Design Final Report

in response to

Caltech Space Challenge 2022

Emrys Space Systems

5/18/2022

Aerospace Engineering Spacecraft Design 2021-2022

Org Chart

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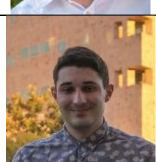	GNC and ACS Jett Groussman	
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Acronyms

Acronym	Definition	Acronym	Definition
ACS	Attitude Control System	**MI&T**	Manufacturing, Integration, and Test
C&DH	Command And Data Handling	**NEXT**	Nasa Evolutionary Xenon Thruster
CIRS	Composite Infrared Spectrometer	**PCEC**	Project Cost Estimating Capability
COMMS	Communications	**PIXL**	Planetary Instrument for X-Ray Lithochemistry
DSN	Deep Space Network	**PPP**	Planetary Protection Protocol
DSS	Digitized Sky Survey	**RCS**	Reaction Control System
DTE	Direct To Earth	**RFP**	Request For Proposal
DV	Change In Velocity	**RTG**	Radioisotope Thermoelectric Generator
EOL	End Of Life	**SC OR S/C**	Spacecraft
G	Acceleration On Earth Surface Units: m/s	**SSPA**	Solid State Power Amplifier
IMU	Inertial Measurement Unit	**STK**	Systems Tool Kit
ISP	Specific Impulse	**TEDA**	Titan Environment Dynamics Analyzer
ISPC	In-Situ Propellant Collection	**TEI**	Trans Earth Injection
KRUSTY	Kilopower Reactor Using Stirling Technology	**TOF**	Time Of Flight
LAV	Landing Ascent Vehicle	**TPS**	Thermal Protection System
LCAM	Lander Vision System Camera	**TWTA**	Traveling-Wave-Tube Amplifier
LGA	Low G Accelerometer	**UVIS**	Ultraviolet Imaging Spectrograph
LOX	Liquid Oxygen		

1 Executive Summary

In response to a request for proposal issued alongside the Caltech Space Challenge 2022, Emrys Space Systems has developed a solution to fulfill the core mission objective: collecting and returning a sample from the surface of Titan. Initially developing two architectures and downselecting to one for the focus of the report, the team believes that they have made all the necessary considerations for a robust and efficient design.

Architecture one consists of three unique vehicles: an Orbiter, a Landing and Ascent Vehicle (LAV), and an autonomous rover. The three vehicles will start as a single spacecraft launched via the Space Launch System Block 1B Cargo in March of 2026 where it will begin an 816-day long transit to Titan. To accomplish this, architecture one leverages an electric propulsion system featuring a unique combination of two NEXT Ion Thrust Engines and a KRUSTY 10 kW nuclear reactor. Upon arrival to Titan, the Orbiter will begin a short duration science observation of the environment of Titan and identify the landing spot for the LAV using CIRS and UVIS. Once a suitable landing spot is well understood, the LAV will separate and begin a landing sequence to Bolsena Lacus, our selected target lake. Upon arrival to the shores of the lake, the rover will be deployed and conduct an autonomous operations sequence to collect both the sample and all the necessary in-situ science data with the robust instrument package. With return to the LAV, the samples will be transferred from the rover to the sample return capsule. Departing in December of 2038, the LAV will launch from Titan, rendezvous with the Orbiter to refuel and then continue its journey back to Earth. Arriving to Earth in October of 2045, the sample return capsule will be deployed and the remainder of the LAV will be placed in a graveyard orbit around Earth.

Architecture two leverages the resources available on Titan, namely the abundance of methane and oxygen. By implementing an in-situ propellent collection (ISPC) system, propellant mass can be omitted from the launch mass leading to a lower launch mass. The ISPC system would allow for the LAV to collect all the necessary propellant for a direct transit back to Earth from Titan, cutting out the need for an orbital rendezvous to refuel. The overall system would consist of a cruise stage featuring the KRUSTY nuclear reactor and the two NEXT engines, the LAV, and the rover.

By utilizing a systems engineering approach, we implemented a trade study which used the strongly weighed Technology Readiness Level and cost and found that Architecture 1 was our winning architecture. After downselection, structural analysis was completed followed by a mechanical redesign for weight savings. In addition, refined analysis was completed for the thermal needs of the spacecraft and the attitude control system.

Emrys Space Systems has developed a robust and thorough design in response to the request for proposal set forth by the Caltech Space Challenge 2022. Our design consists thorough concept of operations, a novel hardware design, detailed science objectives and traceability, and an in depth look at risk and cost.

2 Mission Background

In this section, the team establishes the system engineering baseline with needs, goals, objectives, and system level requirements. In addition, the science baseline is established with the science traceability matrix.

2.1 Needs Analysis

The team has chosen to respond to the Titan Sample Return mission from Caltech for the 2021-2022 design challenge. This mission calls for a spacecraft to travel to Saturn's moon, Titan, and retrieve a sample from the liquid lakes from Titan, as well as obtain in-situ measurements of the Titan surface, concluding with the return of all data and the sample to Earth. Titan is a place of high value to the scientific community, as it is the only known celestial body in our solar system to have liquid oceans and lakes on its surface. Unlike Earth, however, Titan's oceans and lakes are comprised of hydrocarbons; mainly methane and ethane in their liquid state, due to Titan's cold atmosphere. In addition, the Huygens mission discovered deposits of ice located on the Titan surface which the team considered as a potential source for in-situ propellent collection systems, should a spacecraft design require it.

Historically, scientists have focused primarily on searching for life on Earth-like planets with a high possibility of carbon-based life. Recently, a new hypothesis has emerged: instead of life only being possible with rigid Earth-like conditions, life could spawn in any number of circumstances dissimilar but analogous to Earth. According to James Stevenson et al. of Cornell University, a theoretical molecular structure known as Acrylonitrile Azotosome, also known as Vinyl Cyanide, shown in Figure 2.1-1, could exist, since it can be formed from liquid hydrocarbons located in Titan's lakes. Vinyl Cyanide would function as a cell membrane for Titanian life forms, allowing complex cellular construction of extra-terrestrial life forms. The internal structure of these Titanian cells would be a place of high interest to biologists as they are expected to differ vastly from the cells that are present on Earth.

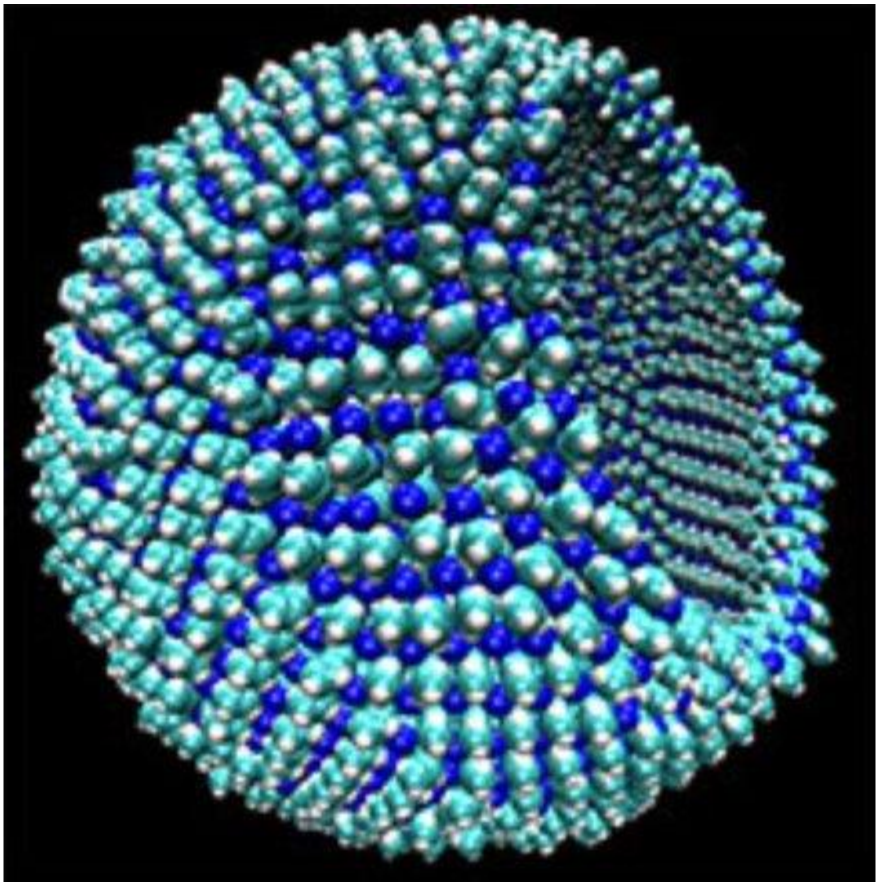

Figure 2.1-1 Acrylonitrile Azotosome Theoretical Cell Membrane

Titan is also a place of interest given that it is the only celestial body in our solar system with a hydrological cycle like Earth. This would give valuable insight into other worldly weather cycles and inform future spacecraft missions of what conditions to expect when traveling to the Titan surface.

2.2 Objectives and Goals

The biggest objective of the mission is to retrieve and return a sample of liquid from the surface of Titan. To ensure the liquid samples from Titan arrive on Earth in conditions as close as possible to their natural state, atmospheric chemical composition, pressure, and temperature will be measured by the LAV once on the ground. This will provide the best chance of sample preservation since any changes to the sample pressure and temperature could result in damage or contamination and limit the amount of information that can be gained from the sample once it has been successfully returned to Earth. By gathering this data, we will concurrently meet the other mission objective of collecting in-situ environmental data.

2.3 System Level Requirements

The system level requirements were gathered from the RFP, which outlined the constraints on the design, are shown in

Table 2.3-1. Derived requirements can be traced back to these system level requirements.

Table 2.3-1 System Level Requirements

Req #	Requirement Statement
SYS1.0	Spacecraft shall return a sample of at least 500 mL of surface liquid from Titan
SYS2.0	Launch vehicle shall launch by 2030
SYS3.0	Design shall address all required propulsive maneuvers and corrections to the orbit
SYS4.0	In-situ measurements from Titan shall be returned to Earth by 2045
SYS5.0	The sample and all science data shall be returned to Earth no later than December 2045
SYS6.0	Total mission cost shall not exceed $5B (FY21)

2.4 Science Traceability Matrix

From the requirements and objectives set by the RFP, a Science Traceability Matrix (STM) can be developed to further drive the design of the spacecraft and the mission. The first requirement is to have the spacecraft retrieve 500 mL of liquid sample from Bolsena Lacus. This requirement ensures that there will be a viable number of samples to be analyzed on Earth with multiple experiments and to prevent contamination of any one sample either during the return phase of the mission or during the experiments themselves. The samples will be retrieved using a mechanical retrieval method located on the spacecraft's ground operation system. Once the sample has been obtained, the SHERLOC camera will be used to detect Acrylonitrile Azotosome within the sample to verify if the theoretical structure exists. In addition, sample composition will be monitored to determine if any metabolic processes are taking place within the sample, and all results will be transmitted back to Earth via the DSN.

To ensure preservation of the 500ml of liquid sample during the return phase of the mission, the temperature and atmospheric pressure of the liquid sample at the retrieval site will be observed and simulated throughout the return phase of the mission. An on-board thermometer and pressure gage in conjunction with subsystems will be used for maintaining the measured parameters for sample preservation. As mentioned above, one sample will be heated to gain insight into any biological decomposition or other chemical reactions that may occur during the return phase of the mission. This will be done by radiating heat from the ground operation spacecraft over the sample container with the liquid sample inside. Once this is complete, the SHERLOC camera will be used once again to detect any significant changes, which will be transmitted back to Earth via the DSN.

Another major goal for the mission is to measure chemical concentrations of the sample over the duration of the mission, as this would allow scientists to determine if any metabolic processes are taking place in the sample during the return phase of the mission. Metabolic processes are processes that life forms on Earth do in order to maintain homeostasis. If there were significant changes in the sample, this would provide some evidence that some biological life form could exist within the sample. It would also indicate what nutrients are needed by the life form in order to maintain its biological equilibrium in order to survive. This information is imperative to determining if the biological life forms within the sample were to die or begin decomposing. Having this information during the return phase of the mission would provide biologists with the rate of decomposition, as well as information on what kind of nutrients would be needed in future missions wanted to retrieve the life forms alive.

During the descent phase of the mission, the cameras on board the LAV will collect a topographical map of Titan's surface. This will ensure that no significant geological hazards are present during ground operations as well as giving further insight into the known environment of Titan. Once the LAV has landed, the Titan environment will be analyzed using the TEDA (Titan Environment Dynamics Analyzer) which will also give further information of the Titan environment and ensuring that environmental conditions are suitable for mission success during the ground operations of the mission.

While ground operations are taking place, the Orbiter will obtain Titan's atmospheric temperature distribution and chemical composition using the CIRS and UVIS. This will allow for detection of any significant changes to the Titan atmosphere between Cassini/Huygens in 2004 and the current mission. Data will be transmitted to Earth via the DSN during the ground operations of the mission.

Lastly, to ensure mission success and the preservation of the Titan environment and Earth's environment, COSPAR's Planetary Protection guidelines for Category V missions will be referenced. The probability of impacting any celestial body will be less than 0.01% over the duration of a minimum of 50 years. This will be accomplished with extensive mission and trajectory design as well as implementing safekeeping systems to ensure a collision does not occur.

In addition, the possibility of contamination of the Titanian or Earth environment will be less than 0.1%. This will be accomplished using sterilization techniques during manufacturing and using a sealed sample containment system for the return phase of the mission.

The full development of the STM can be seen in Appendix B at the end of this report.

3 Mission Design

3.1 Design Process

Throughout November 2021, the major customer requirements and objectives were ascertained from the Caltech Space Challenge Request for Proposal for a Titan Sample Return, as discussed in the preceding sections. Over the next few months, two architectures were designed and analyzed. The application of the systems engineering process during design can be found in Section 9. After sharing these designs in design reviews and incorporating feedback from Northrop Grumman, JPL, and Lockheed Martin, the team downselected to the architecture that best served the mission needs and continued to refine the design.

3.2 Architecture 1 Summary

Architecture 1, shown in Figure 3.2-1, features a set of vehicles namely the Orbiter, the Landing and Ascent Vehicle (LAV), and the Rover. Each of these vehicles are designed to function and separate as mission segments are completed. The Orbiter consists of the CIRS and UVIS instruments, a KRUSTY nuclear reactor, and two NEXT Ion Propulsion engines in addition to standard spacecraft bus hardware. The LAV consists of the sample containment system alongside standard bus hardware components to complete the primary mission objective. The Rover has a set of instruments which can be found in **Table 7.2-1 Rover Science Instruments** and the sample collection system.

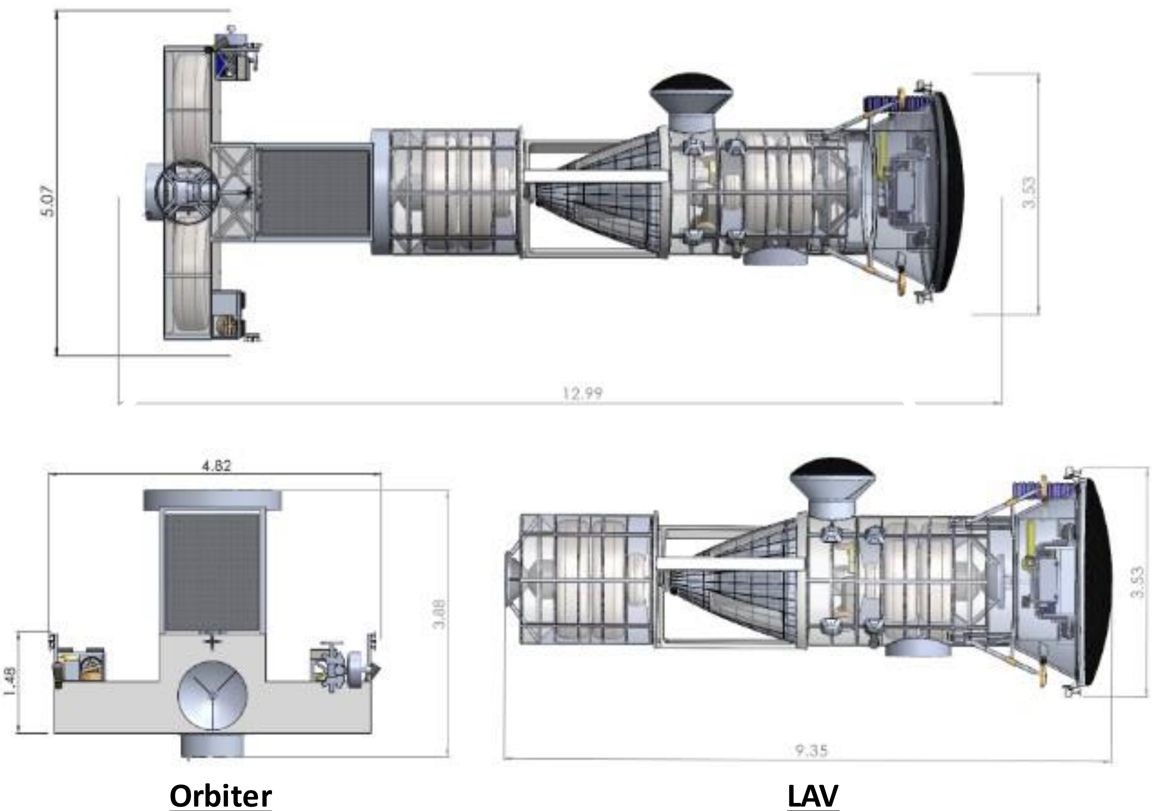

Figure 3.2-1 Architecture 1 Summary

3.3 Architecture 2 Summary

Architecture 2, shown in Figure 3.3-1, utilizes a new technology to significantly lower launch mass. In-situ propellant collection leverages the resources on Titan, namely the abundance of methane and oxygen present on the surface. This architecture uses a direct descent vehicle like Architecture 1's LAV and a cruise stage featuring the NEXT engines and KRUSTY nuclear reactor. Upon arrival to Titan, the LAV will commence a propellant processing phase concurrent to the nominal mission operations of sample and environmental data collection. The LAV takes a direct approach from Titan to Earth and delivers the sample prior to the requirement of December 2045.

