out of the capabilities of other grounds stations such as the STDN or TDRSS. Due to the mission timeline, the 34 m BWG antennas will be utilized because the 70 m antennas will be decommissioned by the time the Styker-1 orbiter will arrive at Triton. However, the proposed replacement to the 70 m antenna is to use four 34 m BWG antennas arrays that will have X-band uplink capabilities and both X- and Ka-band downlink capabilities, while offering similar or better performance as the 70 m antenna [11]. The Styker-1 orbiter is outfitted with both X- and Ka-band to utilize the strength of both frequencies. The Ka-band has the capabilities of providing much higher data rates, but the X-band is more resilient to interferences.

The Styker-1 orbiter will use a 3.6 m parabolic high gain antenna which is sized to be the most efficient when operating on Ka-band, as either increasing or decreasing the diameter of the dish would reduce the data link margin. The transmission power for the telecom system is depended on the power mode of the orbiter at the time, but the default power allotted for transmission is 100 W during transmission modes. The bit error rate error is set at 10^{-6} in order to keep errors to a minimum without requiring a larger E_b/N_0 that would prevent the telecom achieving a positive link margin. A 10 dB margin for downlink with DSN and a max distance between Earth and Triton are conservative assumptions that will allow the calculated data rates can be achieved in non-ideal situations. The achievable data rates are predicted in Table 4- below using data link tables based on the combinations of using X- or KA-band with the four 34 m BWG array or single 34 m BWG antenna.

Table 4-21 Data Link Table for Stryker-1 Downlink, Maximum Data Rates

		Array with Ka-band	Single with Ka-band	Array with X-band	Single with X-band	
1	Frequency (GHz)	32.4	32.4	7.5	7.5	Based on band used
2	Bit error rate	1.00E-06	1.00E-06	1.00E-06	1.00E-06	Assumed
3	Range (km)	4.61E+09	4.61E+09	4.61E+09	4.61E+09	(Farthest point from Earth to Triton)
4	**Symbol rate (ksps)**	**30.5**	**7.2**	**4.1**	**1.0**	**Max based on 10 dB Margin**
5	Transmitter power (dB)	20.0	20.0	20.0	20.0	From Transmitter power
6	Cable loss (dB)	-0.62	-0.62	-0.62	-0.62	Assumed
7	Antenna gain (dBi)	59.1	59.1	46.4	46.4	(Brown, eq 9.27, pg 462)
8	EIRP (dB)	78.5	78.5	65.8	65.8	(5+6+7)
9	Free space path loss (dB)	-315.9	-315.9	-303.2	-303.2	(Brown, eq 9.36, pg 468)
10	Atmospheric attenuation (dB)	-0.15	-0.15	-0.15	-0.15	(10 deg. Brown, Table 9.20, pg 476)
11	Polarization loss (dB)	-0.2	-0.2	-0.2	-0.2	Assumed
12	DSN receiver gain (dBi)	86.0	79.7	73.3	67.0	(Brown, Table 9.15, pg 494)

13	Point loss (dB)	-4.3	-4.3	-0.2	-0.2	(Brown, eq 9.61, pg 475)
14	Receiver cable loss (dB)	-0.001	-0.001	-0.001	-0.001	Assumed
15	Total received power (dB)	-156.0	-162.3	-164.7	-171.0	8+9+10+11+12+13+14
16	Receiver noise temp (K)	18	18	18	18	(@zenith. Brown, Table 9.15, pg 494)
17	System noise density (dB/Hz)	-216.0	-216.0	-216.0	-216.0	(Ex 9.3 Link Tables, Pg 477)
	Carrier Link Performance					
18	Carrier power/total power (dB)	-8.5	-8.5	-8.5	-8.5	(Brown, eq 9.59, pg 474)
19	Carrier power received (dB)	-164.6	-170.8	-173.3	-179.5	(15+18)
20	Carrier noise bandwidth (dB-Hz)	10.8	10.8	10.8	10.8	From Carrier Track Bandwidth
21	Carrier Signal to Noise (dB)	40.7	34.4	32.0	25.7	(19-17-20)
22	Carrier Sig to Noise req by DSN(dB)	10	10	10	10	DSN Required
23	Carrier Margin (dB)	30.7	24.4	22.0	15.7	(21-22)
	Data Link Performance					
24	Data power/total power (dB)	-0.7	-0.7	-0.7	-0.7	(Brown, eq 9.60, pg 474)
25	Data Power received (dB)	-156.7	-163.0	-165.4	-171.7	(15+24)
26	Data symbol rate (dB-Hz)	-44.8	-38.6	-36.2	-29.9	(Ex 9.3 Link Tables, Pg 477)
27	E_b/N_0 achieved (dB)	14.5	14.5	14.5	14.5	(25+26-17)
28	E_b/N_0 required (dB)	4.5	4.5	4.5	4.5	Assumed
29	Data link margin (dB)	10.0	10.0	10.0	10.0	(27-28)

In order to ensure that the data can be delivered on time and without needing ideal condition or needing the proposed array to be completed the mission ConOps was based around the telecom system being able to produce a data rate of 7.2 ksps. A 7.2 ksps data rate will allow the data to be fully delivered by using the DSN for 8 hours a day, 7 days a week, for just over 1 year. The data link table in Table 4- below shows that the Styker-1 orbiter can complete the mission using any of the four combinations of DSN previsions.

Table 4-22 Data Link Table for Stryker-1 Downlink, Required Data Rate

		Array with Ka-band	Single with Ka-band	Array with X-band	Single with X-band	
1	Frequency (GHz)	32.4	32.4	7.5	7.5	Based on band used
2	Bit error rate	1.00E-06	1.00E-06	1.00E-06	1.00E-06	Assumed

3	Range (km)	4.61E+09	4.61E+09	4.61E+09	4.61E+09	(Farthest point from Earth to Triton)
4	Symbol rate (ksps)	7.2	7.2	7.2	7.2	Max based on 10 dB Margin
5	Transmitter power (dB)	20.0	20.0	20.0	20.0	From Transmitter power
6	Cable loss (dB)	-0.62	-0.62	-0.62	-0.62	Assumed
7	Antenna gain (dBi)	59.1	59.1	46.4	46.4	(Brown, eq 9.27, pg 462)
8	EIRP (dB)	78.5	78.5	65.8	65.8	(5+6+7)
9	Free space path loss (dB)	-315.9	-315.9	-303.2	-303.2	(Brown, eq 9.36, pg 468)
10	Atmospheric attenuation (dB)	-0.15	-0.15	-0.15	-0.15	(10 deg. Brown, Table 9.20, pg 476)
11	Polarization loss (dB)	-0.2	-0.2	-0.2	-0.2	Assumed
12	DSN receiver gian (dBi)	86.0	79.7	73.3	67.0	(Brown, Table 9.15, pg 494)
13	Point loss (dB)	-4.3	-4.3	-0.2	-0.2	(Brown, eq 9.61, pg 475)
14	Receiver cable loss (dB)	-0.001	-0.001	-0.001	-0.001	Assumed
15	Total received power (dB)	-156.0	-162.3	-164.7	-171.0	(8+9+10+11+12+13+14)
16	Receiver noise temp (K)	18	18	18	18	(@zenith. Brown, Table 9.15, pg 494)
17	System noise density (dB/Hz)	-216.0	-216.0	-216.0	-216.0	(Ex 9.3 Link Tables, Pg 477)
	Carrier Link Performance					
18	Carrier power/total power (dB)	-8.5	-8.5	-8.5	-8.5	(Brown, eq 9.59, pg 474)
19	Carrier power received (dB)	-164.6	-170.8	-173.3	-179.5	(15+18)
20	Carrier noise bandwidth (dB-Hz)	10.8	10.8	10.8	10.8	From Carrier Track Bandwidth
21	Carrier Signal to Noise (dB)	40.7	34.4	32.0	25.7	(19-17-20)
22	Carrier Sig to Noise req by DSN(dB)	10	10	10	10	DSN Required
23	Carrier Margin (dB)	30.7	24.4	22.0	15.7	(21-22)
	Data Link Performance					
24	Data power/total power (dB)	-0.7	-0.7	-0.7	-0.7	(Brown, eq 9.60, pg 474)
25	Data Power received (dB)	-156.7	-163.0	-165.4	-171.7	(15+24)
26	Data symbol rate (dB-Hz)	-38.6	-38.6	-38.6	-38.6	(Ex 9.3 Link Tables, Pg 477)
27	E_b/N_0 achieved (dB)	20.8	14.5	12.1	5.8	(25+26-17)
28	E_b/N_0 required (dB)	4.5	4.5	4.5	4.5	Assumed
29	**Data link margin (dB)**	**16.3**	**10.0**	**7.6**	**1.3**	**(27-28)**

For the case of an emergency, three 0.65 m low gain antennas (LGA) are placed on the orbiter to give full 2π coverage. A 125 sps connection to the Styker-1 orbiter can be made using the X-band uplink capabilities of an array of 34 m BWG or single 34 m BWG antenna. A small margin 1.2 dB margin is achieved if only a single 34 m BWG antenna is used, but a 7.4 dB margin is possible if an array of 34 m BWG antennas is used. The emergency uplink link table is shown in Table 4- below.

Table 4-23 Data Link Table for Stryker-1 Emergency Uplink

		Array with X-band	Single with X-band	
1	Frequency (GHz)	7.4	7.4	X-band
2	Bit error rate	1.00E-05	1.00E-05	Assumed
3	Range (km)	4.65E+09	4.65E+09	(Farthest point from Earth to Neptune)
4	Symbol rate (ksps)	0.125	0.125	Assumed
5	Transmitter power (dB)	56.0	56.0	From Transmitter power
6	Cable loss (dB)	-0.001	-0.001	Assumed
7	Antenna gain (dBi)	73.1	66.9	From DSN
8	EIRP (dB)	129.2	122.9	(5+6+7)
9	Free space path loss (dB)	-303.2	-303.2	(Brown, eq 9.36, pg 468)
10	Atmospheric attenuation (dB)	-0.15	-0.15	(Assumed at 10 deg. Brown, Table 9.20, pg 476)
11	Polarization loss (dB)	-0.2	-0.2	Assumed
12	S/C antenna gian (dBi)	6	6	Assumed Emergency LGA
13	Point loss (dB)	-0.1	-0.1	(Brown, eq 9.61, pg 475)
14	Receiver cable loss (dB)	-0.62	-0.62	Assumed
15	Total received power (dB)	-169.1	-175.4	(8+9+10+11+12+13+14)
16	Receiver noise temp (K)	225.7	225.7	(@zenith. Brown, Table 9.15, pg 494)
17	S/C antenna tem (K)	100	100	Assumed
18	System noise temp (K)	325.7	325.7	(16+17)
19	System noise density (dB/Hz)	-203.5	-203.5	(Ex 9.3 Link Tables, Pg 477)
	Carrier Link Performance			
20	Carrier power/total power (dB)	-5.3	-5.3	(Brown, eq 9.59, pg 474)
21	Carrier power received (dB)	-174.4	-180.7	(15+20)
22	Carrier noise bandwidth (dB-Hz)	13.0	13.2	From Carrier Track Bandwidth
23	Carrier/noise ratio received (dB)	16.0	9.6	(21-19-22)
24	Carrier/noise ratio req DSN(dB)	10	10	DNS Required
25	Carrier Margin (dB)	6.0	-0.4	(21-22)
	Data Link Performance			
26	Command power/total power (dB)	-1.5	-1.5	(Brown, eq 9.60, pg 474)
27	Command Power received (dB)	-170.6	-176.9	(15+26)
28	Command symbol rate (dB-Hz)	-21.0	-21.0	(Ex 9.3 Link Tables, Pg 477)
29	E_b/N_0 achieved (dB)	11.9	5.7	(27+28-19)
30	E_b/N_0 required (dB)	4.5	4.5	Assumed

31	**Command link margin (dB)**	**7.4**	**1.2**	**(29-30)**

The telecomm system will utilize a set of four gimbals that have a capability of pointing the HGA with 0.0075° of accuracy to allow for X- and Ka-band use. The final mass estimate for the Styker-1 orbiter's telecom system is 156 kg and will use 118 W in an idle state. The final mass and power summary table is below Table 4-.

Table 4-24 Mass and Power Summary Table for the Styker-1 Orbiter's Telecom System

Item	Quantity	Mass (kg)	Power (w)	Line Mass (kg)	Line Power (W)
3.6 m HGA	1	90.54	0	90.54	0
0.65 m LGA	3	0.5	0	1.5	0
Transponder	2	4	12	8	24
Gimbals	4	1.8	27.4	7.2	54.8
Command Detector Unit	2	1	1	2	2
Telemetry Control Unit	1	7.3	5.1	7.3	5.1
Waveguide Transfer Switch	4	0.75	0	3	0
Diplexer	2	1.7	0	3.4	0
Transmission Lines	1	13	0	13	0
X-band TWTA (Sleep)	2	4.9	8	9.8	16
Ka-band TWTA (Sleep)	2	4.9	8	9.8	16
Total				**156**	**118**

4.1.5 Command and Data System

The Stryker-1 orbiter's Command and Data Handling system is primarily responsible for sending commands from Earth to the orbiter, and transferring data from the orbiter to Earth. Communication between Earth and the orbiter occurs by utilizing the Deep Space Network on Earth and the telecommunications system on the orbiter as discussed in section 4.1.4.