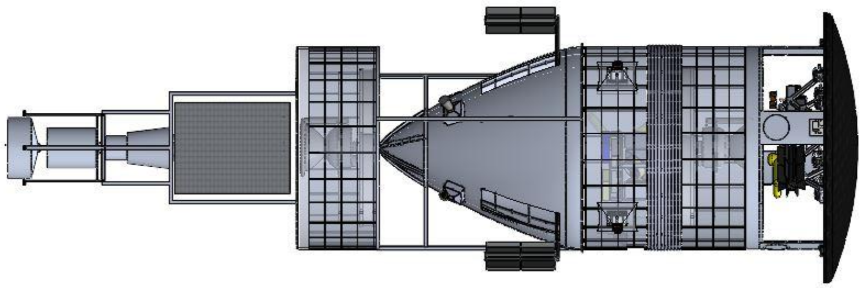

Figure 3.3-1 Architecture 2 Summary

3.4 Architecture Downselection

After creating two viable architectures, the team assessed the Technology Readiness Level (TRL), cost, the mission time on Titan, and launch mass of each architecture. Due to the unique goals of a Titan sample return, the priorities in this assessment centered on reducing risks. Thus, TRL and time spent in a relatively unknown environment, the surface of Titan, were heavily weighted. The in-situ propellant collection system in Architecture 2 significantly reduced its viability due to the low TRL. The results of the downselection trade study in **Table 3.4-1** shows Architecture 1 as best balancing the needs and feasibility of the mission.

Table 3.4-1 Architecture Downselection Trade Study

FOM		Architecture 1		Architecture 2	
Criteria	Weight Factor	Utility	Weighted	Utility	Weighted
TRL > 5	4	**9**	36	1	4
Mission Cost < $5B	2	**6**	12	9	18
Time on Titan > 15 days	3	**3**	**9**	9	27
Launch Mass < 20,000kg	1	**3**	**3**	6	6
Weighted Total		**60**		**55**	

4 Mission Architecture

This section contains the team's mission design, including the concept of operations, landing site selection, trajectory design, and launch vehicle selection.

4.1 Concept of Operations

The concept of operations defines the entire mission timeline of events; specifically, the operations that will take place during the entire mission from launch to capsule recovery. The mission spans 19 years, beginning with a spacecraft launch in 2026 and concluding with Earth re-entry in 2045.

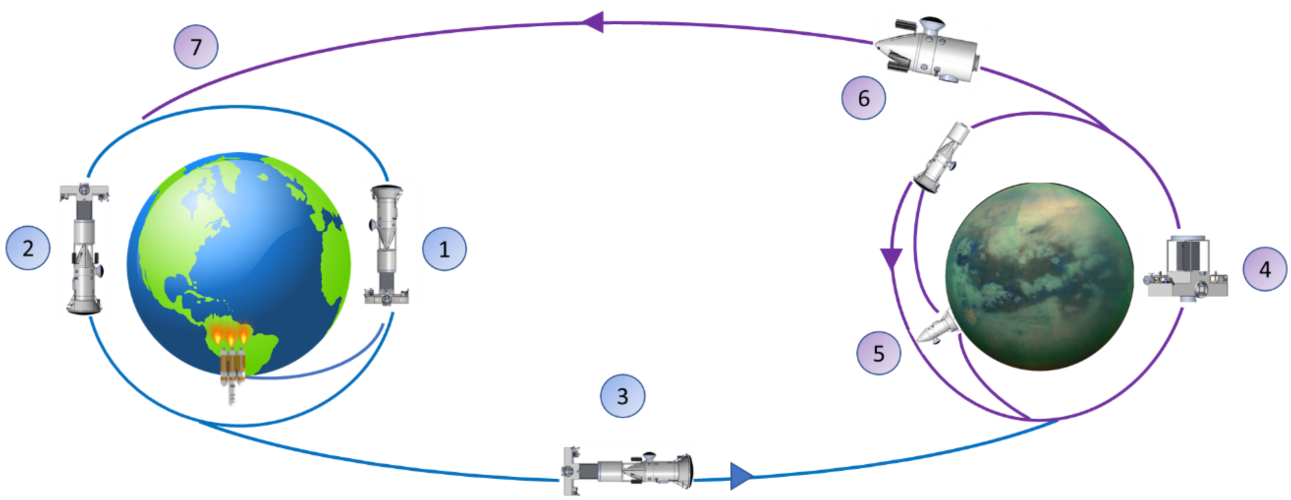

Figure 4.1-1 ConOps: Overview

The mission begins with a spacecraft launch from the SLS Block 1B Cargo into a trajectory for an Earth flyby at a C3 of 49 km^2/s^2. During the arrival to Titan, the spacecraft will perform three continuous thrust maneuvers using NASA NEXT Ion Engines. The first of these occurs at the apoapsis of the Earth flyby trajectory orbit. This will raise the orbit to perform an earth flyby at an altitude of 200 kilometers, as seen in Figure . The spacecraft will then be on a direct trajectory to Titan where it will perform the rest of the two continuous thrust maneuvers. This trajectory will take 4699 days before reaching Titan where the spacecraft will perform a spiral capture. The spiral capture will last 681 days until the spacecraft is successfully at a parking orbit of 1000 kilometers in altitude above the surface of Titan.

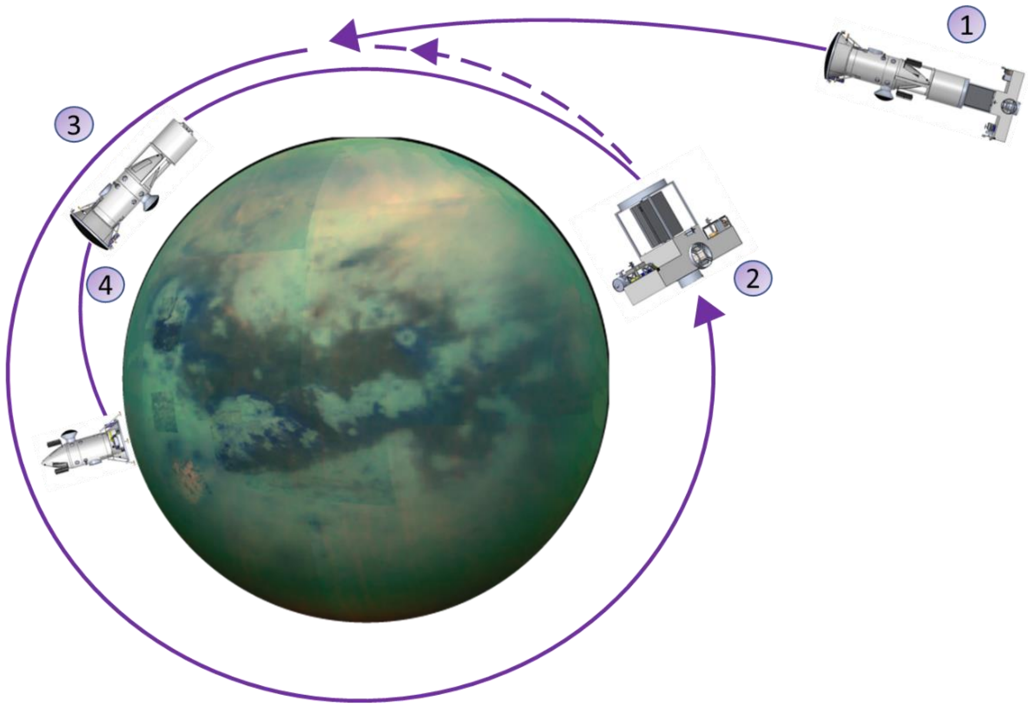

Figure 4.1-2 ConOps: Titan Capture

Once the spacecraft is successfully in its parking orbit, the Landing and Ascent Vehicle (LAV) will separate from the orbiter and descend into the dense Titan atmosphere. After entry, the LAV will deploy parachutes landing on the shore of the methane lake, Bolsena Lacus. On the ground, the LAV will begin by deploying the rover to the Titan surface. The rover will perform health and functional checkouts of its vital equipment and science equipment, then begin its journey across Titan's surface to Bolsena Lacus. Once it arrives at the lake, the rover will collect 5 samples for a total of 0.5 L of liquid samples. The rover will then transport the samples back to the LAV. At the LAV, the rover will transfer the samples to the return capsule and the LAV will prepare for launch. These entire operations will take 24 days beginning on Nov 26th, 2038, until the window for departure on Dec 20th, 2038.

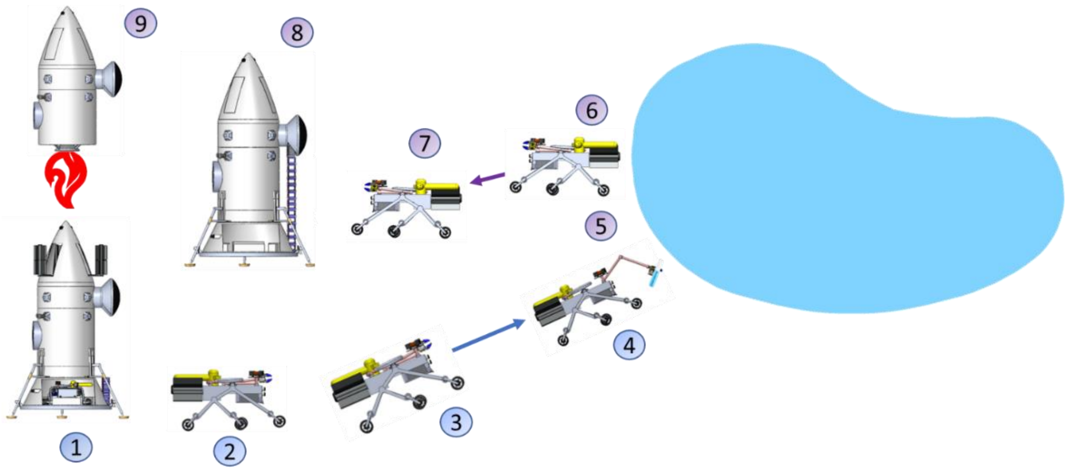

Figure 4.1-3 ConOps: Surface Operations

Upon completion of surface operations and proper sample storage, the LAV will ascend from Titan's surface using its ascent stage propulsion system. It will ascend and circularize to an altitude of 1000 km. Afterwards, the LAV will rendezvous with the orbiter where it will dock and refuel 1300 kg of propellant. The LAV will then separate from the orbiter and perform its trans-Earth injection burn (TEI).

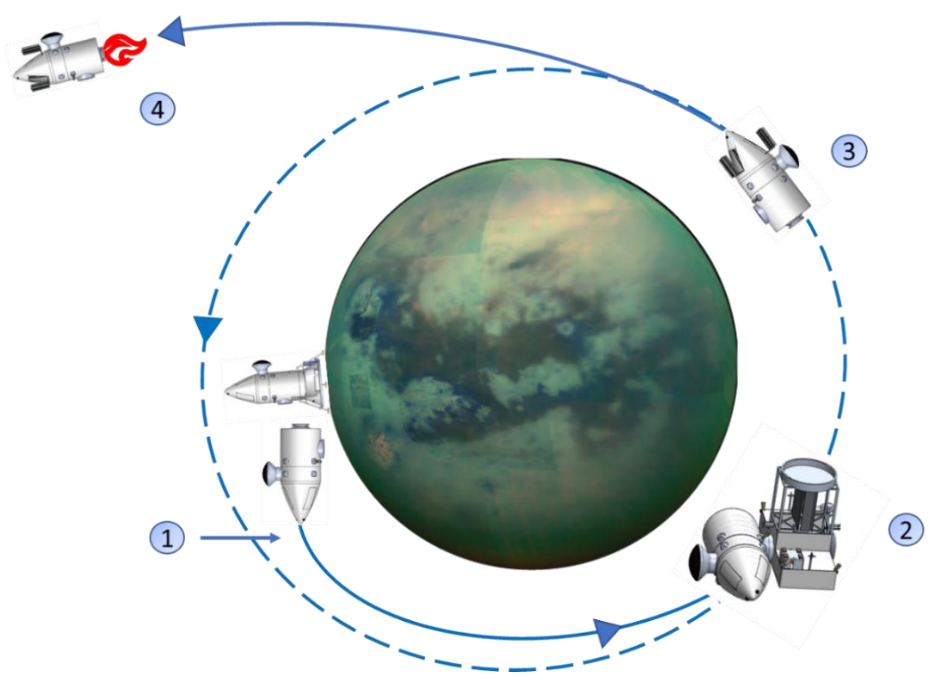

Figure 4.1-4 ConOps: Titan Departure

The LAV will be in a TEI coast for 2479 days before reaching earth on Oct 3rd, 2045. On approach, the LAV will separate from the return capsule sending it on an earth re-entry trajectory where it will approach at an angle of 5 degrees and experience a maximum deceleration of 30 Gs at 51 kilometers. After maximum deceleration of the return capsule, parachutes will deploy, and the vehicle will land softly in the ocean. The LAV will then make an Earth hyperbolic flyby where it will be sent on a path to a heliocentric orbit.

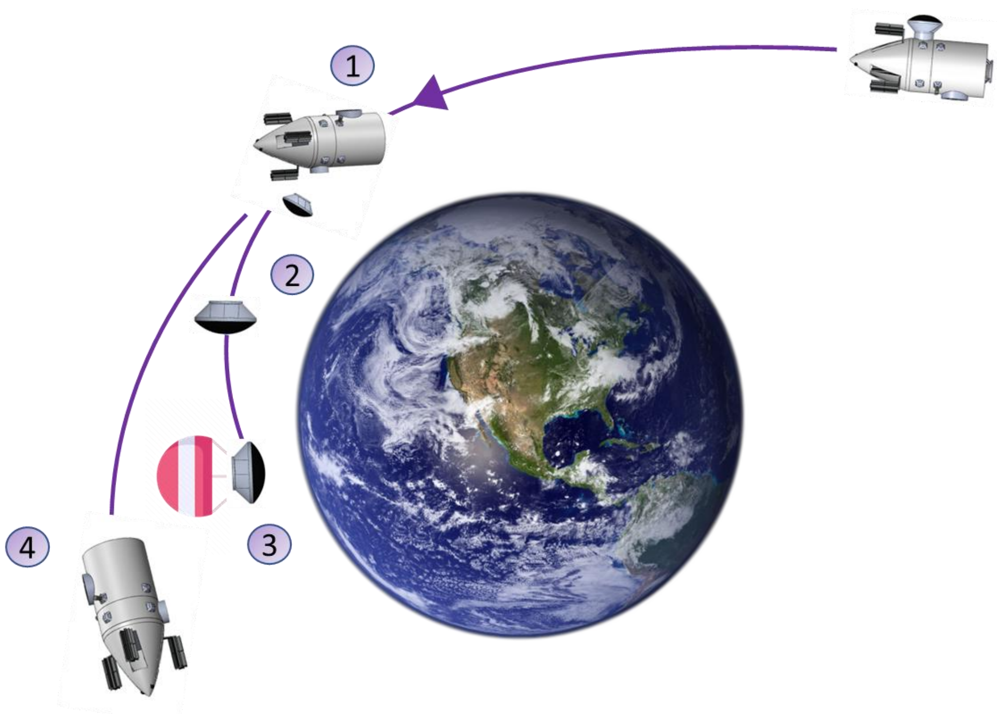

Figure 4.1-5 ConOps: Earth Arrival

4.2 Architecture Sizing and Layout

The selected architecture consists of the Orbiter, LAV, sample collection rover and sample return capsule. This section will breakdown the stowed and mission configuration, as well as the sizing for each vehicle in the architecture.

4.2.1 Stowed Configuration

The stowed configuration of the architecture includes all vehicles attached, with the LAV docked with the orbiter via the docking bay and the sample collection rover stowed within the LAV. While stowed, the height of the architecture is 13 m, the maximum width is 5 m, and the minimum width is 3.5 m. Figure 4.2.1-1 shows the dimensioned stowed configuration of the mission architecture.

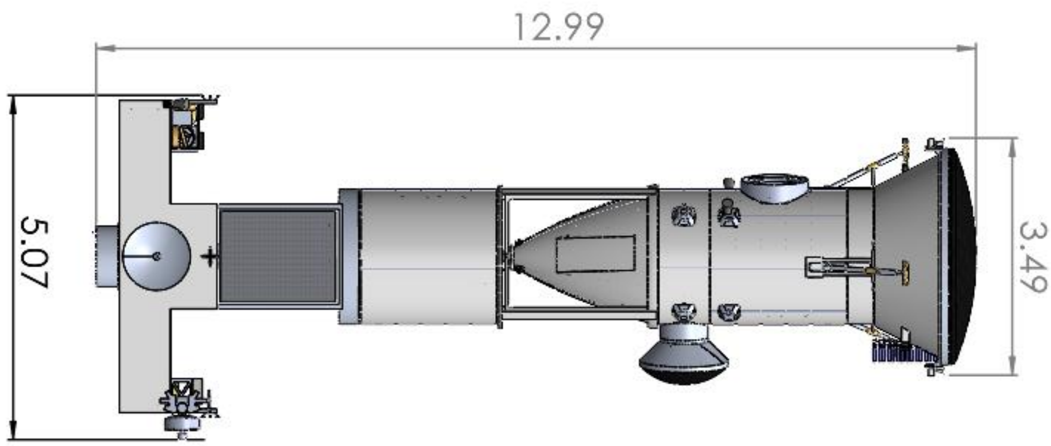

Figure 4.2.1-1 Stowed Configuration of Architecture

4.2.2 Mission Configurations

The mission configuration is broken down into two segments. The first segment is in Titan orbit, where the Orbiter will sit in a parking orbit around Titan while the LAV makes its descent towards the surface. The Orbiter has a maximum height of 3.9 m and width of 4.8 m, as seen in Figure 4.2.2-1.

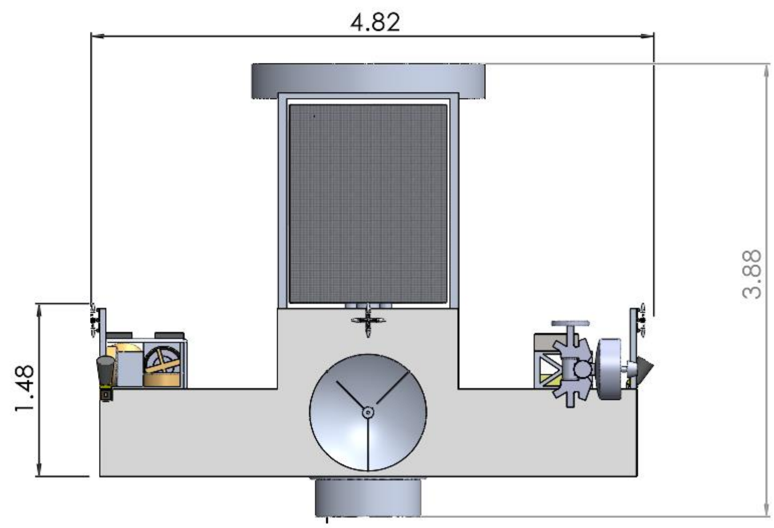

Figure 4.2.2-1 Orbiter Mission Configuration

For the second segment of the mission configuration, the LAV lands on the surface of Titan and starts surface operations. As the LAV approaches the surface, it will deploy its landing legs and land onto the surface of Titan. Once touchdown on the surface occurs, the sample collection rover door will open to release the rover into its mission configuration. The LAV has a maximum height of 6.5 m and a maximum radius of 3.9 m after landing, as seen in Figure 4.2.2-2.

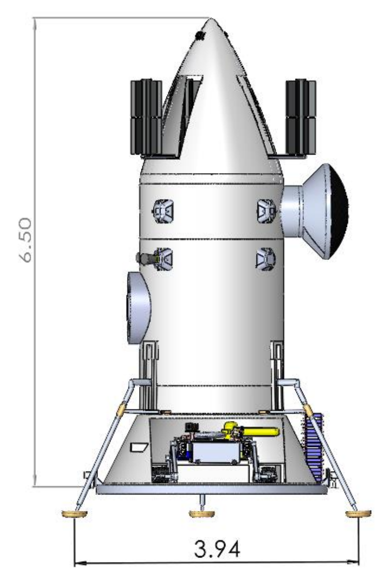

Figure 4.2.2-2 LAV Landed Mission Configuration

The sample collection rover will deploy and drive to the target lake to collect the liquid samples autonomously. Once the samples have been collected, the rover will make its way back to the LAV to store the samples in the sample collection capsule aboard the LAV. The sample collection rover has maximum height of about 1.1 m, with a length of about 1.6 m, and a width of about 1.7 m, as seen in Figure 4.2.2-3.

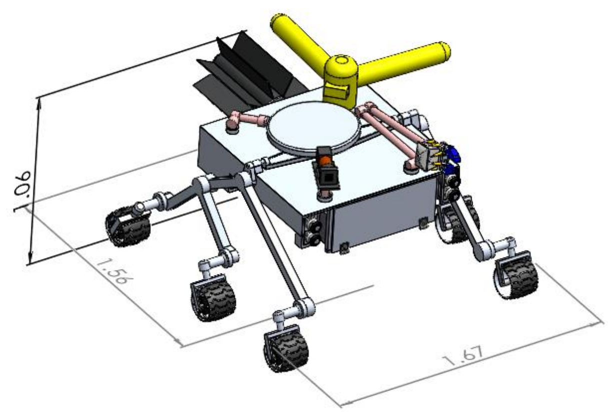

Figure 4.2.2-3 Sample Collection Rover Mission Configuration

Once the samples have been stored in the sample collection capsule, the LAV will then takeoff to rendezvous with the orbiter to refuel and then transit back to Earth. The bottom segment of the LAV that held the rover and landing legs will be left at Titan, while the ascent portion of the LAV will take off to return the sample. The ascent portion of the LAV has a maximum height of about 5.3 m and a maximum radius of about 3.0 m, as seen in Figure 4.2.2-4.

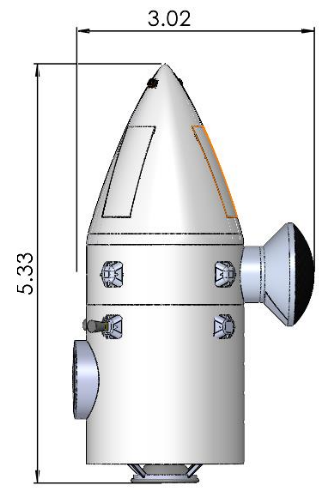

Figure 4.2.2-4 LAV Ascent Portion Mission Configuration

4.2.3 Orbiter Layout

The Orbiter's main functions are transporting the LAV to Titan, refueling the LAV ascent stage, and relaying communications between the LAV and Earth. The Orbiter layout is seen in Figure 4.2.3-1.

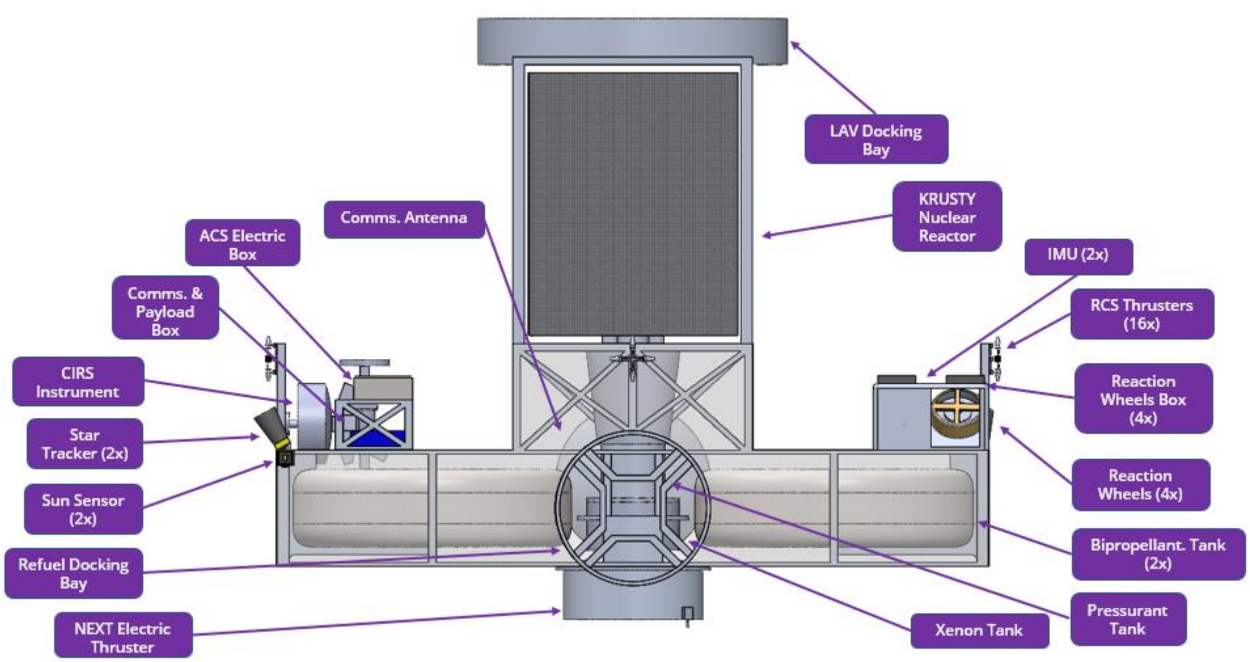

Figure 4.2.3-1 Orbiter Bus Layout

4.2.4 Landing and Ascent Vehicle (LAV) Layout

The LAV's main functions are to land on Titan, deploy the sample collection rover to retrieve liquid samples, and return samples back to Earth. The LAV layout is seen in Figure 4.2.4-1.

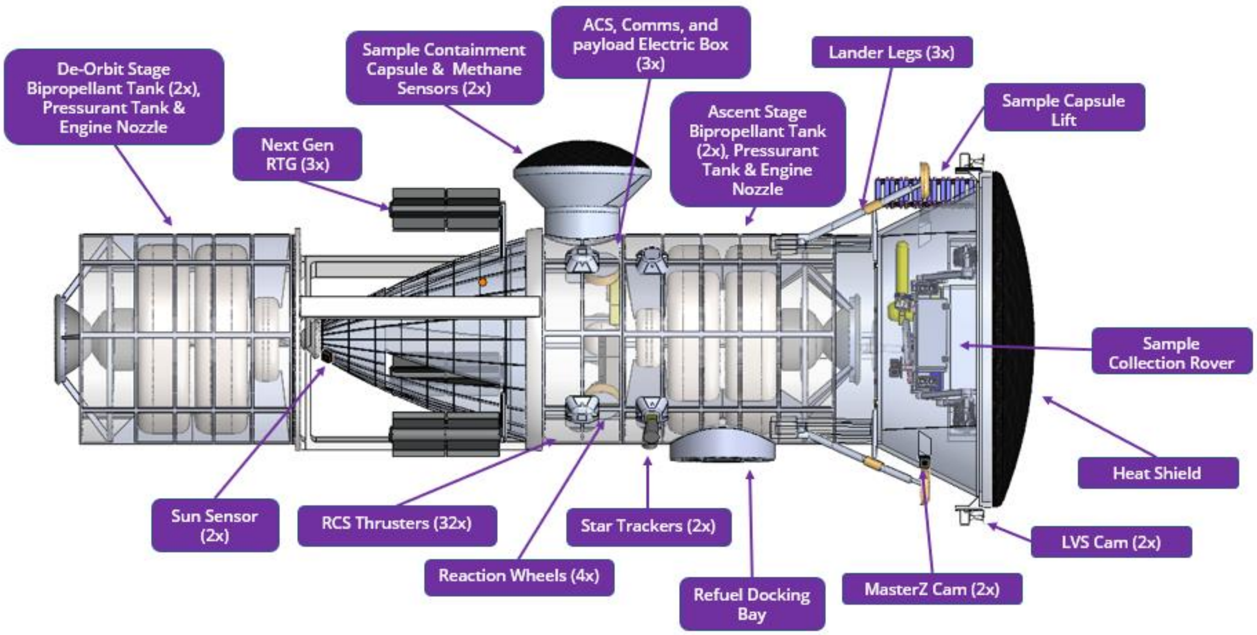

Figure 4.2.4-1 LAV Bus Layout

The sample collection rover is stored within the LAV and gets deployed once landing is completed. The rover's main function is to collect, analyze, and store the liquid samples in the sample return capsule. The rover's layout is seen in Figure 4.2.4-2.

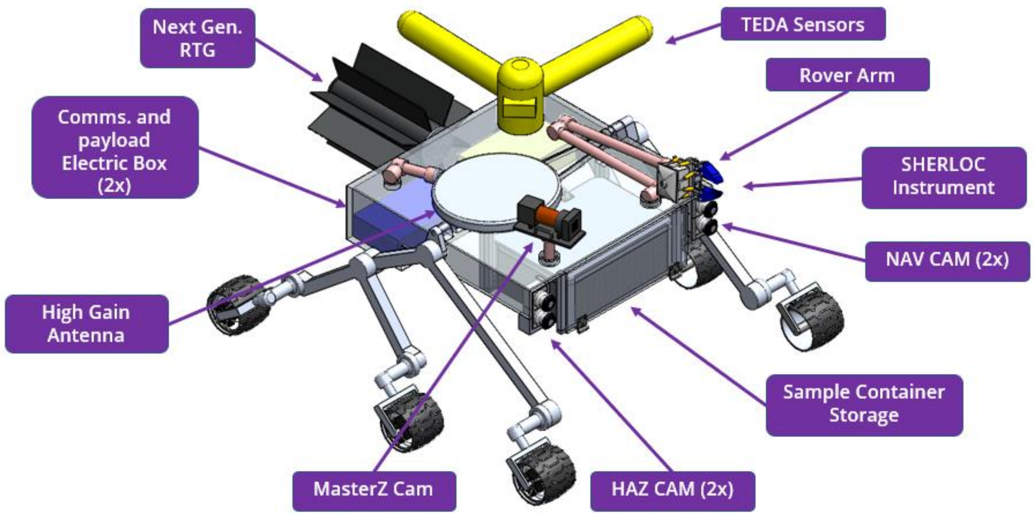

Figure 4.2.4-2 Sample Collection Rover Bus Layout

4.3 Landing Site Selection

The highest priority objective of the mission is to retrieve the liquid sample from Titan's surface. To accomplish this, a target lake must be selected for the mission. Many different lake bodies were considered, as well as Titan's vast methane-based oceans. However, due to the Earth return deadline in 2045 and total mission cost limit of $5 billion dollars, retrieving samples from a single liquid methane source is the optimal route. With these constraints in mind, a lake named Bolsena Lacus became the primary target for sample retrieval, since the surrounding terrain is relatively level and the lake itself is relatively large at 101 km across. Large lake size will allow the spacecraft to

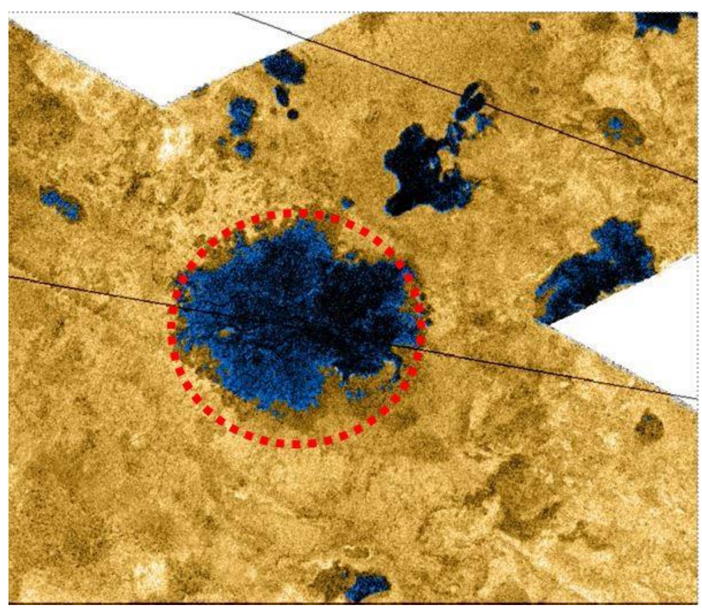

Figure 4.3-1 Bolsena Lacus Target Lake

ascertain the target lake's position from orbit, making the descent phase of the mission much more feasible.

4.4 Trajectory Design

The design for the mission trajectory was performed using a combination of NASA's Mission Analysis Low-Thrust Optimizer (MALTO) and the NASA Ames Trajectory Browser. The approach for the design was to use the NASA Ames Trajectory Browser as an initial broad search of trajectories to Saturn, the results of which appear in Figure 4.4-1 below. The reason Saturn was chosen as opposed to Titan was due to the limited search bodies available using the tool. This, however, would be able to provide rough timelines that could be used to optimize the solution to the current mission constraints using MALTO.

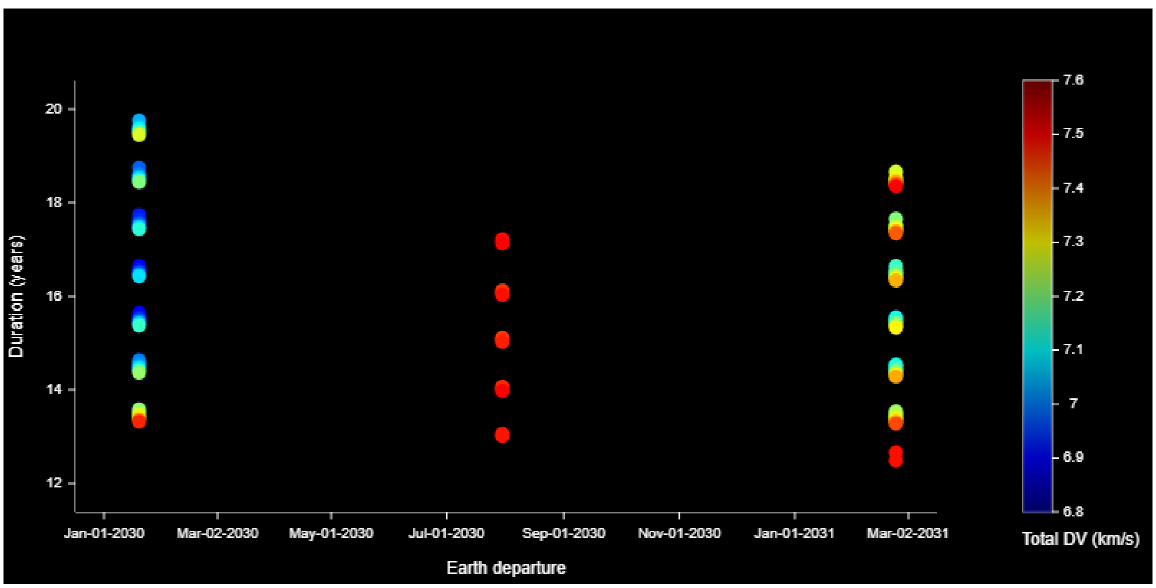

Figure 4.4-1 NASA Ames Trajectory Browser Results

While the NASA Ames Trajectory only calculates physically possible trajectories, it fails to consider many spacecraft constraints that make these trajectories impossible. Spacecraft specifications such as mass, power, and available propulsion technology are all considered in NASA's MALTO software. This software was used to design the final iteration of the mission trajectory. The trajectory was broken up into two segments: the arrival, and the return. This makes it easier for MALTO to converge on an optimized solution.