The orbiter's command and data system requires processing and data storage. The processor chosen is the DDC SCS750. This processor is known to have high reliability and has a TRL of 9. The single board computer includes 3 processors for triple redundant processing and uses majority voting for its output. The processor can run between 200 and 1,800 million instructions per second (MIPS) and between 7 and 30 W.

Data storage on the orbiter is accomplished by a solid state recorder (SSR) and a digital tape recorder (DTR). The DTR is only used to store the full resolution mapping data. Because of this, the recorder only has to be run once in each direction, from end to end, once to write the data to the tape, and once to read the data from the tape. This minimizes any wear on the DTR and its functionality, and ensures its reliability after 48 years during the

HAIL MARY phase. The SSR is used for various storage needs such as storing the data needed to meet the requirements of JPL's RFP, storing the full resolution data of potential landing sites, and a buffer to store the data read from the DTR during the HAIL MARY phase before it is transmitted.

The SSR chosen for the orbiter is from the Airbus Defense and Space Solid State Recorder Line. The configuration chosen uses non-volatile flash technology, to maintain the data recorded even during power fluctuations and disconnects. The capacity chosen is the 1 Tbit size as to minimize the power and mass of the system.

The DTR to be used on the orbiter will be custom made from current tape technology. The recorder will be able to store over 60 TB, the total storage size of the 1 m^2 resolution mapping data.

A mass and power table for the orbiter's command and data system is shown in Table 4-.

Table 4-25 Mass and Power Summary Table for Stryker-1 Orbiter's C&DS

Item	Mass (kg)	Power (w)
Processor	1.5	30
SSR	6	10
DTR	39	1
Total	**47**	**41**

The data collected by the TIS is in 24 bit color images and 12 bit spectrographic images. On board, the processor will record these images onto the SSR temporarily for further processing, and delete them after processing unless otherwise stated.

All raw data gathered and generated will go through lossless compression and then be put into long term storage. Data will be compressed with the Consultative Committee for Space Data Systems (CCSDS) 121.0-B-1 algorithm, while images will be compressed with the CCSDS 122.0-B-1 algorithm. This compression will reduce the storage space needed for the data down at a 2:1 ratio on average. This compressed data will be then recorded to the DTR for long term storage.

The images generated will be processed to look for potential landing sites for the Atmosphere Probe and the Geyser Probe. Full resolution images of potential landing sites will be kept on the SSR after compression until the entire surface has been mapped. Then the sites will be ranked, and the best sites will be transferred back to Earth via the DSN for final selection.

The images will also be processed to determine if they contain the area that is considered to be a part of the "Geyser Zone". If they are determined by the onboard image processing algorithm to contain images of the "Geyser Zone", then the processor will convert the 24 bit color images to 8 bit black and white images and compress this data with the spectrographic data for that area on the SSR. This data will then be put into in the que to be transmitted back to Earth. The data transmitted back will undergo Reed-Solomon (255, 223) encoding for error correction.

The geyser zone is expected to cover 3% of the planet. This area, combined with the lossless compression, error correction encoding, means that a total of 78,963,097,082 bits, or approximately 9.87 GB, are to be sent back. At the estimated symbol rate of 7.2 kilo-symbols per second on orbit at Triton, and assuming a 1/3 coverage it will take 33 million seconds, or just over 1 year, to send back the required data.

Additional science data from the probes will be sent back to Earth from the orbiter through the DSN. The total size of this data is expected to be less than 1 MB.

During the HAIL MARY phase, the data stored on the DTR will be transferred to the SSR and compressed from 1 m resolution to 10 m resolution before being transmitted back to the DSN. The rate the tape can be read continuously is 1 Gbps. During the HAIL MARY phase's data transfer this transfer rate will be sufficient to transfer all of the 8 bit color images and 12 bit stereoscopic data at 10 m^2 resolution and then about 30% of that data at 1 m resolution for areas of interest.

4.1.6 Power

When initially determining the power source for the orbiter, Figure 4-4 from Spacecraft Vehicle Design by France and Griffin [12] was referenced.

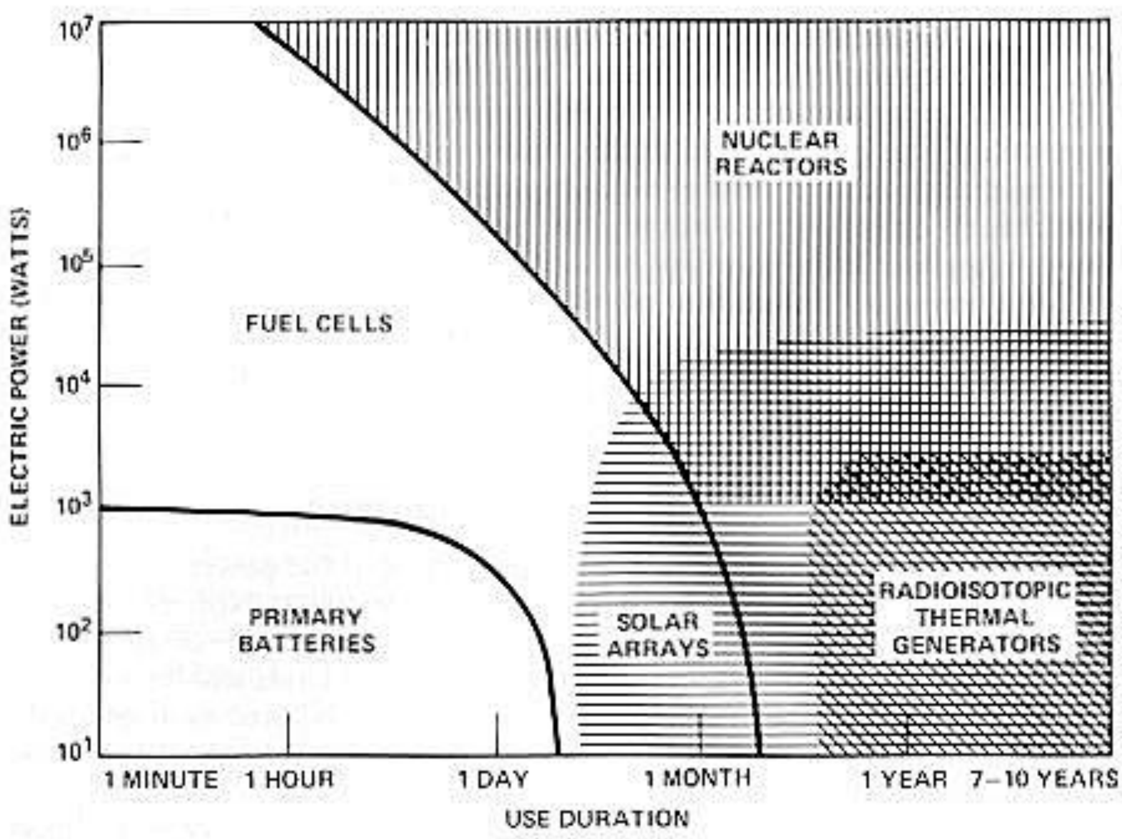


Figure 4-7 Viable Power Sources Given Watts vs. Mission Duration

Figure 4-7 shows viable power sources given mission length and power required. The TAGOS mission is slated to last approximately 18 years to fulfill all RFP requirements, then an additional 29 years to finish the HAIL MARY phase. Some extrapolation, along with reviewing other missions that have operated for similar lengths and required similar power to the TAGOS mission, was required to determine possible choices for a power source. The power source was narrowed down to three options, nuclear, radioisotope thermoelectric generator and solar. Radioisotope thermoelectric generators and solar panels have been used on missions spanning longer than the proposed mission such as the RTGs on Voyager 1 and the solar panels on the Hubble Space Telescope. Nuclear reactors however have not typically been used to power spacecraft in the United States with just one being launched, SNAP-10A, in 1965. While nuclear reactors are planned to be implemented in future NASA spacecraft the current TRL is too low to be implemented in the orbiter since it’s slated for a 2022 launch. This leaves two options available solar cells and radioisotope thermal generators. The mission will be operating on average thirty astronomical units away from the Sun. Using preliminary sizing equations to solve for the mass of the solar panel array required to power the orbiter was greater than the total mass the SLS can take to LEO orbit. The only reasonable power source for the mission is RTGs.

RTGs were first used in spacecraft in dating back to 1961 with SNAP-3B with missions lasting as long as 40 years in the case of Voyager 1. There are two radioisotope thermoelectric generators that are currently in the research phase or being produced now, The Multi-Mission Radioisotope Thermoelectric Generator (MMRTG) and the Advanced Stirling Radioisotope Generator (ASRG) as seen in the Figures 4-5 and 4-6 with their specifications listed in Table 4-26.

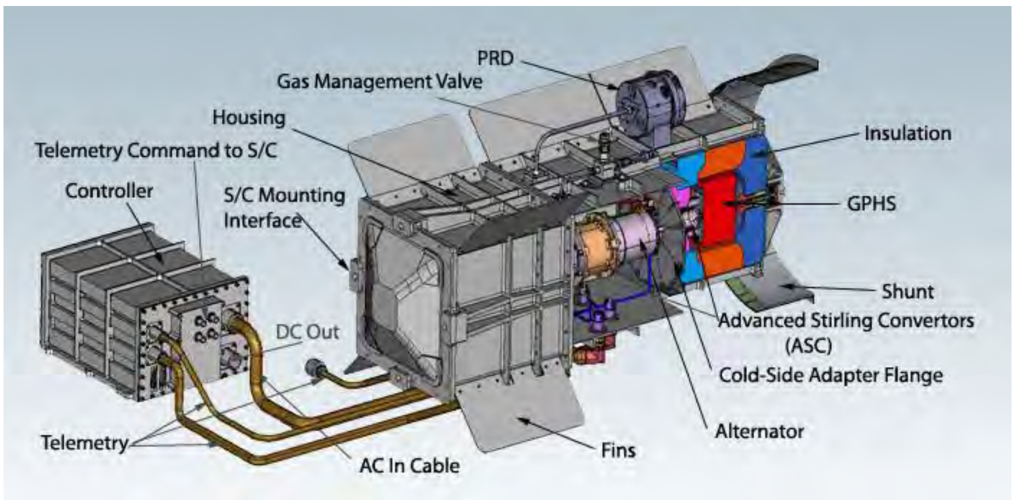


Figure 4-8 Advanced Stirling Radioisotope Generator (ASRG)

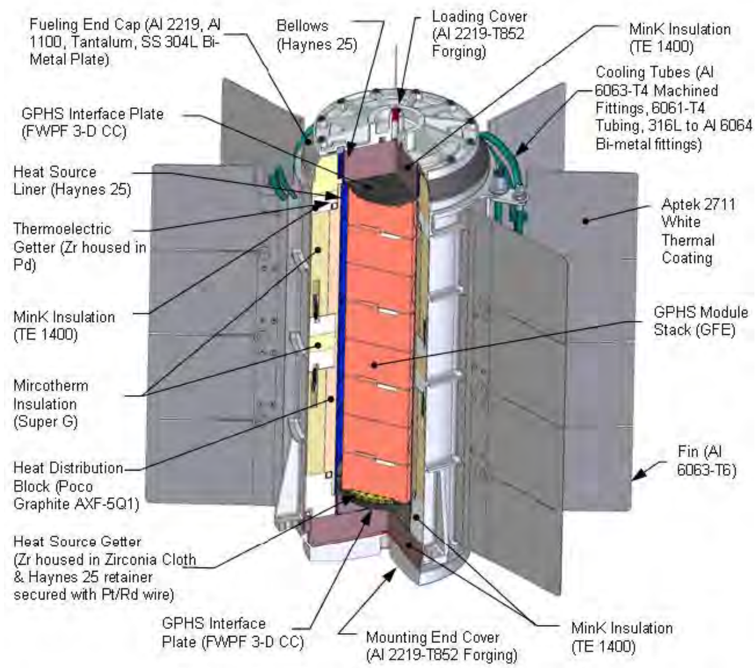


Figure 4-9 Multi-Mission Radioisotope Thermoelectric Generator (MMRTG)

Table 4-26 RTG Trade Study

Model	Power (W)	Power 14 years (W)	Heat (W)	Radioisotope	Fuel (kg)	Mass(kg)
ASRG	140	120	500	^{238}Pu	1	34
MMRTG	110	100	2,000	^{238}Pu	4	45

The United States currently only has enough plutonium for 3 MMRTGs and one has already been committed to Mars 2020 [13] that leaves approximately 8 kilograms of plutonium for other missions this translates to either 2 MMRTGs or 8 ASRGs. While the ASRG has a better power output and uses less plutonium than the MMRTG it was decided to use the MMRTG as the craft's power source. The main reasons for this selection is that the ASRG is currently only an engineering sample and development was halted in 2013 [14] and if restarted it is projected that it wouldn't be ready for launch until 2028 [15]. The launch is slated for 2022 so the low technology readiness of the ASRG prevents it from being chosen as the primary power source. With a total required wattage of 487 the orbiter will require 6 MMRTGs to allow for a 20% contingency. The power system for this mission assumes that we will be able to secure enough plutonium to produce the 6 MMRTGs.