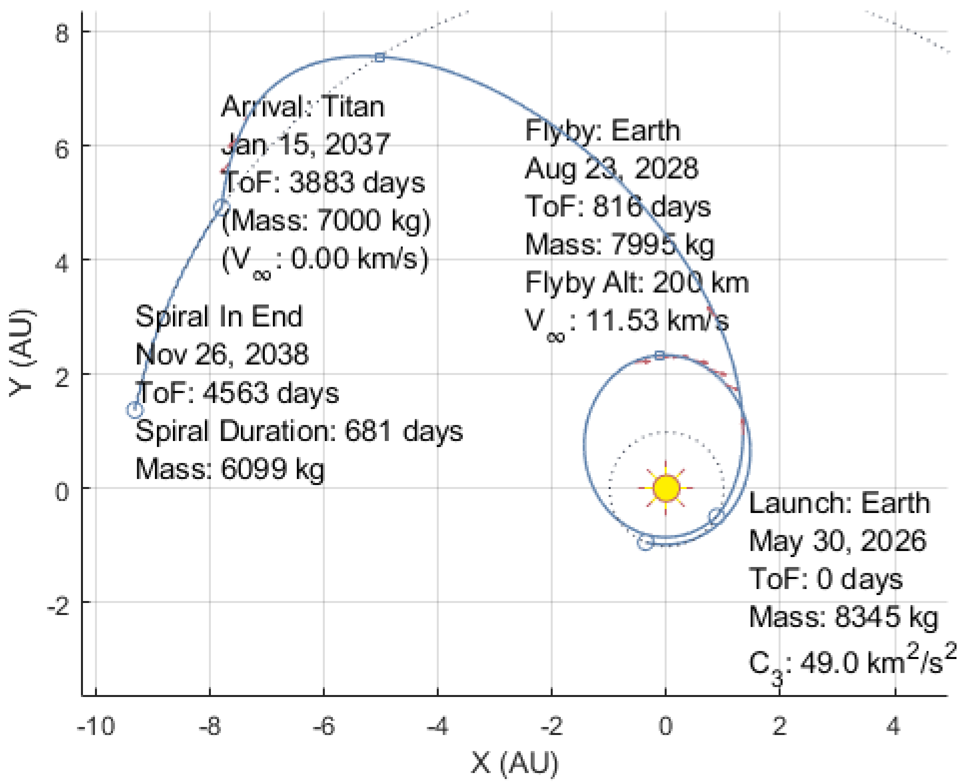

Figure 4.4-2 MATLO Arrival Trajectory

The first segment, arrival, starts with a launch vehicle escape at a C3 of 49 km^2/s^2 at a launch mass of 8345 kg. The vehicle will then have two major continuous thrusting periods along the trajectory totaling to 367 days of thrusting at .306 N. After 816 days, the spacecraft will perform an Earth flyby where it will begin another two major continuous thrusting periods totaling to 536 days of thrusting at .306 N. The total time of flight for transfer from Earth flyby to Titan is 3883 days. The total ΔV required for the entire arrival phase is 3.51 km/s. A spiral capture is then calculated for the spacecraft to circularize into a parking orbit at 1000 km altitude around Titan. This spiral capture begins on Jan 15th, 2037, lasting 680 days and expending 901 kg of propellant.

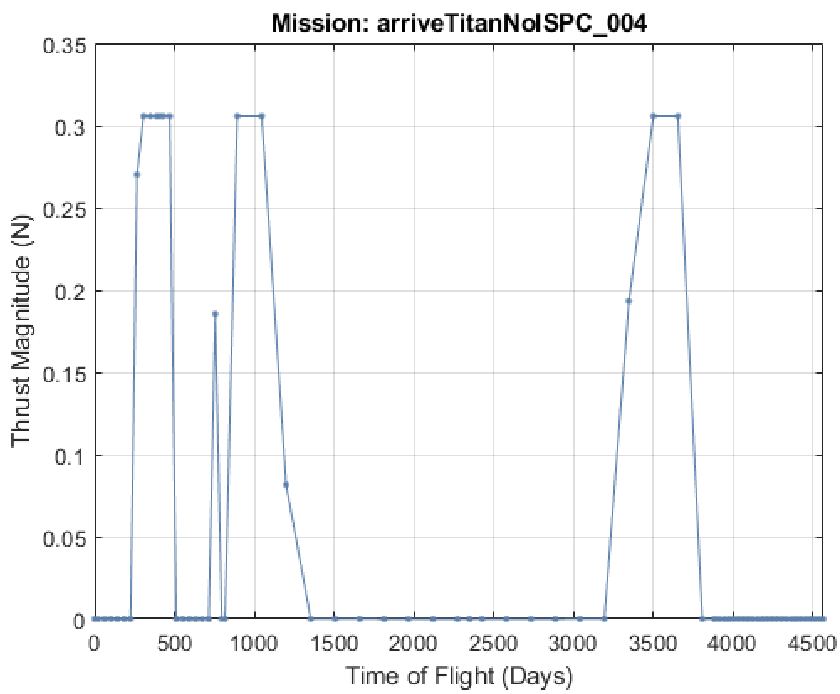

Figure 4.4-3 Thrust vs Time of Flight

Descent and ascent from Titan were modeled using simple Hohman transfers from 1000 km altitude to the surface of Titan. These maneuver ΔVs can be seen in **Table 4.4-1**.

Table 4.4-1 ΔV for Maneuvers

Maneuver	ΔV (km/s)
Descent	0.64
Ascent	1.4
Trans-Earth Injection	2.02

The return mission is modeled as a simple ballistic trajectory with a C3 of 4 km^2/s^2. This trajectory takes 2479 days and arrives at earth with an entry velocity of 11.91 km/s. Earth entry is modeled with an entry angle of 5 degrees at a mass of 64 kg and reference area of a sphere at 1.6 m diameter. This results in maximum deceleration of 30 Gs at an altitude of 51 km.

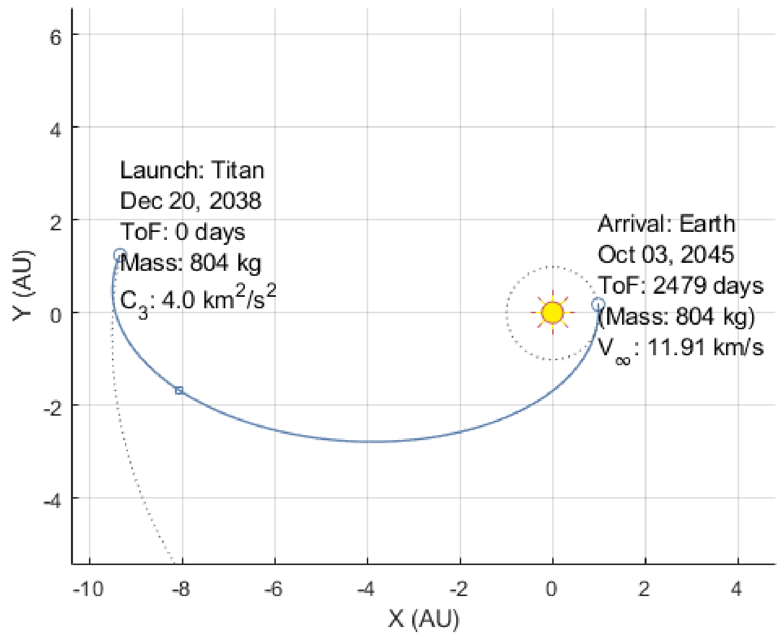

Figure 4.4-4 MALTO Earth Return

Table 4.4-2 ConOps Summary

Leg	Performance
S/C Launch	C3 = 49 km^2/s^2 Launch Date: March 30th, 2026 **Launch Mass: 8,345 kg**
Earth Flyby	Aug 23rd, 2028
Trans-Titan Low-Thrust	ΔV = 3.51 km/s TOF = 816 days
Spiral Capture	Altitude = 1000km **Spiral Time = 681 days** **Fuel Expended = 901 kg**
Titan Descent	Nov 26th, 2038 ΔV = 0.64 km/s
Sample Retrieval	**24-day stay time**
Titan Launch	ΔV = 2.8 km/s **Launch Mass: 2205 kg**
Earth Descent	**Oct 3rd, 2045** **V = 11.9 km/s reentry**

4.5 Launch Vehicle Selection

To select the launch vehicle for our mission, initial research was done in order to identify which vehicles were the best suited. The three vehicles that were selected to be used in the trade study were the Falcon Heavy, Starship, and Space Launch System (SLS). These three vehicles met the necessary ΔV requirements, the initial mass requirements, and the payload fairing capacity. The main requirements for the launch vehicles were the price, proven capability, mass margin, and the fairing sizes. These requirements were then weighed and compared in order to select the best launch vehicle for the mission.

Going through the downselection process the highest weighted requirements were the mass margin and fairing capacity. In ***Table 4.5-1*** shows the table used in conducting the trade study that led to the launch vehicle selection. The vehicles with the highest ratings for those two requirements were the SLS and the Starship, with the Falcon Heavy having the lowest for both but with a much lower cost. The Starship seemed to be the best option as it offered both of these requirements at a lower cost than the SLS. The main flaw with the Starship is that the price was an estimation, and the proven capability was lower than the other two vehicles. With all of this in mind a trade study was completed to decide on the overall best fit for the mission. Upon completion of the trade study the selection was made to go with the Space Launch System (SLS).

Table 4.5-1 Launch Vehicle Trade Study

FOM		Falcon Heavy		SLS		Starship	
Criteria	**Weight Factor**	**Utility**	**Weighted**	**Utility**	**Weighted**	**Utility**	**Weighted**
Cost	1	10	10	2	2	5	5
Mass Margin	3	6	18	9	27	8	24
Proven Capability	1	9	9	6	6	1	1
Fairing Capacity	2	7	14	8	16	10	20
Weighted Total		**51**		**52**		**50**	

As seen in the table above is the trade study used to select the launch vehicle for the mission. All three launch vehicles totals were close with the SLS having the highest weighted score of fifty-two and the Starship having the lowest weighted total of fifty. The launch vehicle that was selected was the SLS Block 1B Cargo as it was able to have a much more allowable mass margin with the new estimates. Even though the SLS has a much higher launch cost, the fairing capacity and mass margin played a much larger role in the selection of the launch vehicle.

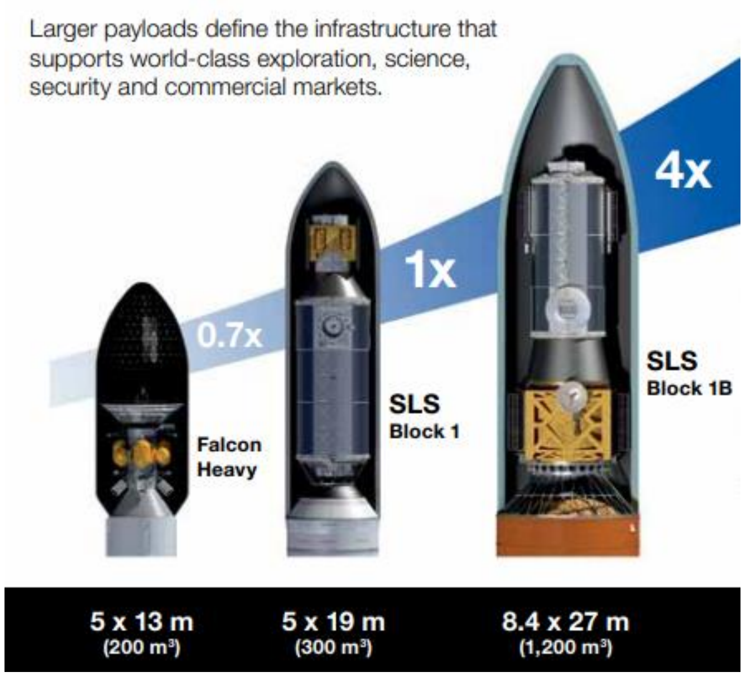

Figure 4.5-1 Payload Fairing Capacity

5 Orbiter

5.1 Derived Requirements

Table 5.1-1 includes the derived requirements developed to ensure the orbiter design satisfied mission requirements and needs.

Table 5.1-1 Orbiter Derived Requirements

Req #	Requirement
T2.1-3	The orbiter should prioritize autonomous functions
T2.4.3-1	Spacecraft ACS systems should have 3-axis controls
T2.4.3-2	Spacecraft ACS systems should allow for 180 [degree] maneuverability

5.2 Science Instruments

The purpose of the Orbiter's instrument is to gain insight into the Titan atmospheric environment and weather patterns. This will be done using the CIRS and UVIS cameras during the ground operations of the mission.

Table 5.2-1 Orbiter Science Instruments

Orbiter						
Instrument	**Purpose**	**Qty** (#)	**Total Mass** (kg)	**Total Volume** (L)	**Total Power** (W)	**Data Rate** (kbps)
CIRS	Obtain atmospheric composition or Titan to gain insight into Titan environment using IR	1	39.24	351.73	32.89	6.000
UVIS	Obtain atmospheric composition on Titan to gain insight into Titan environment using UV	1	14.46	33.12	11.83	32.1
Total			**53.7**	**384.85**	**44.72**	**38.1**

5.3 Mass Budget

The initial mass budgets were estimated using the instrument mass shown in **Table 5.2-1**. In computing the budget values, a specific percentage for each subsystem was used to calculate each element's mass based on previous missions **[1]**. The total estimated mass budget of the orbiter is 2,380.53 kg, as seen in Table 5.3-1.

Table 5.3-1 Orbiter Mass Budget

Subsystem	Budget (kg)	Calculated (kg)
Structure	151.09	384.30
Thermal	15.63	9.04
ACS	52.10	54.23
Power	109.41	450.00
Cabling	41.68	34.46
Propulsion	78.15	64.80
Telecommunications	36.47	15.24
C&DH	36.47	27.50
Total	520.99	1,039.90
Margin	120.23	239.90
Payload	95.90	85.01
On Orbit Dry	737.11	1,364.48
Propellant	1,224.90	2,150.60
Pressurant	40.00	40.00
Adapter	378.52	378.52
Launch Mass	2,380.53	3,033.60

After the estimated budget was compiled for the orbiter and its subsystems, calculated or actual masses were found for each subsystem. Going through this research and part selection for each subsystem resulted in a new calculated total mass of 3,033.60 kg, which is also reflected in **Table 5.3-1**.

5.4 Propulsion

Now to discuss the propulsion system for the orbiter, first how we selected which type of propulsion and then the specifications for the engine.

5.4.1 Propulsion System Selection

The first leg of the mission trajectory is the trans-Titan injection for a total ΔV of 2.98 km/s. Three main propulsion systems were considered: liquid bipropellant propulsion, solid propulsion, and electric propulsion. Electric propulsion carries high Isp values, as shown in Figure 5.4.1-1, but has low thrust and consumes a large amount of power. Solid propulsion carries a simpler overall design with a higher Isp than electric but is a one-time use with low maneuverability. Liquid bipropellant propulsion tends to have higher Isp than solid but carries a more complex design as well as the implications of storing fuel if it is cryogenic. The trade study compares Isp, cost, maneuverability, and the overall mass of the system to help choose the propulsion system for the orbiter. The winning system was the electric propulsion due to its high Isp and low overall mass.

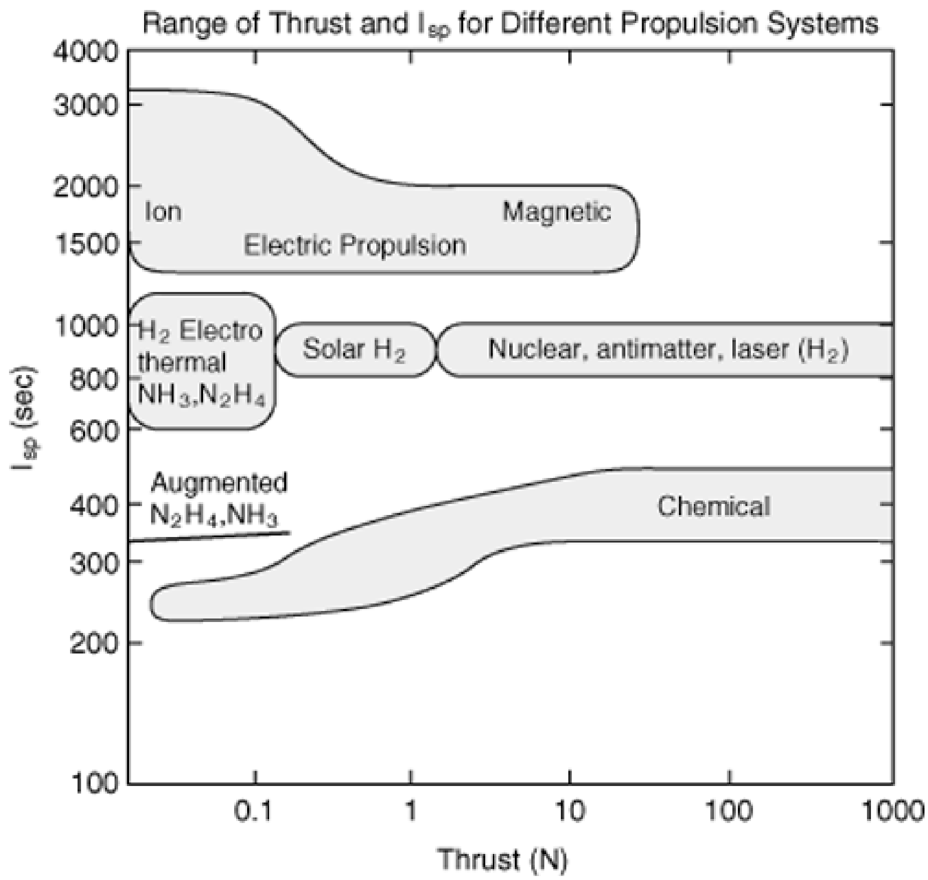

Figure 5.4.1-1 Thrust and Isp for Propulsion Systems

5.4.2 System Overview

The orbiter will use an electric engine developed by NASA, the Evolutionary Xenon Thrust (NEXT). The orbiter will carry two engines with the total propulsion system requiring a thrust of 306 mN and a minimum Isp of 2035 seconds. The overall power needed for this mission segment is about 8.704 kW. For storing the propellant, tanks with the specifications shown in Table 5.4.2-1 are to be made to store the 915.3 kg needed for the maneuver.