While orbiting Triton the orbiter is assigned with mapping the surface and determining the surface composition using the scientific payload listed in Table 4-27.

Table 4-27 Orbiter Payload Power

Instrument	Power (W)	Description	Purpose
Camera	2.5	Consists of narrow angle camera for high resolution images	Surface mapping
Visible and Infrared Mapping Spectrometer	24	Separates visible and infrared light into its various wavelengths which is analyzed to determine composition	Surface composition
Gimbals	54.8	Two gimbals attached to the VIP to allow tracking of area being observed	Accuracy requirements
Total	**81.3**		

These instruments were chosen due to their low power consumption. The payload combined with the other subsystems listed in Table 4-28 results in the orbiter requiring 487 watts.

Table 4-28 Orbiter Subsystem Power

Subsystem	Power (W)
Thermal Control	80
Attitude Control	35
Power	15
C&DS	41
Comm.	118
Propulsion	117
Subsystem Power	**406**
Payload	81
Total	**487**

The power requirements of the orbiter require 5 MMRTGs but when accounting for a 20% contingency 6 MMRTGs is necessary. At the end of the mission the power generated from all MMRTGs drops to less than 500 W. The 6 MMRTGs also allow for the orbiter to perform scientific operations throughout the journey to Triton. However not all subsystems require full power the entire mission, to deal with this power modes are implemented, these limit power to certain subsystems during certain phases of the mission to lower power needs. The power states have been broken up into six phases of the mission as stated in Table 4-29 below.

Table 4-29 Orbiter Power Modes

	Transit to Neptune	Neptune Capture	Triton Transfer	Triton Orbit	Data Transfer	HAIL MARY
Payload	81	81	81	81	0	0
Propulsion	18	117	117	18	18	7
ACS	35	35	35	35	35	35
C&DS	41	41	41	41	41	41
Telecom.	118	118	118	143	218	218
Thermal	80	80	80	80	80	70
Power	15	15	15	15	15	15
Total	**388**	**487**	**487**	**413**	**407**	**386**

4.1.7 Attitude Control System

The requirements for the orbiter's attitude control system are derived from the RFP science requirements. These requirements are listed in Table 4- below. The first of these requirements is the determination of the orbiter position, attitude, and motion through space. The orbiter must also be actively stabilized for pointing of the narrow angle camera and spectrometer. Lastly, a pointing accuracy of 0.01 degrees is required for those instruments in order to obtain images of specific locations as the orbiter orbits Triton.

Table 4-30 Attitude Control System Requirements

#	Attitude Determination and Control System Requirements
1	Determination of orbiter position, attitude, and motion
2	Actively Stabilized for instrument pointing
3	Pointing accuracy of 0.01 degrees for remote sensing instruments and high gain antenna

The first requirement of the ACS is the determination of the orbiter position and attitude. This is accomplished using various sensors. First, in order to accurately determine the orbiter position and attitude in three dimensions, two reference points are needed. For this purpose, star trackers were selected and placed on the orbiter. The purpose of these sensors is to use the relative positions of two stars in space to determine the orbiter attitude and position. The considered star trackers are shown in Table 4-31 below. Of the three options, the Blue Canyon RH NST was chosen due to having the lowest mass and power consumption. In order to obtain full coverage of the reference stars, a total of 3 of these star trackers are implemented in the orbiter design.

Table 4-31 Star Tracker Selection

Star Tracker	Mass (kg)	Power (W)	Update Rate (Hz)
Blue Canyon RH NST	2	2.5	10
Terma HE 5AS	2.2	7	4
Sodern SED 26	7.17	9.9	10

In addition to star trackers, sun sensors are also included in the ACS. Their purpose is to recalibrate the position of the orbiter in the event that the star trackers to lose their reference, such as the orbiter undergoing an unexpected tumble. The sun sensors are used to get the orbiter reoriented correctly by using the Sun as a reference. The potential sun sensor selections are shown in table Table 4-32 below. The three candidates all have similar masses and power consumption, but the New Space Fine Digital Sun Sensor was chosen due to having the higher accuracy. A total of 6 of these are placed on the orbiter to always have the Sun in view from multiple angles.

Table 4-32 Sun Sensor Selection

Sun Sensor	Mass (kg)	Power (W)	Accuracy (°)	Operating Temp. (° Celsius)
Solar Mems SSOC -D60	0.035	0.35	<0.3	-45 to 85
New Space Fine Digital Sun Sensor	0.035	0.0375	<0.1	-25 to 75
Solar Mems SSOC-A60	0.025	0.036	<0.3	-45 to 85

In order to determine the motion of the orbiter inertial measurement units (IMU) are utilized. These sensors are capable of measuring translational and rotational motion in all three axes. This data is used along with the star trackers to accurately determine the orbiter position and attitude as well as its motion through space. Information from the IMU is also utilized in a feedback control loop in order to increase the accuracy and stability of the attitude control system. Table 4-33 below shows the potential selections for the IMU. The VN-100 IMU was selected due to having significantly lower mass and power consumption than the other options. Although one IMU is capable of measuring motion for all three axes, two are placed on the orbiter for redundancy.

Table 4-33 IMU Selection Trade Study

IMU	Mass (kg)	Rate (Hz)	Power (W)	Operating Temp. (° Celsius)
Vectornav VN 100	0.0035	400	0.185	-40 to 85
Honeywell MIMU	4.7	375	22	-30 to 65
Northrup Grumman LN 200	0.748	400	12	-54 to 71
Memsense MS 3050	0.06	800	9	-40 to 60

The three actuator methods considered for the ACS were thrusters, reaction wheels and control moment gyros. The options were evaluated based on pointing accuracy, maneuver rate, propellant use, and power consumption. The design driver for this selection was the power consumption. The reason for this is falls on the orbiter's necessity to be powered by RTGs. Due to the extended mission length, the radioactive decay of RTG power severely limits the end of life power of the orbiter. In addition to this, limited plutonium availability must be taken into consideration when designing the orbiter. Although reaction wheels and control moment gyros provide more accuracy, the power required to supply either one for all axes of rotation would be over the budgeted power limit. It is for these reasons that a thruster reaction control system was chosen.

Table 4-34 ACS Actuator Method Selection

Actuator Type	Pointing Accuracy (°)	Maneuver Rate	Propellant Use	Power Use (W)
Thruster RCS	0.1-0.5	High	Medium	~10
Reaction Wheel	0.001-0.5	Low	Low	~40
Control Moment Gyro	0.001-0.5	High	Low	~90

The thruster reaction control system pointing accuracy of 0.1 degrees does not meet the required value of 0.01 degrees. This deficiency in the thruster capabilities led to the addition of the Visual Instrument Platform (VIP). This platform utilizes two NEA Electronics G-35 gimbals, each with an accuracy of 0.0075 degrees. This platform can be rotated independent of the orbiter to handle the precision instrument pointing and fulfill the ACS pointing accuracy requirement.

As mentioned in the propulsion system, monopropellant thrusters are selected for the ACS due to their pulsing capabilities. This allows for more accurate maneuvers at the cost of lower I_{sp} as compared to bipropellants. A thruster with a low thrust range is selected to further increase this accuracy. Table 4-35 below depicts the potential candidates for the ACS thrusters. The selected thruster is the Aerojet MR-103G due to having the second highest I_{sp} and a thrust value within the desired range of less than 2 N. In order provide three axis control to the orbiter, 4 thrusters for each axis are utilized for a total of 12 ACS thrusters.

Table 4-35 ACS Thruster Selection

Model	Thrust (N)	Mass (kg)	I_{sp} (s)
Aerojet MR-103D	1.02	0.33	224
Aerojet MR-103G	1.13	0.33	224
Aerojet MR-103M	0.99	0.16	221
Aerojet MR-111C	5.3	0.33	229
Aerojet MR-111E	2.2	0.33	224

In order to determine the propellant requirements of the ACS, the orbital environment of Stryker-1 was analyzed to calculate any disturbances that must be compensated for by the attitude control thrusters. These disturbances include the effect of the gravity gradient from Triton acting on the orbiter, the torque applied from Triton's magnetic field, solar pressure acting on the surface area of the orbiter, and the drag from Triton's atmosphere causing the orbiter to rotate. The results of the calculations are shown in Table 4-. These values were added up and used to calculate the total angular momentum that must be accounted for over the duration of the time in orbit to determine the propellant budget for the ACS system.

Table 4-36 Orbital Disturbances Calculations

Disturbance	Torque Value (Nm per axis)
Gravity Gradient	2.6×10^{-4}
Magnetic Torque	3.9×10^{-5}
Solar Pressure Torque	1.6×10^{-7}
Atmospheric Drag	3.0×10^{-4}

Table 4-37 below summarizes the chosen equipment for the attitude control system. This includes all sensors and thrusters required to fulfill all subsystem requirements. The total line mass for the subsystem was determined to be 13.77 kg with a total line power of 61.1 W with the assumption that only two thrusters fire at once.

Table 4-37 Mass and Power Summary

Item	Quantity	Mass (kg)	Power (W)	Line Mass (kg)	Line Power (W)
Sensors					
RH NST Star Tracker	3	2	2.5	6	7.5
New Space Fine Digital Sun Sensor	6	0.035	0.0375	0.21	0.225
VN 100 IMU	2	0.0035	0.185	0.007	0.37
Controls					
Mr-103D Thruster	12	0.33	13.3	3.96	26.6*
Total				**13.77**	**61.105**

4.2 Atmosphere Probe

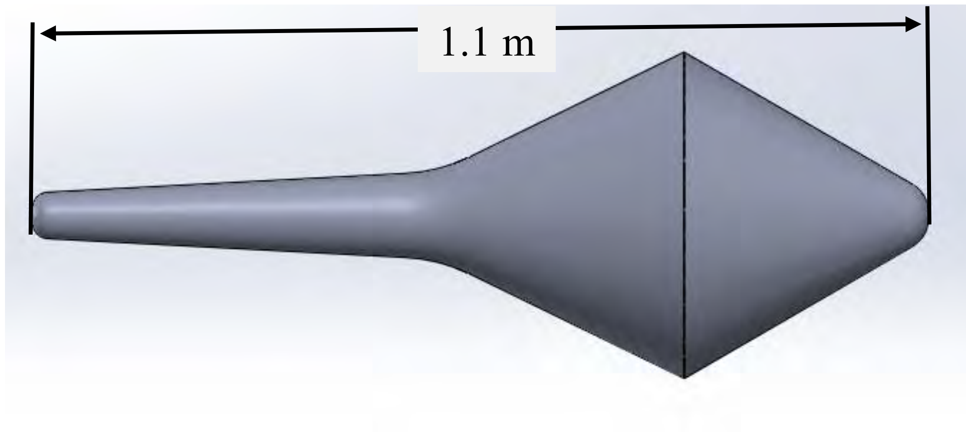


Figure 4-10 Atmosphere Probe CAD Model

The Atmosphere Probe will have a shape very like a spinning top as shown in Figure 4-10 above. This allows for a projectile descent motion to be accomplished. It is equipped with thermal protection, a propulsion system, telecommunications system, command and data system, and a power system. The mass statement for the Atmosphere Probe is shown in Table 4-38 below. Each of these systems are detailed in the following sections.

Table 4-38 Atmosphere Probe Mass Statement

	Mass (kg)
Structure	17
Thermal Control	2
Power	18
C&DS	1.5
Comm.	26
Payload	1.4
Total	**65.9**

4.2.1 Structure

The Atmosphere Probe structure consists of a cone shaped forward frame made of Titanium. The frame is covered with a conical composite panel to protect vital instruments from the descent through Triton's atmosphere. The probe has a forward heavy design to assist with descent and a composite tail for stability. The titanium frame and composite exterior is designed to reduce mass of the probe and overall mass of the architecture.

4.2.2 Thermal

Although the Atmosphere Probe is a lot smaller than the vehicle, it will still need a well thought out thermal analysis. Table 4-39 states the requirements needed to design the thermal control system for the probe.