Table 5.4.2-1 Tank Specifications

Electric Prop (Xenon)	Arch 1
Propellant Mass	915.3kg
Tank Mass	91.53kg
Tank Volume	0.4818m^3
Tank Radius	0.4863m
Tank Thickness	0.0092m

As seen in Figure 5.4.2-1, the orbiter will carry an additional two tanks for refueling the LAV ascent stage. Refueling is the action with the highest risk, but hardware such as the Rapidly Attachable Fluid Transfer Interface (RAFTI) by Orbit Fab as seen in Figure 5.4.2-2 has been shown to be safe and reliable, though further testing will be needed and conducted.

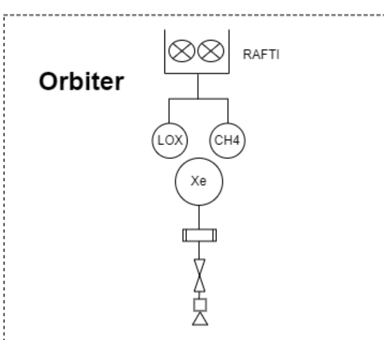

Figure 5.4.2-1 Orbiter PID

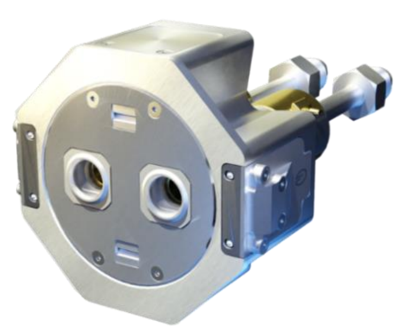

Figure 5.4.2-2 RAFTI Refueling

5.5 Structures

The Orbiter was designed using SOLIDWORKS 2021, a computer-aided design software. The structure of the orbiter is a fixed truss type structure using I-beams and T-sections. These provide the structure with high strength while maintaining a lower mass margin. Using SOLIDWORKS 2021 simulation software, the team was able to perform a static load analysis on the structure of the orbiter. The analysis included the force of the LAV assembly, which rests on the top structure of the orbiter and the force of the KRUSTY, which resides in the middle of the structure of the orbiter. In addition, the orbiter will also experience a load of six Earth Gs on its entire structure during the launch phase. The forces withstood by the orbiter are seen in Figure 5.5-1.

The orbiter structure is constructed of Aluminum Alloy 7075-T6 (SN). The team selected this material for use on the orbiter was due to its high strength properties and low mass density compared to other usable materials for spacecraft structures. The orbiter structure has a mass of 317.5 kg with a material yield stress of 505 MN/m^2. The maximum stress experienced by the orbiter is 357 MN/m^2, which is well below the material yield stress. Thus, the orbiter factor of safety is 1.42.

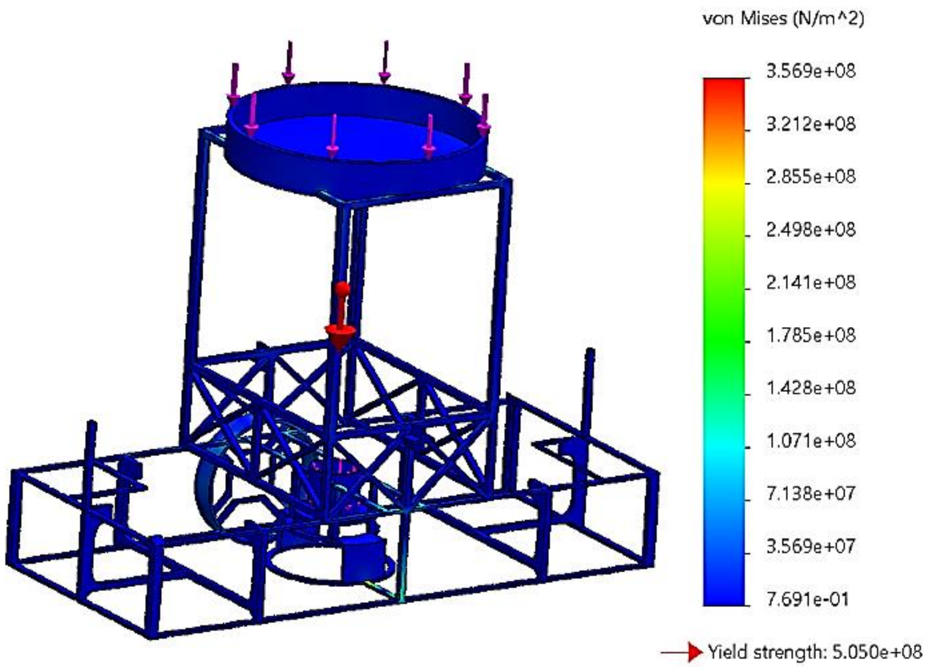

Figure 5.5-1 Orbiter Static Load Analysis

5.6 Thermal

The thermal analysis approach taken is explained in Section 5.6.1. All future results will reference back to this philosophy. The orbiter thermal results appear in Section 5.6.2.

5.6.1 Analysis Philosophy

Thermal control can be broken into active vs passive and heating vs cooling. It is important to note the following: passive cooling often requires geometric additions in shapes that affect aerodynamics and both passive and active cooling typically create a more targeted area of influence while heat tends to disperse widely. Although the orbiter will not find itself in an atmosphere negating the importance of the first consideration, the second still holds true and as such, heating was determined to be the predominant aspiration for the holistic craft. Cooling can still exist where necessary for precision instruments which need it.

The sources of heat gain/loss considered are: direct sunlight, sunlight reflected off the nearest celestial body (where applicable), natural infrared radiation from the nearest celestial body, onboard heat generation, and lastly the vehicle's heat radiated out to space (or the celestial bodies' atmospheres where applicable). These sources would be considered across the following segments: near Earth orbit, Earth half of transfer, Titan half of transfer, near Titan orbit, and finally on the Titan surface.

An expected projected area for sunlight exposure is determined from a simplified shape of the orbiter and estimations of expected orientations. Total surface area for outgoing emissions also uses this simplified shape estimate. Utilizing the equation found below whose controllable inputs from a design perspective are projected area for sunlight, exposed surface area for outgoing emissions, material emissivity and absorptivity, and desired temperature of the craft, we can solve for the output being the required heater load $Q_{heaters}$ **[1]**.

$$Q_{incoming} - Q_{outgoing} + Q_{heaters} = \epsilon A \sigma \left(T_{craft}^4 - T_{atmosphere}^4 \right)$$

The driving force behind the craft's desired temperature is solely the instrument operational temperatures found on each craft. An infographic of instruments and their associated operational temperatures can be found in Figure 5.6.1-1 below.

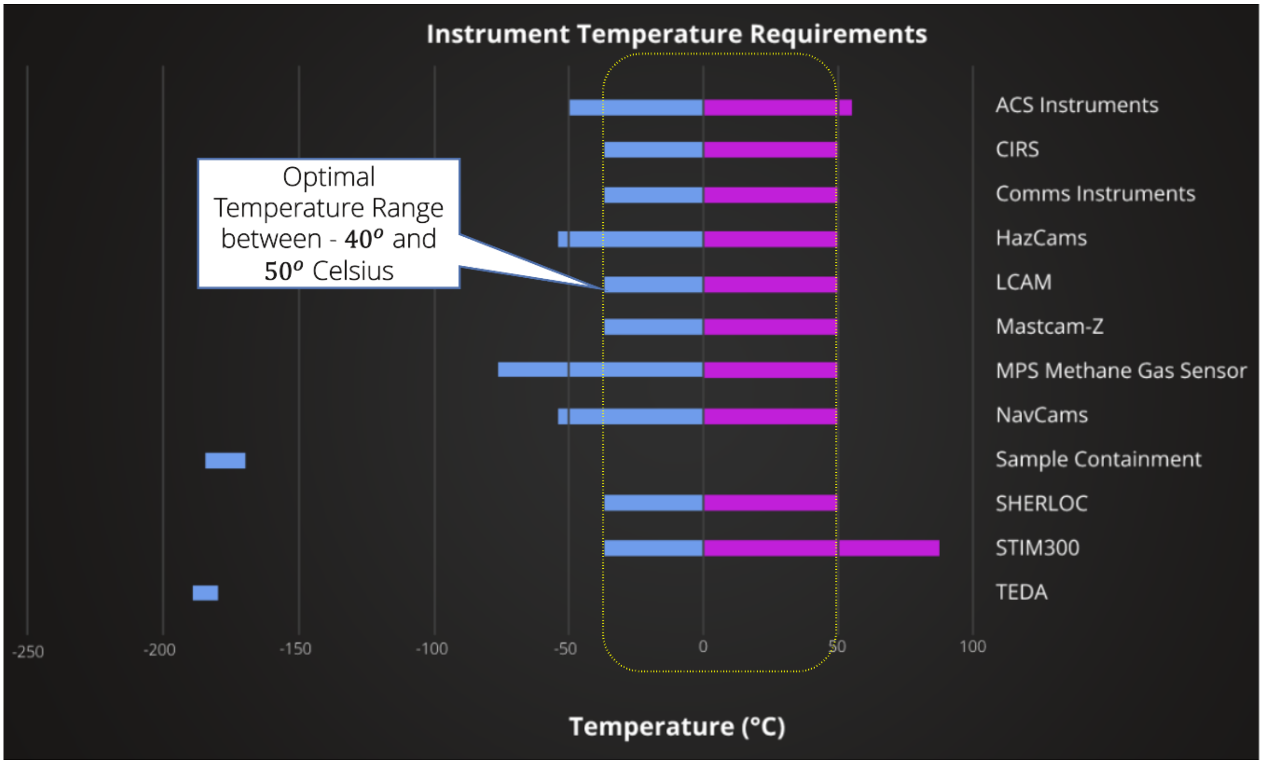

Figure 5.6.1-1 Instrument Operational Temperature Ranges

All craft sections are assumed to operate with the capacity of independent temperature ranges given their individual needs, however as most spacecraft sections happen to contain the same instruments, the optimal temperature range was found to be -40°C to 50°C. The sections of the spacecraft follow as shown by Figure 5.6.1-2, wherein the sections are identified as:

1. LAV (De-Orbit)
2. LAV (Thruster)
3. LAV(Sample)
4. Rover
5. Orbiter (Arch 1)/Breakaway Engine (Arch 2).

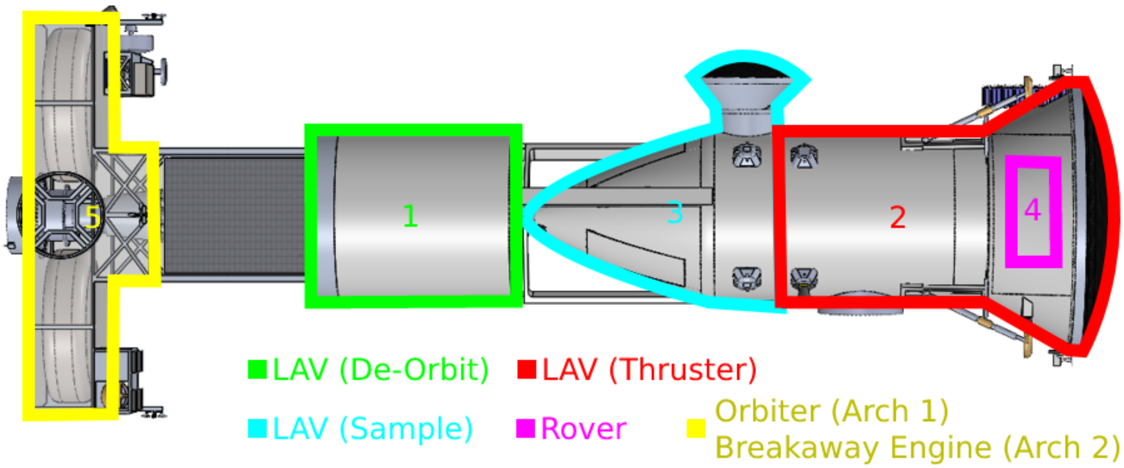

Figure 5.6.1-2 Thermal Lumped Masses

With all the above information present, design began with consideration to two unique scenarios at each segment: a worst case coldest and worst case hottest. In the worst case hottest, all assumptions are made as stated prior. In the worst case coldest, no incoming sunlight heat (direct nor reflected) would be assumed, as this could represent a period of eclipse behind a planet or perhaps the dark end of the craft facing away from the sun during travel. The worst-case cold is useful as it gives us the most extreme power draw necessary from our heaters.

5.6.2 Analysis Results

Working with the thermal analysis philosophy shared in section 5.6.1, the orbiter has an allowable temperature range of -40°C to 50°C. The orbiter has a minimum natural temperature achieved of -167°C and a maximum natural temperature achieved of -159°C. The skin material used to achieve these results is aluminum. The orbiter will require heating throughout its entire flight, hitting a maximum power draw of 639 W, to reach a controlled temperature of -40°C throughout the entirety of the mission. These results prove satisfactory. Note that, as a lumped system, the amount of heating added is not necessarily location-specific. The substantive manifestation of this heating to be added will be patch heaters spread throughout the craft. Due to true-to-life transient natures, patch heater locations will be deduced via physical testing and considerations will be made with bias against heating any imaging sensors to preserve performance.

5.7 ACS/GNC

The mission ACS and GNC system design considered past interplanetary exploration systems heritage. The orbiter's ACS/GNC systems will be operative during the transfer toward Titan with the combined spacecraft and when the orbiter is solely orbiting Titan. Optimal Euler angles will be assigned to different parts of the trajectory. After separating from the launch vehicle, a feedback control loop allows the spacecraft to maintain as close as possible to the commanded orientation at each point along the trajectory towards and around Titan. An ACS/GNC block diagram is depicted below in

Figure *5.7-1*.

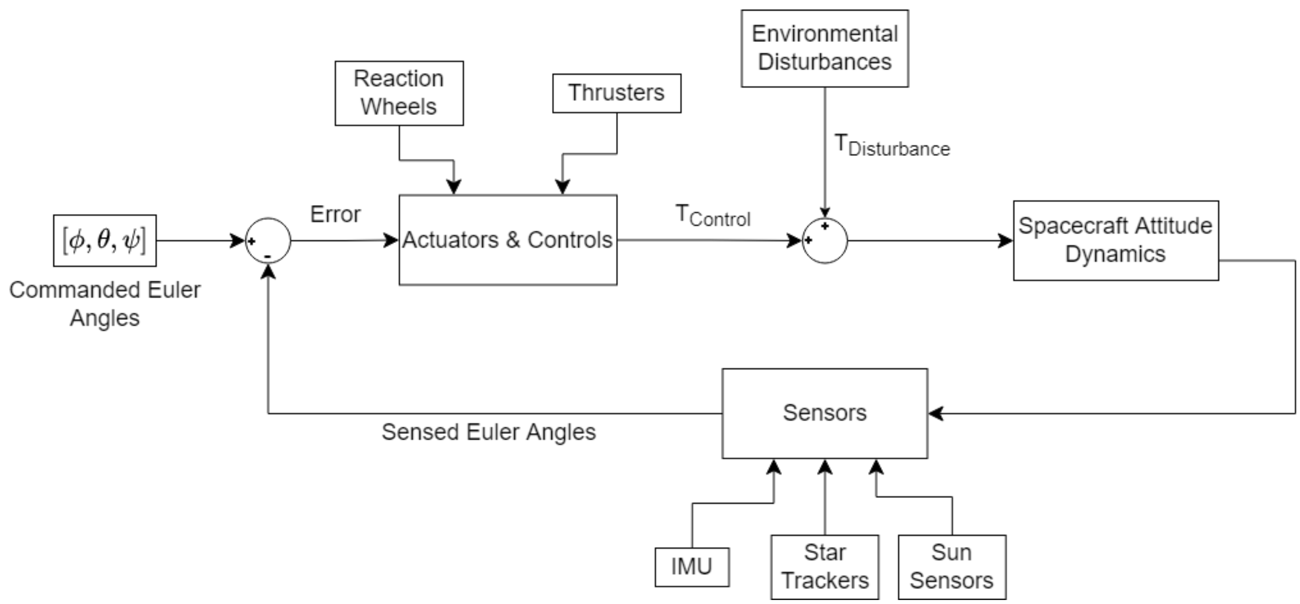

Figure 5.7-1 Block Diagram for Framework of ACS/GNC System

The orbiter will have IMUs, star trackers, and sun sensors on board to provide readings used by onboard computers to command the reaction wheels or Reaction Control System (RCS) thrusters. These commands will autonomously change the orientation of the spacecraft if the readings deviate too much from the optimal orientation of the spacecraft at a certain point of its trajectory.

Additionally, ground engineers can send commands to control the orientation of the spacecraft using the Deep Space Network (DSN). State data from the spacecraft will also be sent to Earth via the DSN throughout its mission. Table 5.7-1 below lists the sensors and control devices that will be housed on the Orbiter.

Table 5.7-1 Orbiter ACS/GNC Sensors and Control Devices

Name	Type of Device	Primary (#)	Backup (#)	Total (#)
Sun Sensors	Sensor	4	4	8
Star Trackers	Sensor	1	1	2
Inertial Measurement Units (IMUs)	Sensor	1	1	2
RCS Thrusters	Control Device	12	4	16
Reaction Wheels	Control Device	3	1	4

The orbiter will carry the majority of necessary ACS propellant required for the mission as environmental data will be sent back to Earth while the orbiter is operational. The orbiter will have a total of 155 kg of propellant. This includes all propellant needed for maneuvers and station keeping as well as 10 kg of extra propellant.

5.8 Power

The power analysis approach taken is explained in Section 5.8.1. All future results will reference back to this philosophy. The orbiter power results appear in Section 5.8.2.