Table 4-39 Atmosphere Probe Thermal Requirements

Requirement	Description
4.2.2.1-1	Keep probe within operational temperatures when around Triton's atmosphere
4.2.2.1-2	Thermally isolate all instrumentation
4.2.2.1-3	Keep thermal control system for the probe within power and weight limits

The Atmosphere Probe is requires less power compared the orbiter prompting the decision to use batteries as its power source. This will require the use of a heater instead of thermal wattage from an RTG. Table 4- also shows the instrumentation with its respective operational and non-operational temperatures.

Table 4-40 Atmosphere Probe Operational and Non-Operational Temperature Ranges

	Operational Temperature Range °C	Non-Operational Temperature Range °C
Accelerometer	-20 to 50	-20 to 60
Cryogenic Temperature Diode	-20 to 50	-20 to 60
Pressure Transducer	-20 to 50	-20 to 60
Batteries	-10 to 30	-10 to 50

At Triton, the Atmosphere Probe will encounter a surface pressure of approximately 1.4 Pa. MLI does not insulate well at pressures over 1.33 Pa, thus MLI will not be as effective on the probes as it will be on the orbiter. For this a different type of insulation will be necessary. The Atmosphere Probe will still have layers of MLI attached, but it will also have foam insulation on the internal side of the probe to protect it from any outside radiation as well as keeping the instrumentation within operational temperatures throughout its mission. In this case, aerogel seems to be a good candidate as foam insulation for the probe. Aerogel traps gas to lower conductivity and has low density. This type of insulation is also very lightweight, thus contributing to the weight limit of the probe. Approximately 20 RHUs will be installed throughout the probe. These heating units generate heat through radioactive decay and will each produce 1 W of heat. This amount will be satisfactory in being able to keep the instrumentation within operating temperatures [5].

4.2.3 Telecommunication

The telecommunication system of Atmosphere Probe is designed to establish and maintain a connection to the Styker-1 orbiter in order to transmit data and receive commands. The major requirement for the Atmosphere Probe's telecom system is be able to transmit all its scientific data before impact. The Atmosphere Probe's mission profile requires the telecom system to be able to establish a connection no matter the probe's orientation since there is no active attitude control system to control the probe.

The Atmosphere Probe will utilize three S-band patch antennas positioned to give the telecom system omnidirectional properties. The Styker-1 orbiter will point its 3.6 m HGA towards the Atmosphere Probe during its descent in order maintain a connection. The transmission power for the telecom system is limited due to the power system's capabilities, but the power allotted for transmission is 5 W per antenna, 15 W total. The bit rate error is set at 10^{-6} in order to keep errors to a minimum.

The relatively small distances the signals have to travel means using S-band patch antennas are able to provide the required 16.4 ksps with a large margin of 61.6 dB at the max distance of Triton's surface to the Striker-1's orbit. The achievable data rates are predicted using data link tables at the max distance of 502 km, shown in Table 4- below.

Table 4-41 Data Link Table for Atmosphere Probe Downlink

	S-band		
1	Frequency (GHz)	2.1	S-band
2	Bit error rate	1.00E-06	Assumed
3	Range (km)	502	(Farthest point from Orbiter to Probe)
4	Symbol rate (ksps)	16.4	Required
5	Transmitter power (dB)	7.0	From Transmitter power
6	Cable loss (dB)	-0.62	Assumed
7	Antenna gain (dBi)	5.0	1 5W Patch Antenna
8	EIRP (dB)	11.4	(5+6+7)
9	Free space path loss (dB)	-152.9	(Brown, eq 9.36, pg 468)
10	Atmospheric attenuation (dB)	-0.15	(Assumed at 10 deg. Brown, Table 9.20, pg 476)
11	Polarization loss (dB)	-0.2	Assumed
12	Receiver gain (dBi)	35.4	From Orbiter Data
13	Point loss (dB)	0.0	(Brown, eq 9.61, pg 475)
14	Receiver cable loss (dB)	-0.62	Assumed
15	Total received power (dB)	-107.2	(8+9+10+11+12+13+14)
16	Receiver noise temp (K)	18	(@zenith. Brown, Table 9.15, pg 494)

17	System noise density (dB/Hz)	-216.0	(Ex 9.3 Link Tables, Pg 477)
	Carrier Link Performance		
18	Carrier power/total power (dB)	-8.5	(Brown, eq 9.59, pg 474)
19	Carrier power received (dB)	-115.7	(15+18)
20	Carrier noise bandwidth (dB-Hz)	10.8	From Carrier Track Bandwidth
21	Carrier Signal to Noise (dB)	89.6	(19-17-20)
22	Carrier Sig to Noise req(dB)	10	
23	Carrier Margin (dB)	79.6	(21-22)
	Data Link Performance		
24	Data power/total power (dB)	-0.7	(Brown, eq 9.60, pg 474)
25	Data Power received (dB)	-107.8	(15+24)
26	Data symbol rate (dB-Hz)	-42.1	(Ex 9.3 Link Tables, Pg 477)
27	E_b/N_0 achieved (dB)	66.1	(25+26-17)
28	E_b/N_0 required (dB)	4.5	Assumed
29	Data link margin (dB)	61.6	(27-28)

The final mass estimate for the Atmosphere Probe's telecom system is 26 kg and will use 29 W during descent stage. The final mass and power summary table is in Table 4- below.

Table 4-42 Mass and Power Summary Table for the Atmosphere Probe Telecom System

Item	Quantity	Mass (kg)	Power (w)	Line Mass (kg)	Line Power (W)
S-Band Antenna	3	0.5	0	1.5	0
Transponder	2	3.5	4	7	8
Telemetry Control Unit	1	3.6	2.6	3.6	2.6
Transmission Lines	1	3	0	3	0
SSPA (Active)	3	3.5	6	10.5	18
Total				**26**	**29**

4.2.4 Command and Data System

The Atmosphere Probe will use the same processor as the orbiter, the DDC SCS750 in a low power mode. This processor is known to have high reliability and has a TRL of 9. The single board computer includes 3 processors for triple redundant processing and uses majority voting for its output. The processor can run between 200 and 1,800 million instructions per second (MIPS) and between 7 and 30 W and weighs 1.5 kg. The data collected from the probe's instruments is transmitted in real time along with other telemetry data. The Atmosphere

Probe will transmit up to 10 data points every 100 milliseconds with the expected symbol rate of the telecommunications subsystem of 16.4 ksps.

4.2.5 Power

The Atmosphere Probe when attached to the main craft and is integrated into the main power system and receive any necessary power from the MMRTGs. Since the Atmosphere Probe operates for only 10 minutes, battery power was chosen rather than include another MMRTG. The Atmosphere Probe payload selection is as follows in the Table 4-43 below.

Table 4-43 Atmosphere Probe Payload Power

Instrument	Power (W)	Description	Purpose
Accelerometer (Omega)	32	Finds Drag force on probe using descent acceleration	Atmospheric density
Cryogenic temperature diode (Omega)	2.5	Used for Determining temperature in cryogenic conditions	Temperature of atmosphere
LP 1400 Pressure Transducer	3.5	Measures pressure during descent	Atmospheric pressure
Total	**38**		

The instrumentation for the Atmosphere Probe was chosen due to its low power consumption and ability to measure the low temperatures that we estimate the probe to encounter. In Table 4-44 below, the power of the all subsystems necessary to have the Atmosphere Probe complete its mission are listed.

Table 4-44 Atmosphere Probe Power Statement

Subsystem	Power (W)
Thermal Control	35
Attitude Control	25
Power	5
CDS	21
Comm.	29
Propulsion	0
Mechanisms	3
Subsystem Power	**118**
Payload	3.5
Total	**121.5**

Since the Atmosphere Probe only uses power during its descent and ceases to operate upon hard landing the probe only needs to be powered for the ten-minute descent and a twenty-minute start up period. With a typical voltage of 28 volts, a depth of discharge of 1 and an assumed battery efficiency of 97% the required capacity of the battery is 2.5 amp hours with a 20% contingency. A preliminary trade study was conducted to select the best option for the battery for the Atmosphere Probe, the results are listed in table 4-45 below.

Table 4-45 Atmosphere Probe Battery Selection Trade Study

Cells	Voltage	Capacity (Ah)	Mass (kg)	# Series	# Parallel	1 Off Contingency	Total #	Total Mass (kg)
LSE50	3.7	50	1.5	8	1	2	16	24
LSE100	3.7.	100	2.79	8	1	2	16	44.64
LSE175	3.7	175	4.65	8	1	2	16	74.4
P/N 31771	28.8	7	6.35	1	1	2	2	12.7

The best option is it utilize a premade battery, the P/N 31771 made by Space Vector, with a capacity of 7 amp hours and a nominal voltage of 28.8. The premade battery was chosen rather than assembling individual cells as it was found that typically space batteries cells were rated at higher amp hours but low voltage requiring 8 in series to achieved the desired voltage and with an additional string for redundancy the weight of the assembled cells ended but being more than the premade battery we have selected. A premade battery would allow the probe to have low amp hours while keeping the weight down. Another battery is included to increase the redundancy of the power system.

4.3 Geyser Probe

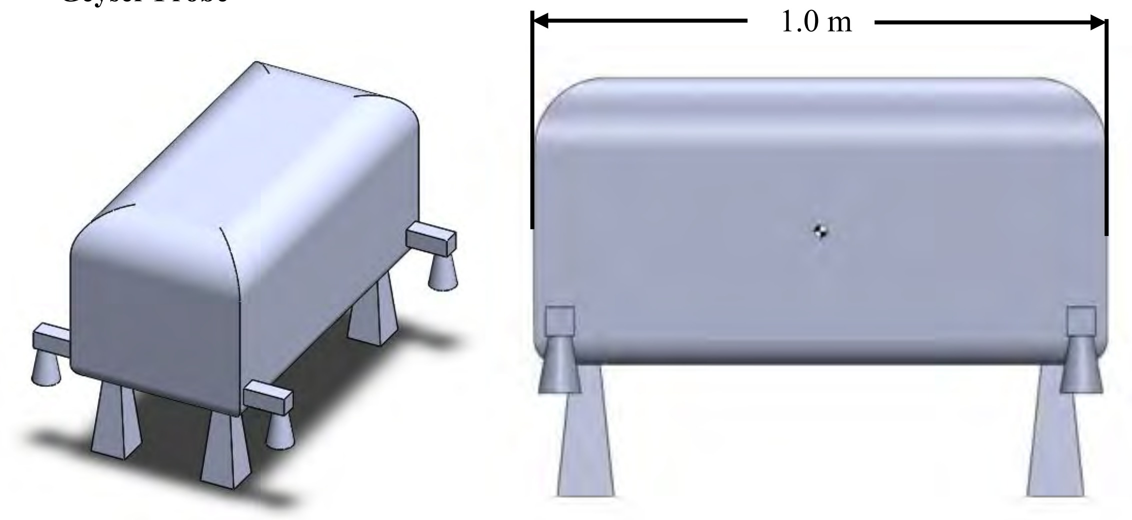


Figure 4-11 Geyser Probe CAD Model

The Geyser Probe, shown in Figure 4-11 above, is a soft-landing probe. It is equipped with thermal protection, a propulsion system, telecommunication system, command and data system, power system, and an attitude control system. The masses of these systems are detailed in Table 4-46 below. Each of these system's masses are described below.

Table 4-46 Geyser Probe Mass Statement

	Mass (kg)
Structure	29
Thermal Control	3
ACS	1
Power	43
C&DS	1.5
Propulsion	20
Comm.	22
Payload	11
Total	**130.5**

4.3.1 Structure

The Geyser Probe will consist of a titanium chest like top frame covered by composite panels to protect the internals. There are four landing struts placed at the corners of the chest and a suspension system to help with the impact of landing at Triton. The attitude control thrusters are mounted at the ends of the titanium and composite body. Once again, the titanium and composite combination is used to reduce probe mass and overall architecture mass.

4.3.2 *Thermal*

The Geyser Probe has similar thermal requirements as the Atmosphere Probe. Table 4- shows those requirements.

Table 4-47 Geyser Probe Thermal Requirements

Requirement	Description
4.3.2.1-1	Keep probe within operational temperatures when around Triton's atmosphere
4.3.2.1-2	Thermally isolate all instrumentation
4.3.2.1-3	Keep thermal control system for the probe within power and weight limits

Table 4-48 shows the instrumentation that will be on the Geyser Probe and its respective temperature limits.

Table 4-48 Geyser Probe Operational and Non-Operational Temperature Ranges

	Operational Temperature Range °C	Non-Operational Temperature Range °C
Mass Spectrometer	-20 to 50	-20 to 60
Dust Detection System	-15 to 40	-15 to 50
Batteries	-10 to 30	-10 to 50

The Geyser probe is entering Triton's atmosphere, thus MLI will still be need, but will not be as effective as it will be on the orbiter. Aerogel is used on the Geyser Probe as foam insulation. The Aerogel insulation is implemented on the internal part of the probe, thus keeping the instrumentation thermally insulated. 15 RHUs are implemented throughout the probe, helping to keep the probe within operational temperatures throughout its mission. Since the Geyser Probe contains less volume than the Atmosphere Probe, it will not be requiring as many RHUs as the Atmosphere Probe. Each RHU will generate 1 W of heat onto the probe thus helping the probe be within operational temperature [5].