5.8.1 Power Analysis Philosophy

Power needs are contingent on propellant, thermal, comms, and mechanism analyses. Propellant and thermal considerations are the predominant consideration, as they require significantly greater power. The propellant peaks are as follows: the ion engine peaks twice at 8.5 kW on route to Titan, and once at 8.5 kW during spiral capture towards Titan. The thermal peaks all occur during the Earth to Titan transfer and are as follows: LAV (De-Orbit) requires 703 W, LAV (Thruster) 309 W, LAV (Sample) 417 W, and Orbiter 639 W. With all peaks and mission trajectories considered, the pertinent minimum needs come out to be a power lifetime greater than 10 years, an orbiter power generation of at least 9.1 kW, and an LAV power generation of at least 1.4 kW. Figure 5.8.1-1 introduces the generation vs use duration design points for each technology.

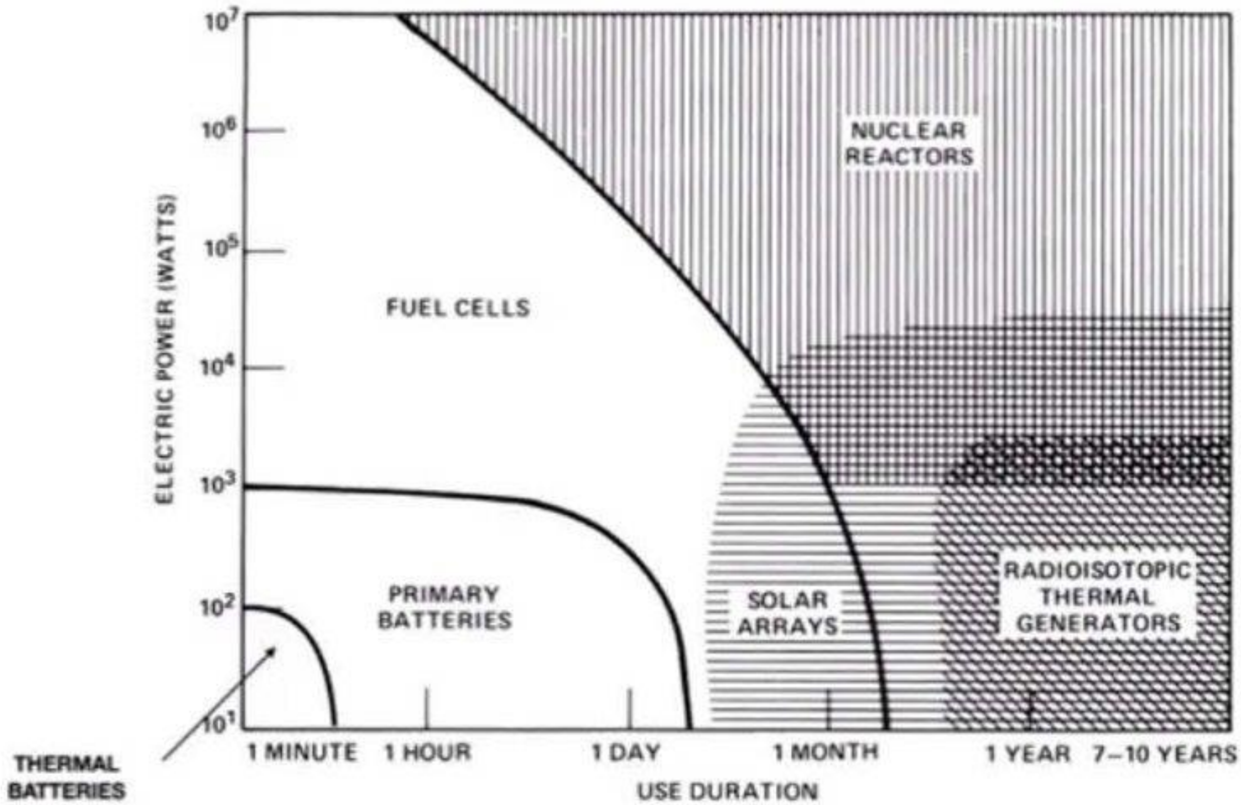

Figure 5.8.1-1 Power Source Design Points

As can be seen via the above infographic, thermal batteries, primary batteries, and fuel cells simply host too short of a lifetime to be utilized in the intended mission. This leaves solar arrays, RTGs, and nuclear reactors. Via analysis, solar arrays are entirely infeasible due to their unrealistic volume and mass requirements to generate necessary power out towards Titan. This leaves RTGs and nuclear reactors. RTGs were decidedly used where applicable. Although technically possible to utilize RTGs for large draws such as the Orbiter's 9.1 kW requirement, they effectively lapse their utility due to the sheer number of them required to hit such a target generation. For the orbiter, a nuclear reactor is effectively a necessity.

5.8.2 Analysis Results

Working with the power analysis philosophy shared in section 5.8.1, the orbiter requires 9.1 kW and utilizes a 10 kW nuclear reactor power source known as KRUSTY. The benefits of using the KRUSTY are that it is a single unit, has a high power density, maintains extreme EOL power retention, and is a promising technological leap. The cons of implementing the KRUSTY are that it boasts a lower TRL, brings with it the burden of an absurdly large thermal load, holds connotations yielding shaky public perception, and finally lacks mission experience. A validated test rig as well as the team's interpretation of a functional in-flight model derived from the validation research's ideas can be seen below in Figure 5.8.2-1.

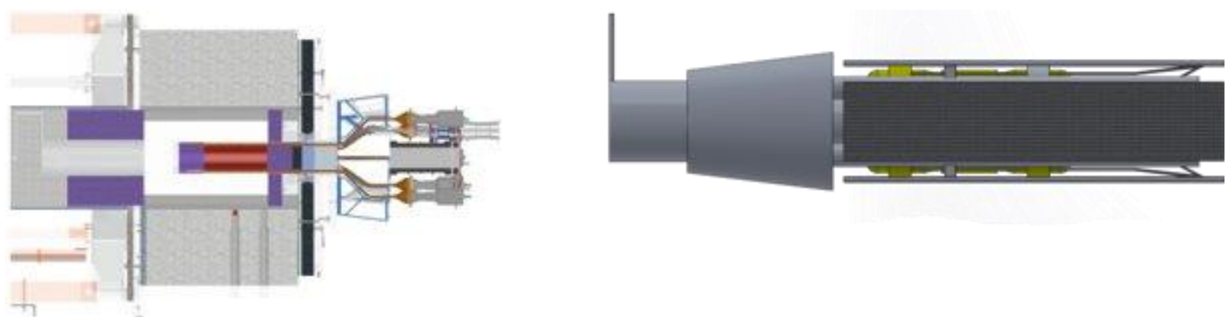

Figure 5.8.2-1 Test Rig (Left) vs In-Flight (Right)

While using a nuclear reactor which has yet to be proven flight-capable brings risks, KRUSTY is the amalgamation of work materialized not only under the Kilopower initiative at NASA headed by Marc Gibson, but of milestones of progress contributed by many others as well. Mr. Gibson began his work in 2007 holding responsibility for "the engineering and development of nuclear systems for in-space and planetary surface power in support of the Space Technology Mission Directorate" **[5]**. In 2010, the planetary decadal survey suggested a technology assessment study with hopes of determining if fission reactors can provide an alternative to radioisotope systems. In 2012, a proof-of-concept test successfully demonstrated that using flattop fissions could produce 24 Watts. In 2014, NASA conducted a nuclear power assessment study as many wondered if small fission was suitable for “Titan/Saturn system missions” or a “Uranus orbiter probe” **[6]**. Finally in 2015, the Kilopower project was kickstarted under the STMD Game Changing Development Program with the timeline of three years, $20 million budget, and the goal to build and test a prototype reactor. Build and test a prototype reactor is exactly what the team did, successfully might one add. Whilst presenting their work, the following considerations and acknowledgements were made:

- KRUSTY was designed from the ground up with flight-like components. This doesn’t intrinsically suggest a test rig on Earth independently validates in-flight capabilities, however it is promising in the sense that the transition from an on-ground test rig into a flight-ready model would be optimally unhindered.
- KRUSTY adheres to all Federal and DOE regulations including 10CFR830.
- An independent interagency nuclear safety review panel would conduct a NEPA review prior to flight.
- The reactor design includes inherent fault tolerance, so that loss of cooling would result in an automatic reduction in fission power that prevents uncontrolled scenarios.

5.9 Communications

For this design, all data from the rover and LAV will be transmitted through the orbiter and sent back to Earth. The rover and LAV will be capable of sending data to Earth without the orbiter as an emergency scenario if communication with the orbiter is not possible, however more power will be needed for the LAV and rover in these scenarios.

To begin calculations for the communications design, all sensors and equipment that would be needed for state of health of the spacecraft and scientific equipment were obtained and each of their data rates were determined. Once this information was obtained, the maximum distance between Earth and Titan was determined and used for all communications calculations between the orbiter and the ground station at Earth **[1]**. With the given data rates shown in **Table 5.9-1**, a transmitter power of 1 Watt, antenna diameter of 2 meters, and an X-band frequency of 8.4 GHz was determined to be the best combination to fit the design needs of the communication system. Reed-Solomon encoding was used for the design to achieve the symbol rate shown in **Table 5.9-1** **[1]**.

Table 5.9-1 Orbiter Instrument Data Rates

Orbiter Max Range Instrument Data Rates		
Bradford Sun Sensor (16)	704	bps
Max Rover Data Consumption Rate	69692.45	bps
Max Lander Data Consumption Rate	1811072	bps
CIRS	6000	bps
Star Tracker (2)	235200	bps
IMU (2)	1536000	bps
State of health	32500	bps
UVIS	30100	bps
Max Data Consumption	3721268	bps
Symbol Rate	4255.262	ksps

Continuing with the Max Range Carrier Link Performance shown in **Table 5.9-2**, the values shown were determined using **Table 5.9-2** and *Brown's Elements of Spacecraft Design*. The carrier link margin was determined to be suitable for even the worst-case scenario of the spacecraft.

Table 5.9-2 Max Range Carrier Link Performance

Max Range Carrier Link Performance		
System Noise Density	-216.047	dB/Hz
Modulation Index	68	degree
Carrier Power/Total Power	-8.52849	dB
Received Carrier Power	-144.139	dB
Carrier Noise Bandwidth	14.77	dB-Hz
Carrier Signal to Noise	57.1383	dB
Carrier Signal to Noise Req.	10	dB
Carrier Link Margin	47.1383	dB

Lastly, for the Max Range Link Performance seen in **Table 5.9-3**, the required performance ratio of 4.5 was achieved giving assurance that communications will be performed at all stages of the mission using the maximum data received by the orbiter, LAV and the rover.

Table 5.9-3 Max Range Link Performance

Max Range Link Performance		
Data Power/Total Power	-0.65668	dB
Data Power Received	-136.267	dB
Data Symbol Rate	-66.2893	dB-Hz
Eb/No	13.49087	
Eb/No Required	4.5	
Margin	8.99086	

Shown below in

Table 5.9-4 is a list of the telecommunication hardware used by the orbiter during non-emergency situations. The components denoted with 0 W of power will be used only during emergency situations.

Table 5.9-4 Orbiter Telecommunications Hardware (Non-Emergency Comms)

Orbiter Telecomm Hardware (Non-Emergency)		
Component	Mass (kg)	Power (W)
2m Parabolic Dish	21.2	30
Biconical LGA (4)	1.816	0
Turnstile Antenna	0.3	0
X-Band SSPA (2)	2.74	31
Deep Space Transponder (2)	6.4	0
S-Band SSPA (1)	1.98	0
S-Band Transponder	2.95	20
TWTA (2)	6.2	134
Coax Cable	7.8	0
Diplexer	0.5	0

Attenuator	0.1	0
Total:	51.976	215

Shown below in **Table 5.9-5** is a summary of our requested time on the DSN. During the first 3,902 days only state of health and position data will be sent. During the approximately 30 days on Titan, communications via the DSN will include state of health data along with 20% of collected science data. For the first 435 days during Earth return, all science data along with state of health data will be sent back. After all science data has been dumped, communications will return to state of health data and data pertinent to the payload.

Table 5.9-5 Requested DSN Scheduling Time

Requested DSN Time		
Transit to Titan	1.81 hrs/day	3,902 days
Titan Surface Ops	16 hrs/day	30 days
Earth Return 1	3.24 hrs/day	435 days
Earth Return 2	1.81 hrs/day	1841 days
Total	12,260 hours	6,208 days

5.10 C&DH

The orbiter will implement two RAD 5545 radiation hardened multi-core processors manufactured by BAE Systems to manage the data and commands for the mission. They will be wired in parallel for redundancy in case of a failure. Along with these processors, the orbiter will utilize a 1 TB solid-state recorder for data storage. **Table 5.10-1** shown below summarizes the mass and power of each component used by the orbiter's command and data handling system.

Table 5.10-1 Orbiter C&DH Hardware

Orbiter C&DH Hardware		
Component	**Mass (kg)**	**Power (W)**
RAD5545 Processor (2)	4	35
1 TB Storage	8	22
Total:	12	57

6 Landing Ascent Vehicle

The Landing Ascent Vehicle consists of multiple stages and will conduct mission operations from Titan arrival through Earth return. The following sections will look at the various subsystems of the LAV.

6.1 Derived Requirements

Table 6.1-1 includes the derived requirements developed to ensure the LAV designs satisfies mission requirements and needs.

Table 6.1-1 Landing Ascent Vehicle Derived Requirements

Req #	Requirement
T2.1-2	The liquid sample storage system should be capable of simulating the environment on the surface of Titan from the time of sample collection to Earth return
T2.1-4	The landing and ascent stages should be autonomous vehicles
T2.4.3-1	Spacecraft ACS systems should have 3-axis controls
T2.4.3-2	Spacecraft ACS systems should allow for 180° maneuverability
T2.3-1	ACS systems should incorporate redundant sensors and thrusters
T2.5-1	Spacecraft computers should include redundant or overlapping hardware and autonomous fault detection and correction software
T2.1.1-1	Sample storage and in-situ measurement systems shall be operational in Titan surface temperature conditions

6.2 Mass Budget

Using the same initial process as seen in section 5.3 for creating the mass budget resulted in an estimated mass budget for the LAV a total of 2,105.25 kg. During this estimation, it became noticeable that the mass budget was far from the calculated masses. To work around this discrepancy, the budget was recreated by doing more research of similar missions that involved an ascent vehicle and lander. Then, the final estimate came out significantly closer to the calculated mass, as seen in **Table 6.2-1**.

Table 6.2-1 Mass Breakdown of the LAV

Subsystem	Budget (kg)	Calculated (kg)
Structure	616.38	135.60
Thermal	63.76	64.80
ACS	212.55	40.50
Power	446.35	756.05
Cabling	170.04	165.23
Propulsion	318.82	41.00
Telecommunications	148.78	15.24
C&DH	148.78	27.50
Total	2,125.45	1,245.92
Margin	490.49	287.52
Payload	159.93	97.63
On Orbit Dry	2,775.87	2,876.99
Propellant	519.50	892.10
Pressurant	18.00	79.80
Launch Mass	3,313.37	3,448.89

6.3 Propulsion

The next stage of the trajectory is the descent, which required a total ΔV of 0.639 km/s. Only solid and liquid bi-propellant were considered since electric propulsion would not have sufficient thrust for descent. The figures of merit were system mass, system volume, burn time, and the load the spacecraft will incur from acceleration. As seen in **Table 6.3-1** below, bipropellant was the winner due to the low system mass.

Table 6.3-1 Liquid vs Solid Propulsion Trade Study

Utility Value (1-9)		Liquid Propulsion (LOX/Methane)		Solid Rocket Propulsion	
Criteria	Weight Factor	**Utility**	**Weighted**	Utility	Weighted
System Mass	4	**9**	**36**	3	12
System Volume	2	**3**	**6**	7	14
Burn Time	3	**7**	**21**	7	21
S/C Acceleration	1	**9**	**9**	9	9
Weighted Total		**72**		56	

The team decided to choose LOX/Methane(CH4) propellant to eliminate the need for multiple tanks and engines between the LAV and orbiter. When looking into available engines, the team decided it was best to engage another company. The system has a required thrust of 15 kN and Isp of 330 seconds. With this, the descent vehicle will burn for about 177 seconds with a descent acceleration of 2.081Gs.

The descent propulsion system will carry 2 tanks of identical size for simplicity for the LOX/Methane. Table 6.3-2 shows the tank specifications:

Table 6.3-2 Lander Tank Specifications

Liquid Biprop	
Propellant Mass	734.6kg
Tank Mass	78.65kg (x2)
Tank Volume	0.6556m^3
Tank Radius	0.5389m
Tank Thickness	0.0081m

The LAV ascent stage will require a ΔV of 1.39 km/s to enter Titan orbit where it will rendezvous with the orbiter and refuel before heading back to Earth, requiring a ΔV of 2.02 km/s. The engine will have a thrust of 25kN and an Isp of 330 seconds. More specifications for the ascent vehicle engine are seen in **Table 6.3-3**.

Table 6.3-3 Liquid Biprop Specifications

Liquid Biprop	**Architecture 1**
$\Delta V_{\text{trans-earth Injection}}$	2.02km/s
Thrust	25kN
Isp	330s
Ascent Burn Time	32.41s
Ascent Acceleration	3.177G
TEI Burn Time	76.91s
TEI Acceleration	3.177G

This thrust was chosen to minimize thermal burden on the engine and stress burden on the structure. The ascent to the orbiter needs about 802.1 kg of propellant while the trans earth injection requires 1300 kg of propellant. Thus, the tanks in the orbiter and ascent vehicle will be identical, with dimensions as shown in **Table 6.3-4** below.

Table 6.3-4 Ascent Vehicle Tank Specifications

Liquid Biprop	**Architecture** 1
Propellant Mass (TEI)	1299kg
Tank Mass	139.1kg (x2)
Tank Volume	1.159m^3
Tank Radius	0.6517m
Tank Thickness	0.0098m

6.4 Structures

The LAV was designed using SOLIDWORKS 2021, a computer-aided design software. Simulations were completed with SOLIDWORKS Simulation.