4.3.3 *Propulsion*

The design process of the propulsion system for the Geyser Probe began with defining the requirements for the propulsion system in order to ensure all mission needs are met. The first major requirement is the propulsion system has to fulfill the ΔV of 750 m/s for the descent and landing processes. The second major requirement is the propulsions system has to accommodate the needs of the ACS system since main thrusters will be used for 3-axis attitude control as well.

A trade study between the various types of propulsion system types was performed in order to select the most appropriate system for meeting mission requirements. Ultimately, a liquid monopropellant was selected in order to meet the mission requirements without the need of a complex system. A summary table of the trade study is listed in Table 4- below.

Table 4-49 Trade Study for Type of Propulsion System [6]

Requirement	Monopropellant	Bipropellant	Solid
I_{sp} (s)	165-260	300-450	<300
Impulse Range (N-s)	<45000	>45000	>45000
Restart	Yes	Yes	No
Pulsing	Yes	No	No

Ion engines and cold gas are not applicable to the mission because of high thrust requirements for the descent burn. Solid motors were not selected due to lack of restart capability which is required since the probe will use the main thrusters as ACS thrusters as well and the landing conditions are not set. The advantage of a monopropellant system is that the complexity of the system can be kept low to minimize the chance of failure.

The propellant that was selected for the Geyser Probe's propulsion system is hydrazine. Hydrazine was chosen as the fuel due to its proven heritage as a monopropellant and simple storability requirements compared to cryogenic fuels.

A trade study between several large thrusters was done in order to select the engine for the Geyser Probe, shown in Table 4-50 below. Aerojet's MR-104A/C was selected due to having the highest I_{sp} of the flight proven thrusters. The Geyser Probe will implement 4 gimbaled MR-104A/C thrusters, which will function as the main thruster and ACS. Four thrusters allow for the mission ConOps to be carried out even if one thruster fails.

Table 4-50 Trade Study of N_2H_4 Thrusters

Manufacture	Model	Thrust (N)	Mass (kg)	Diameter (m)	Height (m)	I_{sp} (s)	Heritage
Aerojet [16]	**MR-104A/C**	**572.5**	**1.86**	**0.152**	**0.332**	**239**	**Flight Proven**
Aerojet [16]	MR-107S	360	1.01	0.0645	0.22	236	TRL 6
Moog-ISP [17]	MONARC-90HT	116	1.12	0.084	0.3	234	Flight Proven
Moog-ISP [17]	MONARC-445	445	1.6	0.148	0.41	234	Flight Proven

Figure 4-12 Geyser Probe's Thruster (Aerojet MR-104A/C) [16]

The Styker-1 orbiter's propulsion system will implement a blow-down system. A blow-down system was selected to keep the system simple. A pressure regulated system was not selected due to its complex nature and is not needed since a bipropellant is not being used. Helium is the selected pressurant as it is the most mass efficient choice of the inert gases. A diaphragm system will be used over PDMs for the fuel tank design, since diaphragm systems are generally simpler. A solid Titanium tank will be used to store the fuel and pressurant. The propulsion system for the Geyser Probe is in a line diagram below in Figure 4-13.

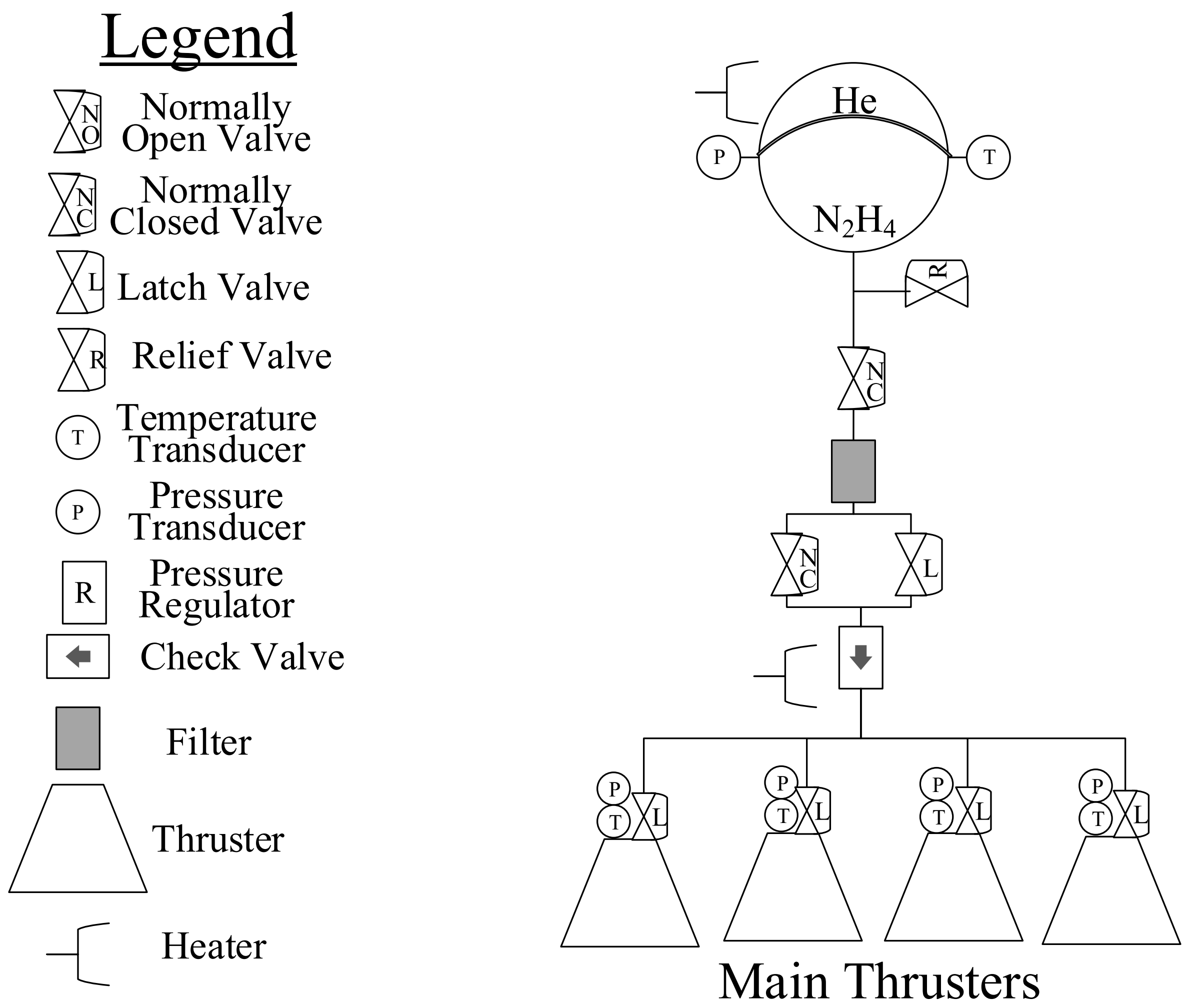


Figure 4-13 Design of the Propulsion System for the Geyser Probe

The Geyser Probe performs a deceleration burn that lasts 43 seconds and provides a ΔV of 750 m/s. The mass of propellant required for the descent mission is 80.4 kg. The propellant mass was calculated using the rocket equation with ideal conditions and additional 20% margin, outlined in Table 4-51 below.

Table 4-51 Calculation of Monopropellant Mass for the Geyser Probe

Hydrazine Monoprop at 26.7° C		
I_{sp}	222	s
ρ_{Hyd}	1,021	kg/m^3
t_b	43	s
ΔV_{total}	750	m/s
On-orbit mass (dry)	163	kg
Propellant mass	80.4	kg
Vol_{Hyd}	0.079	m^3

The propellant tank is sized to be 7.2 kg. The calculations were done assuming solid titanium and diaphragm system will be used with an additional growth margin of 20%, outlined in Table 4- below.

Table 4-52 Sizing of the Propellant Tank for the Geyser Probe

Assumptions			**Calculations Using Brown's Ex. 4.8 [6]**		
Ullage	3%		$\sigma_{allowable}$	345	MPa
Propellant volume	0.079	m^3	Internal volume	0.089	m^3
Max tank pressure	3,300	kPa	Inner tank radius	0.277	m
$\rho_{Titanium}$	4,430	kg/m^3	Tank thickness	1.325	mm
σ_{yield} (Titanium)	690	MPa	Outer tank radius	0.278	m
Safety factor	2		Membrane mass	5.684	kg
			Girth land mass	1.026	kg
			Penetration land mass	0.207	kg
			Supported mass	6.918	kg
			Structural attachment mass	0.138	kg
			Tank shell mass	7.06	kg
			Diaphragm volume	1914	cm^3
			Diaphragm mass	0.191	kg
			Total tank assembly mass	**7.2**	**kg**

The pressurant required is .04 kg of helium with the calculations outlined in Table 4- below.

Table 4-53 Calculation of Required Pressurant for the Geyser Probe

Assumptions		
Initial tank pressure	34.4	MPa
Final pressure≥	2.8	MPa
ρ_{He}	48.68	kg/m^3
Temperature	293	K
Gas constant	2,077	J/kg K
Calculation Using SMAD §18.5.3 [7]		
Mass of pressurant (He)	0.4	kg
Total pressurant volume	0.008	m^3

The final dry mass of the propulsion system was calculated to be 20.2 kg dry and 101 kg wet and consumes 177 W while the main thrusters are in use. The final mass and power summary tables are below in Table 4- and Table 4-.

Table 4-54 Mass Summary Table for the Geyser Probe's Prolusion System

Geyser Probe Propulsion System Mass Summary (Kg)	
Propellant	80
Pressurant	0.4
Propellant Tank	7.2
Thruster (4)	4.0
Heaters (2)	1.0
Valves (5)	3.8
Filter	0.3
Temperature Transducers (5)	0.5
Pressure Transducers (5)	1.5
Line and fittings	1.8
Total propulsion system mass (dry)	**20.2**
Total propulsion system mass (wet)	**101**

Table 4-55 Power Summary Table for the Geyser Probe's Prolusion System

Probe Propulsion System Power Summary (W)	
Heaters (2)	2
Latch Valve (1)	8.5
Catalyst Bed Heater (4)	52.8
Thrusters (4)	114
Total propulsion system Power	**177**

4.3.4 Telecommunication

The telecommunication system of Geyser Probe is designed to establish and maintain a connection to the Styker-1 orbiter in order to transmit data and receive commands. The major requirement for the Geyser Probe's telecom system is be able to transmit all its scientific data before power is depleted. The Geyser Probe's mission profile requires the telecom system to be able to establish repeated connections to Styker-1 orbiter as is passes overhead during its mission duration.

The Geyser Probe utilizes an S-band patch antenna with a redundant S-band patch antenna to prevent a single point failure. The Styker-1 orbiter points its 3.6 m HGA towards the Geyser Probe when overhead in order to establish a connection. The transmission power for the telecom system is limited due to the power system's capabilities, but the power allotted for transmission is 5 W per antenna, 10 W total. The bit error rate error is set at 10^{-6} in order to keep errors to a minimum.

The relatively small distances the signals have to travel means using S-band patch antennas are able to provide the required 16.4 ksps with a large margin of 61.6 dB at the max distance of Triton's surface to the Striker-

1's orbit. The achievable data rates are predicted using data link tables at the max distance of 502 km in Table 4- below.

Table 4-56 Data Link Table for Geyser Probe Downlink

	S-band		
1	Frequency (GHz)	2.1	S-band
2	Bit error rate	1.00E-06	Assumed
3	Range (km)	502	(Farthest point from Orbiter to Probe)
4	Symbol rate (ksps)	16.4	Required
5	Transmitter power (dB)	7.0	From Transmitter power
6	Cable loss (dB)	-0.62	Assumed
7	Antenna gain (dBi)	5.0	1 5W Patch Antenna
8	EIRP (dB)	11.4	(5+6+7)
9	Free space path loss (dB)	-152.9	(Brown, eq 9.36, pg 468)
10	Atmospheric attenuation (dB)	-0.15	(Assumed at 10 deg. Brown, Table 9.20, pg 476)
11	Polarization loss (dB)	-0.2	Assumed
12	Receiver gain (dBi)	35.4	From Orbiter Data
13	Point loss (dB)	0.0	(Brown, eq 9.61, pg 475)
14	Receiver cable loss (dB)	-0.62	Assumed
15	Total received power (dB)	-107.2	(8+9+10+11+12+13+14)
16	Receiver noise temp (K)	18	(@zenith. Brown, Table 9.15, pg 494)
17	System noise density (dB/Hz)	-216.0	(Ex 9.3 Link Tables, Pg 477)
	Carrier Link Performance		
18	Carrier power/total power (dB)	-8.5	(Brown, eq 9.59, pg 474)
19	Carrier power received (dB)	-115.7	(15+18)
20	Carrier noise bandwidth (dB-Hz)	10.8	From Carrier Track Bandwidth
21	Carrier Signal to Noise (dB)	89.6	(19-17-20)
22	Carrier Sig to Noise req(dB)	10	
23	Carrier Margin (dB)	79.6	(21-22)
	Data Link Performance		
24	Data power/total power (dB)	-0.7	(Brown, eq 9.60, pg 474)
25	Data Power received (dB)	-107.8	(15+24)
26	Data symbol rate (dB-Hz)	-42.1	(Ex 9.3 Link Tables, Pg 477)
27	E_b/N_0 achieved (dB)	66.1	(25+26-17)
28	E_b/N_0 required (dB)	4.5	Assumed
29	Data link margin (dB)	61.6	(27-28)

The final mass estimate for the Atmosphere Probe's telecom system is 22 kg and will use 23 W during descent stage. The final mass and power summary table is in Table 4- below.

Table 4-57 Mass and Power Summary Table for the Geyser Probe Telecom System

Item	Quantity	Mass (kg)	Power (w)	Line Mass (kg)	Line Power (W)
S-Band Antenna	2	0.5	0	1	0
Transponder	2	3.5	4	7	8
Telemetry Control Unit	1	3.6	2.6	3.6	2.6
Transmission Lines	1	3	0	3	0
SSPA (Active)	2	3.5	6	7	12
Total				**22**	**23**

4.3.5 Command and Data System

The Geyser Probe will use the same processor as the orbiter, the DDC SCS750 in a low power mode. This processor is known to have high reliability and has a TRL of 9. The single board computer includes 3 processors for triple redundant processing and uses majority voting for its output. The processor can run between 200 and 1,800 million instructions per second (MIPS) and between 7 and 30 W and weighs 1.5 kg. The data collected from the probe's instruments is transmitted in real time along with other telemetry data. The Geyser probe is able to transmit up to 10 data points every 100 milliseconds with the expected symbol rate of the telecommunications subsystem of 16.4 ksps.

4.3.6 Power

During the transit to Neptune the Geyser Probe is attached to the main craft and is integrated into the main power system and receives any necessary power from the MMRTGs. The Geyser Probe has a mission life of 24.5 hours so the addition of a battery rather than an additional MMRTG was chosen. The Geyser Probe has the following instruments using power during its surface mission.

Table 4-58 Geyser Probe Payload Power

Instrument	Power (W)	Description	Purpose
Mass Spectrometer*	24	Determines composition of captured objects	Atmosphere and plume composition
Dust Detection System*	5	Analyzes captured dust particles	Particle size & density
Total	**29**		

Like all other instruments for this mission the Geyser Probe's instrumentation was chosen for their lower power usage while still providing a high enough sampling and data rate to ensure enough data in its 24-hour mission. The total power required for the Geyser Probe is listed in the table below.

Table 4-59 Geyser Probe Power Statement

Subsystem	Geyser Probe (W)
Thermal Control	21
Attitude Control	1
Power	5
CDS	13
Comm.	29
Propulsion	177
Mechanisms	1
Subsystem Power	**247**
Payload	29
Total	**276**

The Geyser Probe operatings for 24.5 hours requiring 276 watts while on the surface taking data. With a typical voltage of 28 volts, a depth of discharge of 1 and an assumed battery efficiency of 97% results in a battery capacity of 105 amp hours with a 20% contingency. A preliminary trade study was conducted to select the best option for the battery for the probe, the results can be seen in the table below.

Table 4-60 Geyser Probe Battery Selection Trade Study

Cells	Voltage	Capacity (Ah)	Mass (kg)	# Series	# Parallel	1 Off Contingency	Total #	Total Mass (kg)
LSE50	3.7	50	1.5	8	1	2	16	24
LSE100	3.7.	100	2.79	8	1	2	16	44.64
LSE175	3.7	175	4.65	8	1	2	16	74.4
VL51ES	3.6	51	1.08	8	3	4	32	12.7

The best option for the Geyser Probe is to construct a battery from commercially available cells. It was determined that VL51ES made by Saft is the best option for the Geyser Probe. With a nominal voltage of 3.6 and a capacity of 51 amp hours the Geyser Probe requires 8 cells in series with three of these strings in parallel. Another string is integrated into the design to increase the redundancy and make total failure less likely.

4.3.7 Attitude Control System

The purpose of the Geyser Probe attitude control system is to control the descent of the probe from the moment it leaves the orbiter until it reaches the ground. This is done by utilizing the four gimballed thrusters to actively stabilize the probe as it falls In order to accomplish this, an IMU sensor is used to determine the motion of the probe. This data is used with a feedback control loop to send commands to the gimbal throughout the descent. Similar to the orbiter, two VN-100 IMUs will be used for redundancy.

5 Management

The projected cost for the TAGOS mission is determined to be below the required $5 billion in 2017 dollars at $3.96 billion. This includes the cost for development, and operations. A complete cost breakdown has been placed in Appendix B.

5.1 Projected Mission Cost

In determining the cost of the mission the first Rough Order Magnitude (ROM) estimate is made using a top-down process which relies on broad design concepts and subsystem-level design parameters. The cost estimating tools are based on cost estimating models and normalized historic databases from Wertz [7]. Table 5- below is a cost break down estimate for the TAGOS mission.

Table 5-1 Total Cost Estimate and Standard Deviation for TAGOS Mission

	Cost (FY17$K)	Std Error ($K)
Total Cost of Deployment (Includes LV)	$1,720,342	$141,160
Upper stage	$28,689	
Propellant	$70	
Dormant Operations	$103,706	
Active Operations	$80,206	
DSN	$720,042	
Phase A-D Reserves (50%) (NASA Space Flight Program & Project Management Handbook)	$874,551	$70,580
Phase E-F Reserves (25%) (NASA Space Flight Program & Project Management Handbook)	$225,989	$-
Total	**$3,753,595**	$211,741

The total cost of deployment is projected using three models: Large Satellite Cost Model, Unmanned Spacecraft Cost Model Version 8.0 (USCM8) and NASA Instrument Cost Model (NICM). Table 5- below shows what is included in determining the cost for the total cost of deployment. A correlation between mass and cost has been shown that an increase of mass will result in an increase of cost, so, spacecraft bus cost was determined using the mass of each subsystem.

Table 5-2 Total Cost of Deployment Breaking

Cost Component
❖ **Spacecraft Bus**
➢ Structure and Thermal
➢ ACS
➢ Electrical Power System
➢ Propulsion
➢ TT&C and Data Handling
➢ Integration, Assembly, & Test (IA&T)
➢ Flight Software
❖ **Payload**
➢ Scientific Payload
▪ Fabrication
▪ Management
▪ Systems Engineering
▪ Product Assurance
▪ Integration and Test
❖ **Launch Segment**
❖ **Program Level**
❖ **Launch and Orbital Ops Support (LOOS)**
❖ **Ground Support Equipment (GSE)**
10% Contractor Fee

Dormant operations represent the time when the Stryker-1 is in route to Triton. Active operations is when the orbiter is transmitting data back to Earth during scientific operations at Triton and during the HAIL MARY phase. Table 5- below uses Wertz's [7] Operation Cost Model to calculate both operation modes.

Table 5-3 Dormant/Active Operations Breakdown

Operations
❖ PMSE
❖ Space Segment Software Maintenance
❖ Ground Segment
➢ Mission Operations
➢ Ground Segment Software Maintenance
➢ Ground Hardware Maintenance
➢ Facilities

To determine the cost for using the DSN for the mission, the Aperture Fee Tool DSN Mission Support Cost for New Frontiers Tool was used with the entered parameters as shown in Table 5- below.

Table 5-4 NASA's DSN Aperture Fee Tool for DSN Mission Support Cost

No (#)	Support Period Name (description)	Antenna Size (meters)	Service Year (year)	Hours per Track (hours)	No. Tracks per Week (# tracks)	No. Weeks Required (# weeks)
1	Dormant Operations	34BWG	2022	2	7.0	52.0
2	Scientific Operations @ Triton	4 BWG Array	2022	8	7.0	52.0
3	HAIL MARY phase	4 BWG Array	2035	4	1.0	52.0
4	Scientific Operations on Return	4 BWG Array	2035	12	8.0	52.0

The 50% Phase A-D reserves, and 25% phase E-F reserves are added to the architecture-level cost estimates per NASA Decadal Survey ground rules.

5.2 Schedule

The mission is slated for launch on April 20th, 2022, after reaching low earth orbit the centaur upper stage will fire and provide the necessary ΔV to arrive at Saturn. After 1,048 days of flight the craft performs a gravity assist at Saturn on February 23rd, 2025. The craft travels for another 3,690 days before arriving at Neptune on April 2nd, 2035. The spacecraft then performs necessary corrections and arrives at the desired Triton orbit in December 2035. The craft then proceeds to map and gather data using its onboard VIP and two deployable probes. All data will be collected and transmitted by January 1st, 2039. This leaves 1 year for the craft to perform additional data capture deemed necessary by mission controllers. After the one year period is completed the craft performs the separation maneuver and commences the HAIL MARY phase. After achieving zero rotational movement relative to the Sun the craft will begin its 29 year free fall into the Sun. The craft transmits higher resolution data the entire duration of its free fall. The craft passes close to the earth in October 2070 and impacts into the Sun in November 2070.

5.3 Organization

In order to ensure the success of the TAGOS mission, every major role was assignment a lead member and a deputy. The breakdown of the team's organization is outlined in the organization chart shown in Figure 5-. Additionally, to ensure all requirements are met and tasks are completed, a work breakdown structure (WBS) was created. The breakdown of the TAGOS mission's functions is outlined in the WBS shown in Figure 5-. The organization chart and WBS are cross-referenced to each other to verify that all requirements and tasks have responsible team members assigned to ensure their completion.