6.4.1 Descent Stage Structures

The structure of the LAV is a ski-stringer type structure. These provide the structure with high strength and while minimizing the mass margin. Using SOLIDWORKS 2021 simulation software, the team was able to perform a static load analysis on the structure of the descent stage. The structure experiences a thruster load of 15 kN on the rocket engine holder, and propellant tank retention load of 17.8 kN. The forces applied on the structure are seen in Figure 6.4.1-1**.**

The descent stage structure is constructed of Aluminum Alloy 7075-T6 (SN). The team selected this material for use on the descent stag due to it high strength properties and low density. The descent stage structure has a mass of 100.5 kg with a material yield stress of 505 MN/m^2. The maximum stress experienced by the descent stage is 366 MN/m^2, which is well below the material yield stress. This gives us a descent stage factor of safety of 1.38.

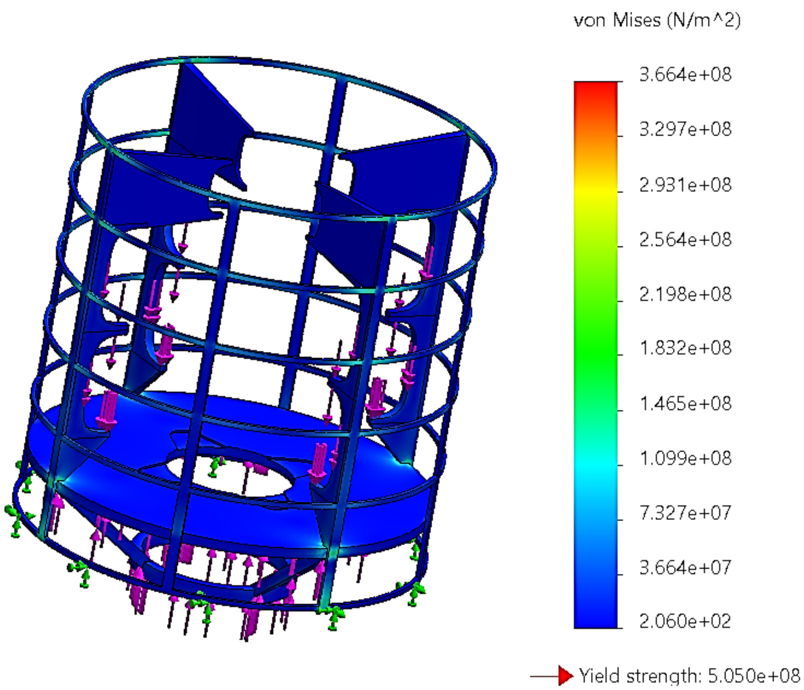

Figure 6.4.1-1 Descent Stage Static Load Analysis

6.4.2 Lander Legs Structures

Analysis on the lander leg was designed and performed using a similar methodology. The lander leg is constructed of Aluminum Alloy 7075-T6 (SN). Each of the 3 LAV lander legs has a mass of 30 kg. The LAV has a landing velocity of 1 m/s. Each lander leg has a spring with a spring constant of 3.44 kN/m. The LAV lander legs each experience 1.21 kN of force. The material has a yield stress of 505 MN/m^2. The maximum stress experienced by each lander leg is 335 MN/m^2, which is well below the material yield stress. Each lander leg has a factor of safety of 1.51. The static analysis of a singular lander leg is shown in Figure 6.4.2-1.

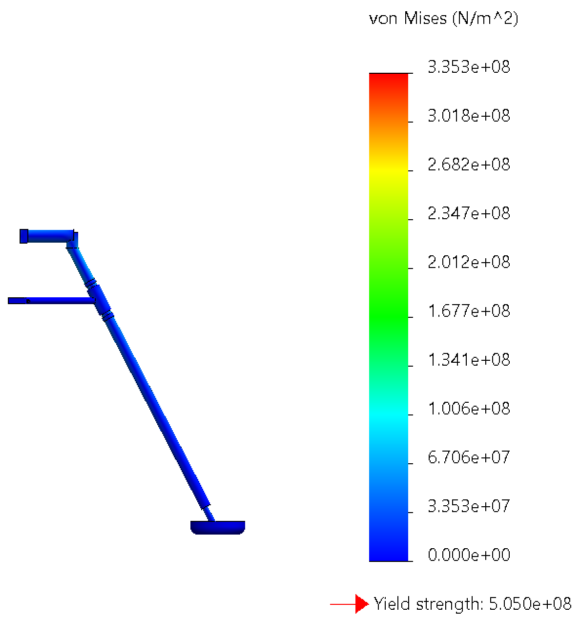

Figure 6.4.2-1 Lander Leg Static Load Analysis

6.4.3 Ascent Stage

The structure of the LAV is a skin-stringer structure type. These provide the structure with high strength and helps maintain a lower mass margin. The structure experiences a thruster load of 25 kN on the rocket engine retention, and propellant tank load of 30 kN on the tank retention. The forces applied on the structure are seen in Figure 6.4.3-1**.**

The ascent stage structure is constructed of Aluminum Alloy 7075-T6 (SN). The descent stage structure has a mass of 125.5 kg with a material yield stress of 505 MN/m^2. The maximum stress experienced by the ascent stage is 409 MN/m^2, which is well below the material yield stress. This gives us an ascent stage factor of safety of 1.23.

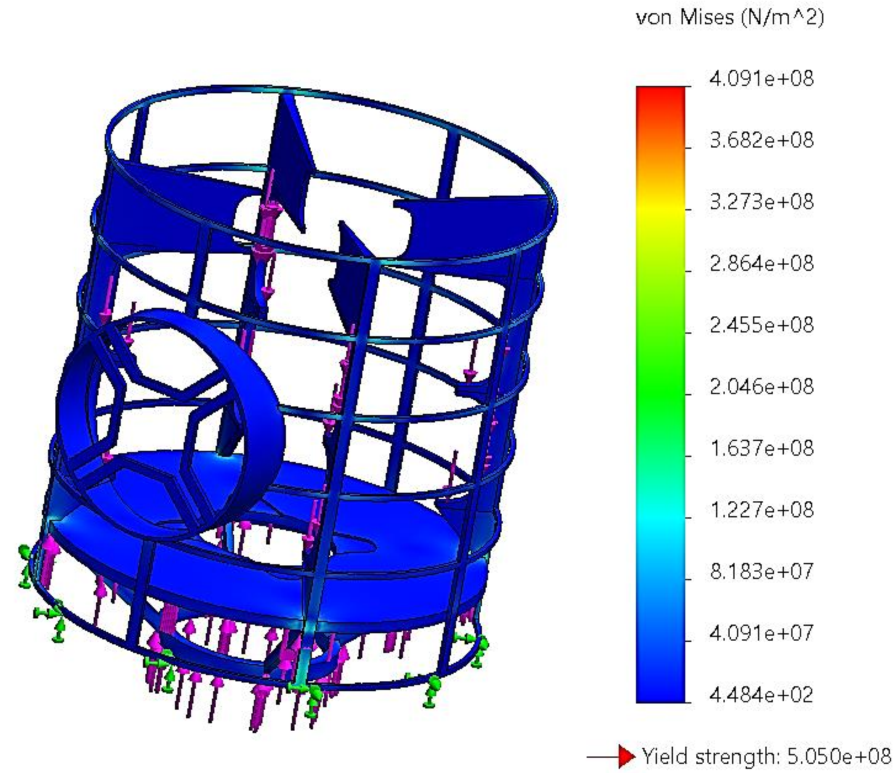

Figure 6.4.3-1 Ascent Stage Static Load Analysis

6.5 Thermal

Thermal analysis for the LAV was split into five lumped masses and a standalone heat shield analysis. All results will be found in their respective subheading below. Section 6.5.1 through Section 6.5.3 use the thermal analysis philosophy shared in Section 5.6.1.

6.5.1 De-Orbit Module

The LAV de-orbit module has an allowable temperature range of -40°C to 50°C. The LAV de-orbit module has a minimum natural temperature of -273°C and a maximum natural temperature of -77°C. The skin material used to achieve these results is aluminum. The LAV de-orbit module will require a maximum heating throughout its entire flight of 703 W to reach a controlled temperature of -40°C throughout the entirety of the mission. Most of this heating will be via patch heaters spread throughout the craft.

6.5.2 Thruster Module

The LAV thruster module has an allowable temperature range of -40°C to 50°C. The LAV thruster module has a minimum natural temperature of -66°C and a maximum natural temperature of 28°C. The skin material used to achieve these results is aluminum. The LAV thruster module will require a maximum heating throughout its entire flight of 309 W to reach a controlled temperature of -40°C to 28°C throughout the entirety of the mission. Most of this heating to be added will be patch heaters spread throughout the craft.

6.5.3 Sample Module

The LAV sample module has an allowable temperature range of -40°C to 50°C. The LAV sample module has a minimum natural temperature of -79°C and a maximum natural temperature of 48°C. The skin material used to achieve these results is aluminum. The LAV sample module will require a maximum heating throughout its entire flight of 417 W to reach a controlled temperature of -40°C to 48°C throughout the entirety of the mission. The substantive manifestation of this heating to be added will be patch heaters spread throughout the craft.

6.5.4 Heat Shields

Heat shield analysis was performed with a fundamentally different thermal philosophy at its core. Unlike the other thermal analyses assuming a lumped body at steady state, the heat shield will experience drastic temperature changes in a short period of time which would ultimately yield quite a non-negligible transient throughout the thickness of the shield. So, external scholarly publications and journals will be referenced to assist with the heat shield validation process. The

SOLIDWORKS analysis results are to be considered a value ceiling as ablation would act primarily to cool temperatures. Utilizing standard PICA material properties found in Thornton along with a hybridization of independently calculated descent values and expected Titan descent values for Dragonfly found in Wright, SOLIDWORKS analysis yields that the external surface temperature reached is 2,156°C **[7,8]**. A visual of said analysis can be seen in Figure 6.5.4-1 below. SOLIDWORKS alleges an internal face temperature of ~600°C which doesn't follow suit of any results found across numerous publications in the field. As such, however, due to the exterior surface temperature reaching satisfactory levels in comparison to the ~2,500°C found in Covington, the assumption is being made that the internal temperatures reached would be, at most, 50°C **[3]**. The internal face of the heat shield falls within the internal components' allowable ranges. In conclusion, with the thickness, materials, and descent experience considered, heat shields internally and externally reach a satisfactory temperature and will not degrade in an unexpected manner or threaten sensitive components and payloads of craft modules. So, the team's heat shield configurations are deemed satisfactory.

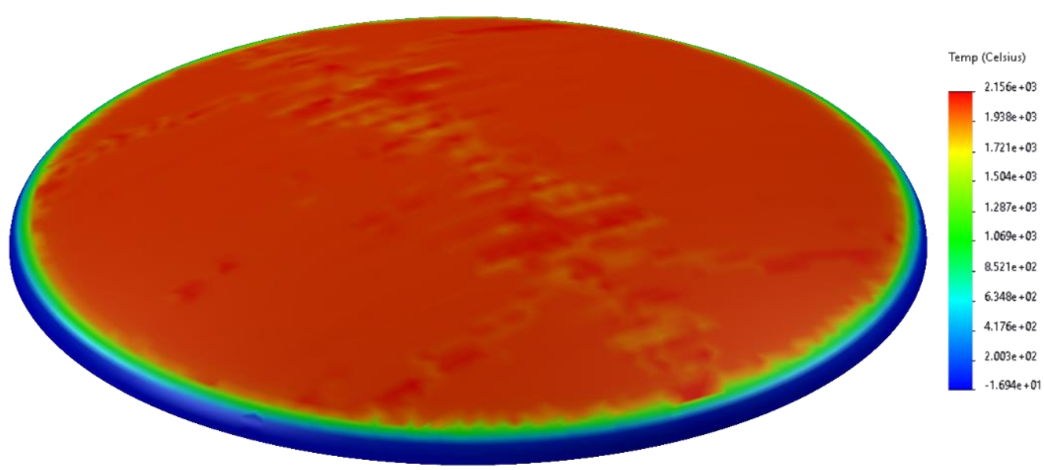

Figure 6.5.4-1 SOLIDWORKS Heat Shield Analysis

6.6 ACS/GNC

The GNC System for the LAV will be mostly identical to the GNC system of the orbiter and use the exact same components. However, the ACS system has 16 additional ACS thrusters for more maneuverability during the crucial EDL phase of the mission. **Table 6.6-1** below shows a list of ACS/GNC sensors and control devices that will be on the LAV.

Table 6.6-1 LAV ACS/GNC Sensors and Control Devices

Name	Type of Device	Primary (#)	Backup (#)	Total (#)
Sun Sensors	Sensor	4	4	8
Star Trackers	Sensor	1	1	2
Inertial Measurement Units (IMUs)	Sensor	1	1	2
RCS Thrusters (MR-103M)	Control Device	12	4	16
RCS Thrusters (MR-107S)	Control Device	12	4	16
Reaction Wheels	Control Device	3	1	4

In addition to the ACS/GNC list above, the LAV carries other sensors for relative navigation and docking with the Orbiter. Radar ranging instruments will allow the LAV to determine relative position and velocity to the Orbiter to assist it in docking. The LAV will carry 40 kg of ACS propellant. This includes all propellant needed for maneuvers as well as 15 kg of extra propellant.

6.7 Power

Working with the power analysis philosophy shared in section 5.7.1, the LAV requires 1.4 kW and uses three 500 W Next Gen RTGs stored in the sample module’s nose cone. A visualization of the mechanism maneuvering the RTGs between internal (aerodynamic) states and external (cooled) states can be seen below in *Figure 6.7-1*.

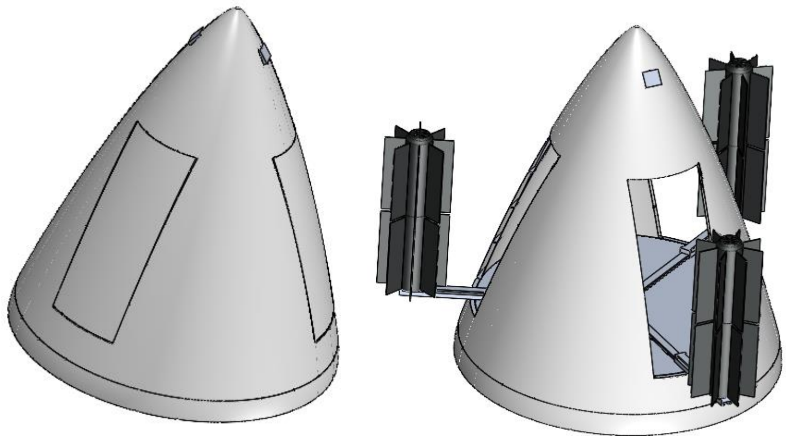

Figure 6.7-1 LAV RTG Mechanism

6.8 Communications

The communication rate and data calculations methods for the orbiter and LAV were discussed Section 5.9 since all data from the LAV will be transmitted through the orbiter and sent back to Earth. The LAV will be capable of sending data to Earth without the orbiter as an emergency scenario if communications are not possible, however more power will be needed for the LAV and rover in these scenarios. With the given data rates shown in **Table 6.8-1**, a transmitter power of 1 Watt, antenna diameter of 1 meter, and a S-band frequency of 3 GHz was determined to be the best combination to fit the design needs of the communication system. Reed-Solomon encoding was used for the design to achieve the symbol rate shown in **Table 6.8-1** [1].

Table 6.8-1 LAV Max Range Instrument Data Rates

LAV Max Range Instrument Data Rates		
Bradford Sun Sensor (16)	704	bps
Star Tracker (2)	235200	bps
IMU (2)	1536000	bps
Lander Vision Camera	0	bps
State of health	32500	bps
Surface Camera (2)	6668	bps
Max Data Consumption	1811072	bps

Symbol Rate	2070.957	ksps
Cable Loss	-0.06	dB
Antenna Diameter	1	m
Frequency (S-Band)	3	GHz
Antenna Gain	27.34243	dB
EIRP	57.28243	dB
Free Space Path Loss	-168.003	dB
Atmospheric Attenuation	-0.15	dB
Estimated Polarization Loss	-0.2	dB
LAV Receiver Gain	3.1	dB
Pointing Loss	-0.1387	dB
Beam Width	0.930133	degrees
Receiver Cable Loss	0	dB
Total Received Power	-108.109	dB
Receiver System Noise	18	K

Continuing with the Max Range Carrier Link Performance shown in **Table 6.8-2** the values shown were determined using **Table 6.8-1** and Brown's Elements of Spacecraft Design, as well **[1]**. The carrier link margin was determined to be suitable for even the worst-case scenario of the spacecraft.

Table 6.8-2 Max Range Carrier Link Performance

Max Range Carrier Link Performance		
System Noise Density	-216.047	dB/Hz
Modulation Index	68	degree
Carrier Power/Total Power	-8.52849	dB
Received Carrier Power	-116.638	dB
Carrier Noise Bandwidth	14.77	dB-Hz
Carrier Signal to Noise	84.63948	dB
Carrier Signal to Noise Req.	10	dB
Carrier Link Margin	74.63948	dB

Lastly, for the Max Range Link Performance seen in **Table 6.8-3**, the required performance ratio of 4.5 was achieved giving assurance that communications will be performed at all stages of the mission using the maximum data received by the orbiter and LAV with the capability to communicate directly with Earth.

Table 6.8-3 Max Range Link Performance

Max Range Link Performance		
Data Power/Total Power	-0.65668	dB
Data Power Received	-108.766	dB
Data Symbol Rate	-63.1617	dB-Hz
Eb/No	44.1196	
Eb/No Required	4.5	
Margin	39.6196	

Shown below in **Table 6.8-4** is a list of the telecommunication hardware used by the LAV during non-emergency situations. The components denoted with 0 W of power will be used during emergency situations.