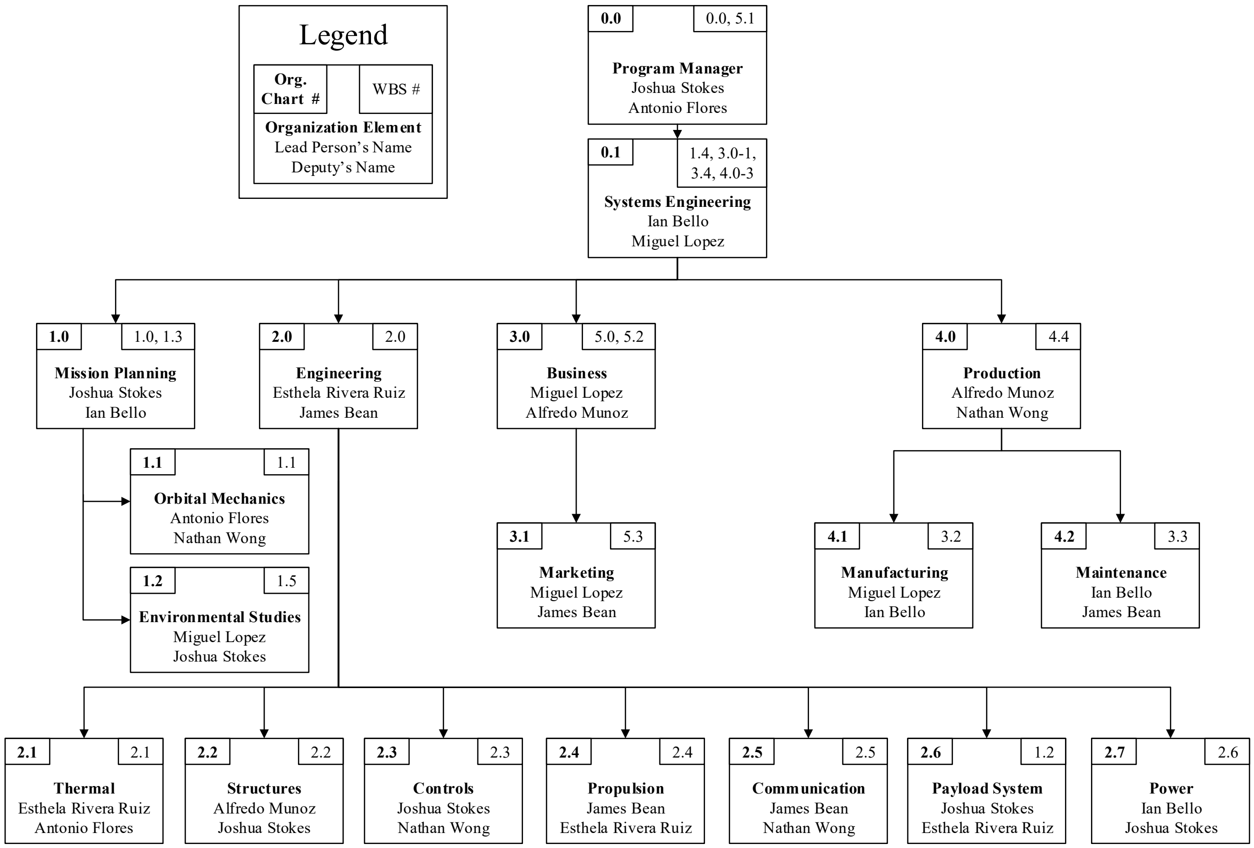


Figure 5-1 Organization Chart Cross-referenced with WBS

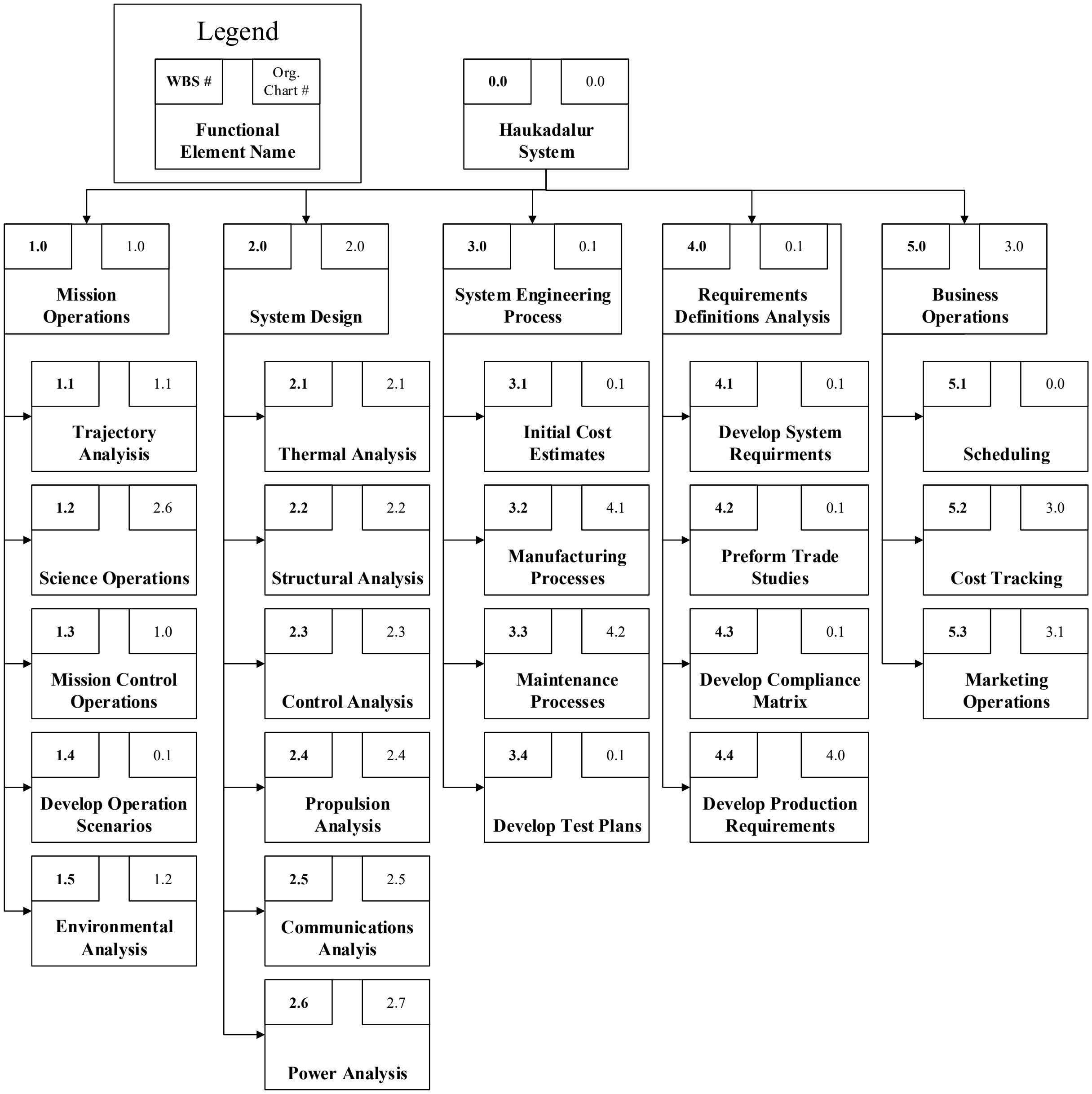


Figure 5-2 Work Breakdown Structure Cross-referenced with Organization Chart

6 Compliance

Table 6- below displays the total Stryker-1 launch mass with a 25% contingency along with the launch margin. This table shows that with the added contingency, the launch mass is currently 722 kg below the maximum value of 7400 kg. Thus, the TAGOS mission is capable of launching on the desired trajectory.

Table 6-1 Stryker 1 Launch Mass and Margin

	Mass (kg)
Orbiter	1,286
Atmosphere Probe	66
Geyser Probe	131
Propellant	3,859
Launch Mass	5,342
Launch Mass with Contingency	**6,678**
Launch Margin	722

The Stryker-1 architecture fulfills all major RFP requirements. Each requirement is referenced to a corresponding section listed in the TAGOS mission compliance matrix below, Table 6-.

Table 6-2 TAGOS Mission Compliance Matrix

Requirement Number	Description	Compliant	Section #
1	Provide geological mapping of geyser zone area with 10 m resolution (Goal: Full surface mapping at 1 m resolution)	✓	2.2.1
2	Determine composition of surface in area not covered by geyser precipitation	✓	2.2.1
3	Determine the composition, particle size, and particle volume density of the entrained solid material released by Triton's geysers in a plum, in an eruption cloud, and on the surface	✓	2.2.3
4	Determine composition of geyser driving exhaust	✓	2.2.3
5	Determine composition, temperature, pressure and density of Triton's atmosphere from 20 km above its thermopause to the surface at 100 m intervals (Goal: from 100 km above surface at 10 m intervals)	✓	2.2.2
6	Shall arrive at Triton by December 2035, complete operations by December 2039 and deliver data before December 2040	✓	3.2
7	Spacecraft must be able to sustain cruise science operations prior to arrival	✓	4.1.6
8	Project must cost less than $5 Billion	✓	5.1

The Stryker-1 architecture fulfills all AIAA data requirements. Each requirement is referenced to a corresponding set of pages listed in the TAGOS mission compliance matrix below, Table 6-3.

Table 6-3 AIAA Data Requirements Compliance Matrix

Requirements	Corresponding Page Numbers
1. Key trade studies and a justification for selection of the overall concept and each of the major subsystems.	1-67
2. Description of proposed flight sequence and mission timeline.	9-20, 70
3. Details of propulsion, vehicle sizing, trajectory, loads, structural, and payload capability analysis. Critical technologies and their current Technology Readiness Level (TRL). Discussion of any required technological breakthroughs or plans for developing technologies to the required maturity.	21-67
4. Discussion of design and concept of operation. Systems that are unique to the proposed design, such as vehicle(s), propulsion subsystem(s), propellant and power subsystems, thermal protection subsystem, and communication subsystems should be addressed in considerable detail.	21-67
5. Subsystems, such as avionics, guidance, navigation, and control which are not the focus of this project, do not require much attention, unless their mass fraction is expected to have significant mission architecture implications.	21-67
6. Discussion of risk mitigation strategies for key technical and programmatic risks.	21-67
7. Drawings of the overall vehicle(s) and key components or subsystems.	2-3, 21-23, 51, 57
8. Estimate of development and operation life cycle cost.	68-70, B-1

References

[1] K. M. Hughes, J. W. Moore and J. M. Longuski, "Prelimary Analysis of Ballistic Trajectories to Neptune Via Gravity Assists From Venus, Earth, Mars, Jupiter, Saturn, and Uranus," vol. 150, 2004.

[2] H. D. Curtis, Orbital Mechanics for Engineering Students, 3 ed., Waltham: Butterworth-Heinemann, 2013, p. 768.

[3] C. Conley, "nasa.gov," NASA, 16 April 2017. [Online]. Available: https://planetaryprotection.nasa.gov/categories. [Accessed 25 April 2017].

[4] J. J. Wijker, Spacecraft Structures, Berlin: Springer-Verlag Berlin Heidelberg, 2008.

[5] D. G. Gilmore and T. A. C. D. G. , Spacecraft Thermal Control Handbook, El Segundo: The Aerospace Corporation, 2002.

[6] C. D. Brown, Elements of Spacecraft Design, Reston: American Institute of Aeronautics and Astronautics, Inc., 2002.

[7] J. R. Wertz, F. D. Eeverett and J. J. Puschell, Space Mission Engineering: The New SMAD, Hawthrone: Microcosm Press, 2015.

[8] Aerojet, "Bipropellant Rocket Engines," 24 May 2005. [Online]. Available: http://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/Bipropellant%20Data%20Sheets.pdf. [Accessed 1 April 2017].

[9] Moog-ISP, "Upper_Stage_Engines," September 2013. [Online]. Available: http://www.moog.com/literature/Space_Defense/Spacecraft/Propulsion/Upper_Stage_Engines_Rev_0913.pdf. [Accessed 1 April 2017].

[10] Orbital ATK, "PSI Tank Data Sheets," September 2015. [Online]. Available: http://www.psi-pci.com/Data_Sheets1_main.htm. [Accessed 1 April 2017].

[11] T. Mai, "Deep Space Network Aperture Enhancement Project (DAEP)," 2015 October 2015. [Online]. Available: https://www.nasa.gov/directorates/heo/scan/services/networks/txt_daep.html. [Accessed 1 April 2017].

[12] J. R. France and M. D. Griffin, Space Vechical Design, Reston: AIAA, 1991.

[13] D. Leone, "U.S. Plutonium Stockpile Good for Two More Nuclear Batteries after Mars 2020," Space News, 11 March 2015. [Online]. Available: http://spacenews.com/u-s-plutonium-stockpile-good-for-two-more-nuclear-batteries-after-mars-2020/. [Accessed 29 April 2017].

[14] C. Dreier, "Closing out the ASRG program," The Planetary Society, 23 January 2014. [Online]. Available: http://www.planetary.org/blogs/casey-dreier/2014/0123-lockheed-closes-out-the-asrg-program-and-ships-hardware-to-nasa-glenn.html. [Accessed 2 May 2017].

[15] J. Withrow, "Request for Information Stirling Development," 2015.

[16] Aerojet, "Monopropellant Rocket Engines," 17 May 2006. [Online]. Available: http://www.rocket.com/files/aerojet/documents/Capabilities/PDFs/Monopropellant%20Data%20Sheets.pdf. [Accessed 1 April 2017].

[17] Mogg-ISP, "Monopropellant_Thrusters," 1 June 2013. [Online]. Available: http://www.moog.com/literature/Space_Defense/Spacecraft/Propulsion/Monopropellant_Thrusters_Rev_0613.pdf. [Accessed 1 April 2017].

Appendix A – Jet Propulsion Laboratory: Request for Proposal

***Distant Geysers:* Project Haukadalur**
Undergraduate Team (Class) Student Design Competition

Request for Proposal:
Map Triton; Sample and Analyze Its Atmosphere, Geyser Plumes, & Surface

Objective: Investigate and return data on the atmosphere, surface, and the active geysers seen on the surface of Neptune's largest satellite, Triton.
Background: In 1989, Voyager 2 made the first and, so far, only visit to Neptune. It returned images unambiguously demonstrating geyser eruptions across a portion of the surface of Triton. This is a major surprise for a small satellite at Neptune's distance from the Sun. The limited imagery (due to the flyby nature of this first visit) displays eruption plumes, extended eruption clouds apparently trapped beneath a thermopause in the atmosphere, and wind streak surface deposits of dark material.
This discovery leads to many questions:
1. What is the mechanism by which geyser exhaust is generated beneath the surface?
2. What is the driving material in the exhaust? What is its physical state (solid particles, liquid, or gas)?
3. What is the dark material entrained in the exhaust?
4. Does the dark material chemically change during the course of its launch, flight, and precipitation to and then on the surface?
5. What is the composition of the surface material surrounding a geyser, near and far?
6. What is the composition of Triton's atmosphere unaffected by geyser plumes, their clouds, and geyser precipitate?
7. What is the overall structure of Triton's atmosphere?

Proposed Design Requirements: Project Haukadalur (pronounced HOW-kah-Doll-yur) shall investigate the geysers found on Triton, its atmosphere, and its surface in the area of the geyser zone.
The project shall provide data for geological (mineral, surface history, and thermal) mapping of Triton's geyser zone with resolution of 10 meters. (Goal: Full surface mapping of Triton at 1 m resolution.)
The project shall determine the composition of the geyser zone surface in a clean area not obviously covered by geyser precipitation.
The project shall determine the composition, particle size, and particle volume density of the entrained solid material released by Triton's geysers in a plume, in an eruption cloud, and on the surface.
The project shall determine the composition of the geyser-driving exhaust (e.g., boiling fresh, mineral water on Earth, boiling brine on Saturn's satellite Enceladus), and its pressure and volume density in the eruption plume.
The project shall 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 (Goal:

100 km above the surface at 10 m intervals). This measurement shall be made 75 km upwind of any geysers in the region or 40 km crosswind and outside the periphery of the geyser region.
The project shall arrive at Triton by December, 2035, complete operations before December, 2039, and deliver its data to the Planetary Data System before December, 2040.
The spacecraft must be able to sustain cruise science operations prior to arrival and science operations thereafter.
The project cost cap is $5B in 2017 dollars, including mission design, spacecraft design, construction, assembly, the launch vehicle, flight operations, data analysis, and data archiving.

Mission Concept Review: During the last half of March, 2017
Preliminary Design Review: During the last half of May, 2017

Proposed Design Submission: The design team must specify a payload and spacecraft configuration and mission design. Possible payload instrumentation might include an imaging system, infrared radiometer or spectrometer, surface composition package, mass spectrometer, dust collector, etc. The spacecraft configuration will determine the design of its subsystems, including attitude control, navigation and course correction, power source, data system, and command system.
The spacecraft may be launched in any suitable window prescribed by the properties of available launch vehicles from the world's spacefaring nations (including NASA's SLS) and the requirements of the spacecraft mass. Reasonable extensions/upgrades to launcher capabilities that are anticipated over the next decade are allowed. Performance requirements should be specified.
Identify a suitable launch vehicle. Determine what range of cruise trajectories and arrival strategies are possible for the payload. Gravitational assist trajectories are encouraged to reduce flight time. (Goal: Identify potential asteroid flyby candidates.)
The design team will specify the extreme operating conditions (e.g., temperature, atmospheric density, radiation exposure) of the vehicle in the environments it will pass through and operate within. Systems for command, control, communication, and data storage and transmission will be specified. The requirements on and characteristics of the power system, thermal protection systems, tracking and positioning systems, any unique systems, etc. required for operations shall be specified. Indicate any needs for planetary protection protocols and their influence on designs.

The design team will prepare and provide a 3-view layout of the proposed vehicle(s). Provide the configuration of all systems and their placement on the structure, including both engineering and science subsystems. Masses for structure, piping, and cabling should be noted, as well as tank capacity and total dry and wet masses as appropriate. Power requirements for all engineering and science subsystems should be compiled.

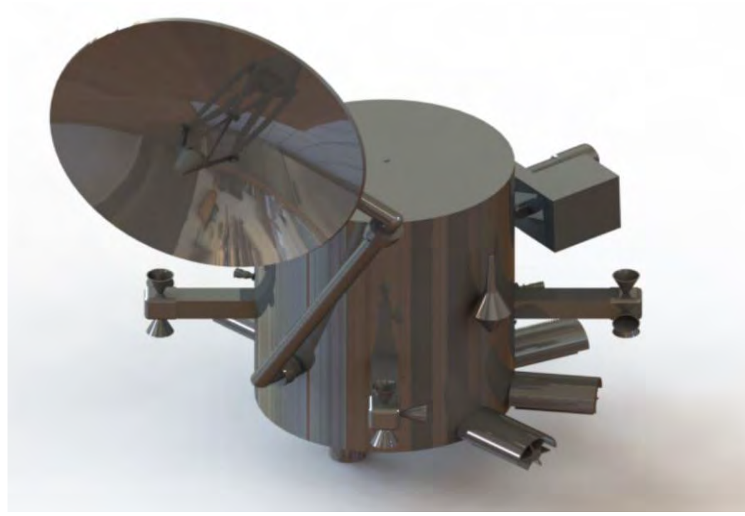
A cost analysis must be provided for the project from initial design, fabrication, and testing, through launch, flight, and data return. It should contain both hardware (including launch vehicle and adapter) and flight and ground system software. Also include estimates for nuclear power system costs (if included in the design), planetary protection protocols, and data archiving in the Planetary Data System.
Estimate costs for interplanetary cruise operations and then science operations beginning with Neptune approach. Flight operations costs should also include add-on estimates of science

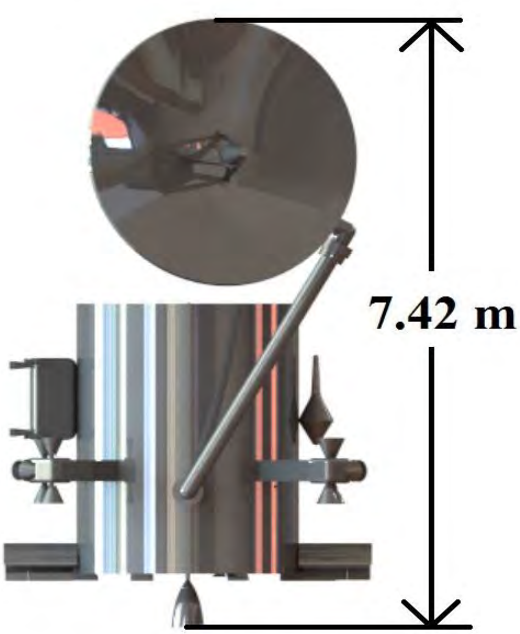
acquisition costs during any gravitational assist flybys or targets of opportunity during the cruise phase including potential asteroid flybys or unexpected targets of opportunity including, for example, the appearance of a comet or a supernova.
Judging Criteria: Project evaluation criteria will focus on the team's understanding of the problem (20%), the technical feasibility of the proposed design (50%), and its capability to meet the specified requirements (30%).

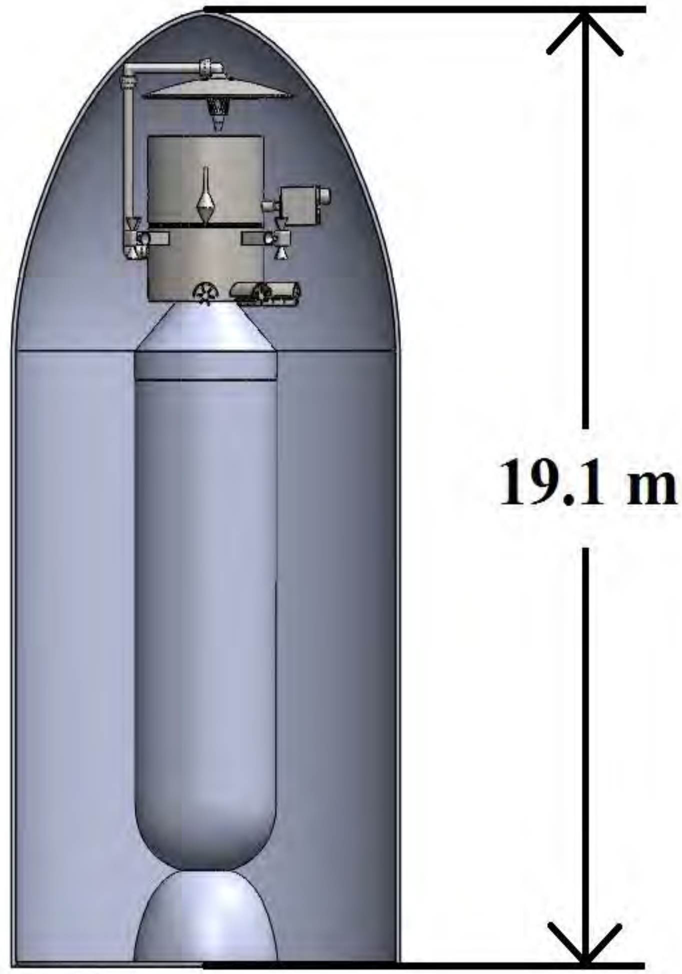
To get started: Wikipedia has a summary of what is known about Triton. However, it should not be considered a primary source of information. Details used for planning should be validated from primary literature. http://photojournal.jpl.nasa.gov/ has imagery of Triton. See, in particular, PIA00059: Triton South Polar Terrain, PIA14449: Triton's Volcanic Plumes, and PIA14448: Triton's Dark Plume. The NSSDC has reliable data on Neptune and Triton at http://nssdc.gsfc.nasa.gov/planetary/. JPL's http://ssd.jpl.nasa.gov/ has useful summary values of physical characteristics and orbital elements. (For any differences between NSSDC and JPL values, choose JPL values. They are updated more frequently and used for interplanetary navigation.)

Historical and Science Notes: Haukadalur is the region of Iceland where the hot-water erupting spring Geysir became the prototype and name for geysers. Eruptive activity has been detected on Earth (varieties of volcanoes and geysers), Io (volcanoes), Enceladus (described as "jets"), Triton (described as geysers), and comets (jets).

Appendix B – Stryker-1 Three View and Payload Fitting



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Appendix C – Mission Cost Breakdown

Large Satellite Cost Model, Unmanned Spacecraft Cost Model (USCM) version 8.0, and NASA Instrument Cost Model (NICM)			
Cost Component	Total Cost (FY17$K)	Std Error ($K)	Std Error Percentage
	2017	2017	
Spacecraft Bus			
Structure and Thermal	$41,439	$9,046	
ACS	$125,125	$53,045	
Electrical Power System	$26,529	$9,988	
Propulsion			
TT&C and Data Handling	$39,761	$7,157	
Integration, Assembly, & Test (IA&T)	$76,866	$29,893	
Flight Software	$23,953	$7,186	0.3
Spacecraft Bus Total Cost	$333,673	$63,181	
Payload			
Scientific Payload			
Fabrication	$41,179	$14,013	0.35
Management	$3,974	$795	0.2
Systems Engineering	$4,782	$1,196	0.25
Product Assurance	$3,120	$624	0.2
Integration and Test	$6,283	$1,571	0.25
Scientific Payload Total Cost	$59,339	$14,187	
Launch Segment	$994,478	$99,448	0.1
Program Level		$73,484	
Launch and Orbital Ops Support (LOOS)	$6,687	$2,006	0.3
Ground Support Equipment (GSE)	$56,655	$20,962	
Total Space Segment Cost to Contractor	$659,877		
10% Contractor Fee	$65,988		
Total Space Segment Cost to Government	$725,864		
Total Cost of Deployment	$1,720,342	$141,160	

Dormant Operations	Cost Category	Annual Cost (2010$K)
PMSE	Labor	$1,374
Space Segment Software Maintenance	Labor	$1,875
Ground Segment		$4,996
Mission Operations	Labor	$3,200
Ground Segment Software Maintenance	Labor	$266
Ground Hardware Maintenance	Labor	$280
Facilities	Facility Lease	$1,250
Total Annual Operations Phase Cost		$8,245
Total Mission Oeprations Cost (2010$K)		$123,677
Total Mission Oeprations Cost (2017$K)		**$138,166**
Active Operations	**Cost Category**	**Annual Cost (2010$K)**
PMSE	Labor	$2,654
Space Segment Software Maintenance	Labor	$1,875
Ground Segment		$11,396
Mission Operations	Labor	$9,600
Ground Segment Software Maintenance	Labor	$266
Ground Hardware Maintenance	Labor	$280
Facilities	Facility Lease	$1,250
Total Annual Operations Phase Cost		$15,925
Total Mission Oeprations Cost (2010$K)		$63,701
Total Mission Oeprations Cost (2017$K)	2017$	**$71,164**
Assumptions	**Value**	**Units**
Software for Space	100,000	SLOC
Software for Ground	25,000	SLOC
Hardware Acquistition Cost	1,400	$K
% of Hardware Acquisition for Ground Hardware Maintenance	20%	

Facility Lease	1,000	sq meters
$/sq meter	1.25	K/sq meter
% of Operations Cost for PMSM (10% - 20%)	20%	
FTE Overhead Adjustment (excluding admin, contractor, travel)	150%	
Number of Engineers for Mission Ops	8/24	
Engineer Annual Salary	80	$K
Engineer FTE	200	$K
Number of Technicians for Mission Ops	4/12	
Technician Salary	60	$K
Technician FTE	150	$K
Number of Years of Operation	15	

Spacecraft Fuel, Oxidizer and Pressurant.			
	Mass (kg)	Cost/kg	Cost
Hydrazine	1652	$17	$28,084
nitrogen tetroxide (NTO)	1652	$25	$41,300
Helium	15.2	$52	$790
		Total	**$70,174**

Appendix D – Triton Data Sheet

Parameter	Value
Mass (kg)	2.14×10^{22}
Mean Radius (km)	1,353.5
Volume (km^3)	10,384,000,000
Surface Gravity (m/s^2)	0.779
Temperature (K)	38
Surface Pressure (Pa)	1.4-1.9
Composition	Nitrogen with traces of methane
Semi-major Axis (km)	354,759
Orbital Period (days)	5.876 (retrograde)
Orbital Speed (km/s)	4.39