Table 6.8-4 LAV Telecommunication Hardware (Non-Emergency Comms)

LAV Telecomm Hardware (Non-Emergency)		
Component	**Mass (kg)**	**Power (W)**
Turnstile Antenna	0.3	0
Parabolic Dish	6.41	30
X-Band SSPA (1)	1.37	0
Deep Space Transponder	3.2	0
S-Band SSPA (1)	1.98	31
S-Band Transponder	2.95	20
Coax Cable	7.8	0
Diplexer	0.50	0
Attenuator	0.1	0
Total:	24.61	81

6.9 C&DH

The LAV will be implementing two RAD 5545 radiation hardened multi-core processors manufactured by BAE Systems to manage the data and commands for the mission. The two will be wired in parallel for redundancy in case of a failure. As previously mentioned, these processors are radiation hardened and are certified for flight. Along with these processors, the LAV will utilize a 1 TB solid-state recorder for data storage. **Table 6.9-1** shown below summarizes the mass and power of each component used by the LAV's command and data handling system.

Table 6.9-1 LAV C&DH Hardware

LAV C&DH Hardware		
Component	Mass (kg)	Power (W)
RAD5545 Processor (2)	4	35
1 TB Storage	8	22
Total:	12	57

7 Rover

The following sections detail the subsystems and hardware on the rover. The rover carries out its nominal mission autonomously.

7.1 Derived Requirements

Table 7.1-1 includes the derived requirements developed to ensure the rover design satisfied mission requirements and needs.

Table 7.1-1 Rover Derived Requirements

Req #	Requirement
T2.1-5	The rover shall be an autonomous vehicle
T2.1.1-1	Sample storage and in-situ measurement systems shall be operation in Titan surface temperature conditions
T2.1-1	Rover shall be able to travel to the sample collection site from the landing site
T2.5-1	Spacecraft computers should include redundant or overlapping hardware and autonomous fault detection and correction software

7.2 Science Instruments

The Rover's camera instruments are used to locate Bolsena Lacus, document its mission, and return to the LAV. The SHERLOC Context Imager is used to analyze the liquid samples obtained from Bolsena Lacus to determine the existence of Vinyl Cyanide and to determine if any metabolic processes are taking place within the sample.

Table 7.2-1 Rover Science Instruments

Instrument	Purpose	**Qty** (#)	**Total Mass** (kg)	**Total Volume** (L)	**Total Power** (W)	**Total Data Rate** (kbps)
Surface Camera (Mastcam-Z)	Obtain color images of Titan's surface from perspective of the rover	1	4	5.31	17.4	1.667
SHERLOC Context Imager	Analyze the composition of the liquid sample using laser light	1	3.11	3.39	48.8	3.333

Nav Cams	Maps out contours of the area around the rover to help mission team make decisions on how to move the rover	2	0.44	0.38	4.3	0.300
HazCams	Aids in autonomous navigation by looking for hazards such as large rocks	2	0.49	0.38	4.3	0.300
Container for Sample	Holds sample on return trip to Earth	6	0.5	1	0	0
SUM			**8.54**	**10.46**	**74.8**	**5.600**

7.3 Mass Budget

For the rover, an initial mass budget was created using the Brown textbook which estimated the mass based off the instruments on board the rover **[1]**. Further research and calculations were done to create a new mass budget still using the Brown method from the textbook. The new mass budget is 1,203.57 kg, as seen in **Table 7.3-1**.

Table 7.3-1 Mass Breakdown for the Rover

Subsystem	Budget (kg)	Calculated (kg)
Structure	268.91	163.90
Thermal	27.82	9.66
ACS	92.73	0.00
Power	194.73	25.65
Cabling	74.18	25.63
Propulsion	139.09	0.00
Telecommunications	64.91	15.99
C&DH	64.91	10.00
Total	927.28	250.83
Margin	213.99	57.88
Payload	62.30	62.30
Launch Mass	1,203.57	371.01

After resolving the mass budget discrepancy, further research went into each subsystem of the rover to find a total calculated mass of 371.01 kg.

7.4 Structures

The Rover consists of a base and 2 suspension arrangements called rocker-bogie system. Rocker-bogie system consists of 3 wheels on each side of the rover. The material used for the rover assembly is Aluminum Alloy 7075-T6 (SN). The rover structure weights about 163 kg and supports payload such as scientific instruments, rover collection arm, thermal systems, and electronic boxes.

7.5 Thermal

Working with the thermal analysis philosophy shared in section 5.6.1, thermal analysis of the rover suggests the following results: The rover has an allowable temperature range of -40°C to 50°C. The rover has a minimum natural temperature of -40°C and a maximum natural temperature of 35°C. The skin material used to achieve these results is aluminum. The rover will require no heating while housed within the LAV as it will be considered part of the LAV section. The rover will also require no active heating while operating independently on the surface of Titan as its expected temperatures range -40°C to 35°C due to the RTG mounted on the rover structure. These results prove satisfactory.

7.6 ACS

The rover does not have a standard attitude control system like that which has been talked about in much of this report. Instead, it is equipped with a navigation and control system or NCS. For a land-based vehicle, the yaw angle is changed by turning the vehicle on the surface and the roll and pitch angles are a function of the terrain underneath the vehicle and the yaw angle of the vehicle. The main job of the NCS is to ensure that the rover always maintains an upright orientation while avoiding hazardous terrain or obstacles such as rocks or liquid lakes.

The instruments on board the rover that will allow the NCS to function correctly are Navigation Cameras (Navcams) and Hazard Avoidance Cameras (Hazcams). Both the Navcams and Hazcams are utilized by the rover to autonomously make decisions on where to go as it completes its mission on the surface of Titan.

7.7 Power

Working with the power analysis philosophy shared in section 5.8.1, the rover requires 390 W and utilizes a single 500 W Next Gen RTG stored on the rear. A visualization of the layout can be seen below in Figure 7.7-1.

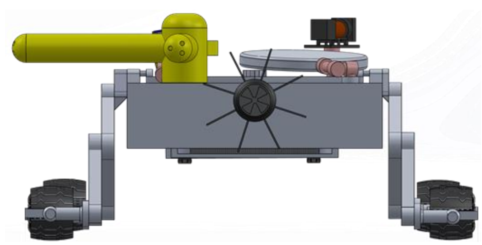

Figure 7.7-1 Rover RTG

7.8 Communications

For the communication system, Brown's "*Elements of Spacecraft Design*" was used for the communications and C&DH design. To begin, all sensors and equipment that would be needed for state of health of the spacecraft and scientific equipment were obtained and each of their data rates were determined. The rover will be capable of sending data to Earth via the LAV without the orbiter as an emergency scenario if communications are not possible, however more power will be needed from the rover in these scenarios. With the given data rates shown in **Table 7.8-1**, a transmitter power of 1 Watt, antenna diameter of 0.25 meter, and a S-band frequency of 2.2 GHz was determined to be the best combination to fit the design needs of the communication system.

Reed-Solomon encoding was used for the design to achieve the symbol rate shown in **Table 7.8-1** **[1]**.

Table 7.8-1 Rover Max Range Instrument Data Rates

Rover Max Range Instrument Data Rates		
TEDA	33333	bps
SHERLOC	992.45	bps
State of health	32500	bps
Nav Cameras (2)	600	bps
Haz. Cameras (2)	600	bps
Surface Camera	1667	bps
Max Data Consumption	69692.45	bps
Symbol Rate	79.6931603	ksps
Cable Loss	-0.06	dB
Antenna Diameter	0.25	m
Frequency (S-Band)	2.2	GHz
Antenna Gain	12.6072538	dB
EIRP	42.5472538	dB
Free Space Path Loss	-165.309054	dB
Atmospheric Attenuation	-0.15	dB
Estimated Polarization Loss	-0.2	dB
Rover Receiver Gain	3	dB
Pointing Loss	-0.13870469	dB
Beam Width	0.930133	degree
Receiver Cable Loss	0	dB
Total Received Power	-120.250504	dB
Receiver System Noise	18	K

Continuing with the Max Range Carrier Link Performance shown in **Table 7.8-2**, the values shown were determined using **Table 7.8-1** and Brown's *Elements of Spacecraft Design* as well **[1]**. The carrier link margin was determined to be suitable for even the worst-case scenario of the spacecraft.

Table 7.8-2 Max Range Carrier Link Performance

Max Range Carrier Link Performance		
System Noise Density	-216.047275	dB/Hz
Modulation Index	68	degree
Carrier Power/Total Power	-8.52849166	dB
Received Carrier Power	-128.778996	dB
Carrier Noise Bandwidth	14.77	dB-Hz
Carrier Signal to Noise	72.4982789	dB
Carrier Signal to Noise Req.	10	dB
Carrier Link Margin	62.4982789	dB

Lastly, for the Max Range Link Performance seen in **Table 7.8-3**, the required performance ratio of 4.5 was achieved giving assurance that communications will be performed at all stages of the mission using the maximum data received by the orbiter and rover with the capability to communicate directly with Earth.

Table 7.8-3 Max Range Link Performance

Max Range Link Performance		
Data Power/Total Power	-0.65668279	dB
Data Power Received	-120.907187	dB
Data Symbol Rate	-49.0142105	dB-Hz
Eb/No	46.1258772	
Eb/No Required	4.5	
Margin	41.6258772	

Shown below in **Table 7.8-4** is a list of the telecommunication hardware used by the rover during non-emergency situations. The components denoted with 0 W of power will be used during emergency situations.

Table 7.8-4 Rover Telecommunication Hardware (Non-Emergency Comms)

Rover Telecomm Hardware (Non-Emergency)		
Component	Mass (kg)	Power (W)
Parabolic Dish	-0.88	30
Choked horn LGA (2)	4.2	0
Deep Space Transponder	3.2	0

X-Band SSPA (1)	1.37	0
S-Band SSPA (1)	1.98	25
S-Band Transponder	2.95	20
Coax Cable	0.5	0
Diplexer	0.5	0
Attenuator	0.1	0
Total:	13.92	75

7.9 C&DH

The rover will be implementing two RAD 5545 radiation hardened multi-core processors manufactured by BAE Systems to manage the data and commands for the mission. The two will be wired in parallel for redundancy in case of a failure. As previously mentioned, these processors are radiation hardened and are certified for flight. Along with these processors, the rover will utilize a 1 TB solid-state recorder for data storage. **Table 7.9-1** shown below summarizes the mass and power of each component used by the rover's command and data handling system.

Table 7.9-1 Rover C&DH Hardware

Rover C&DH Hardware		
Component	Mass (kg)	Power (W)
RAD5545 Processor (2)	4	35
1 TB Storage	8	22
Total:	12	57

8 Sample Collection System

8.1 Derived Requirements

Table 8.1-1 includes the derived requirements developed to ensure the sample collection design satisfied mission requirements and needs.

Table 8.1-1 Sample Collection Derived Requirements

Req #	Requirement
T2.1.1-1	Sample storage and in-situ measurement system shall be operational in Titan surface temperature conditions
T2.1-2	The liquid sample storage system should be capable of simulating the environment on the surface of Titan from the time of sample collection to Earth return
M1.1-4	The probability that a planetary body will be contaminated by the spacecraft should be no more than 10^{-3} for a minimum of 50 years
M1.1-5	The probability of impact on Titan by any part of the launch vehicle should be ≤ 1x10-4 for a time period of 50 years after launch

8.2 Surface ConOps

The rover will deploy its rover arm and grab one of the test tubes from within the sample collection box. Once the sample collection rover is deployed from the LAV, it will then make its way towards the targeted lake to collect the liquid samples. At the scene of the lake, the rover will fill the test tube with the liquid and then place the test tube back into the collection box and repeat the process with 5 more test tubes. Once all the tubes have been safely deposited into the sample collection box, the rover will then make its way back to the LAV to deposit the samples. Once the rover reaches the LAV, the rover will then deposit the sample collection box onto the scissor lift, which will then lift the sample collection box into the sample capsule. The sample capsule, which is docked onto the ascent portion of the LAV that will make its return to Earth to complete the mission.

8.3 Sample Analysis and Protection

During ground operations, the liquid sample will be analyzed by the SHERLOC Context Imager to determine the existence of Vinyl Cyanide as well as determining if any metabolic processes are taking place. One sample retrieved from the lake will be heated and re-analyzed to predict decomposition of the sample during the return phase of the mission. Once this is complete, the liquid samples will be stored on the Rover and delivered to the LAV for the return phase of the mission.

During the return phase of the mission, surface conditions of Titan will be maintained for the samples in an effort to preserve the samples for further analysis on Earth. Sample composition will be measured over time with the Rover's SHERLOC camera to further determine if any metabolic processes are taking place within the sample, as well as to determine and monitor decomposition of the sample. The samples will be sealed withing their own chamber to minimize the chance of the sample being harmed by the space environment as well as to minimize contamination of Earth's environment upon arrival. A methane sensor will be in place to alert the Earth's ground team if there is any leaking from the sample into the storage chamber, which will abort Earth return in accordance with COSPAR's Planetary Protection Protocols **[2]**.

9 Systems Engineering

To ensure the design remained in compliance with all requirements and met all of the needs of the mission, the team used a systems engineering approach. Once the RFP was fully understood, the team moved on to defining the objectives, requirements, and constraints of the mission. Then, derived requirements were developed that added traceability from design decisions to the system level requirements. Section 9.1 contains a full list of all requirements of the system and a compliance record. Derived requirements are mostly phrased using “should” rather than “shall” statements to signify their prevalence as design guidelines that flowed down.

Throughout design, the team used trade matrices, as can be seen throughout the report, to balance the requirements and design optimization efforts like mass and cost reduction. In addition, a compliance matrix was maintained for all requirements and updated concurrently with design changes and updates.

The systems engineering approach also involved thinking about the system beyond meeting the explicit goals and objectives of the RFP. System life cycle, manufacturing, maintenance, and end of mission disposal were included in design considerations and mission design. Once the first iteration of design was completed meeting the requirements, the team returned to the design to refine while lowering mass and power demands where possible which also allowed for better budget management. Thus, the design not only meets, but surpasses the original requirements.

9.1 System Life Cycle

The system’s life cycle is captured in Figure 9.1-1, from the beginning of conceptual design through disposal. Preliminary design concluding in May 2022 will lead into detail design through 2024. Research and development on low TRL technologies, manufacturing, testing, and integration will occur from 2024 to March 2026. Then, the system will launch from Earth on March 30, 2026, complete the operations covered in Section 4.1, and return to Earth on October 3rd, 2045. Disposal will conclude when the returned sample has been safely accepted and the spacecraft has entered its heliocentric orbit.

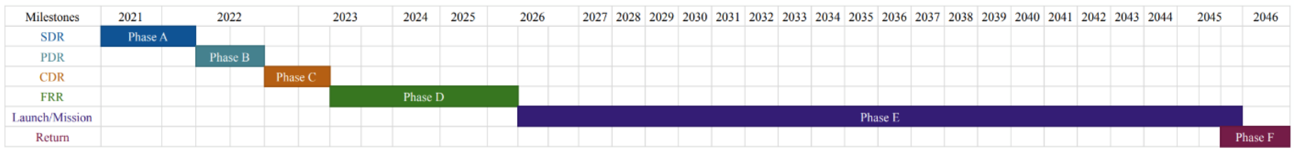

Figure 9.1-1 Mission Lifetime Gantt Chart

9.2 Manufacturing, Integration, and Test

Due to the unique and singular nature of this mission, as well as budget constraints, the team prioritized COTS components which would reduce cost and allow time and budget for technology maturation where needed. Manufacturers were chosen based on past success with similar missions. All manufacturing, integrating, and testing will be conducted in accordance with the COSPAR category V guidelines **[2]**.

Table 9.2-1 includes the manufacturers selected for each sensor and instrument used to accomplish the various science objectives.

Table 9.2-1 Science Instrument Manufacturers

Instrument or Sensor	Company
Surface Camera (Mastcam-Z)	Malin Space Science Systems
SHERLOC Context Imager	NASA JPL
Nav Cams	NASA JPL
Haz Cams	NASA JPL
Sample Container	Emrys Space Systems
Star Tracker	Ball Aerospace
IMU	Advanced Navigation

Base Vision System Camera (BCAM)	NASA JPL
CIRS	NASA Goddard
UVIS	LASP

The orbiter integration will be led by LMSS, which has demonstrated strong capabilities with long term projects like Juno, Lucy, and MAVEN. Since most of the components of the spacecraft are COTS, testing costs will be more heavily focused on the integrated systems.

NASA JPL will be responsible for the MI&T of the rover and LAV. Following a strong history of missions like Perseverance and Curiosity, JPL has the experience to approach the new environment on Titan reliably.

Lockheed Martin Space Systems will conduct the final integration and test campaigns for the spacecraft.

9.3 Maintenance

During the spacecraft's time on Earth, maintenance will be conducted regularly to ensure readiness for launch. Regular checks will be instituted after the KRUSTY and RTGS are installed to certify the functionality of all electronics. After launch and until return to Earth, onboard systems will be responsible for health checks and system maintenance, which will be frequently sent back to Earth.

9.4 Disposal and End of Mission

Once it returns and sends the sample to the surface, the return vehicle will conduct final data dumps to Earth and maneuver into a heliocentric orbit. It will have enough propellant to enter this orbit. After this time, if there is public interest enough to gain funding and the mission is viable, the spacecraft may be retrieved using services from companies like Astroscale and safely returned to Earth for decontamination and historic display or recycling.

In order to remain in compliance with COSPAR planetary protection guidelines, contamination probabilities will be assessed and reduced as much as possible. These efforts will allow the spacecraft to return to Earth safely. These requirements appear in the derived requirements that can be found in Appendix A.

9.5 Risk Analysis

Risks to the team's design were determined from design through system disposal. Due to the scope of this project, risks were limited to a certain level of detail, but thoroughness was still a priority of the assessment. Each risk was classified on the basis of probability of occurrence and severity of the risk's consequences, as shown in **Table 9.5-1** Then, mitigations were determined for the risks that brought them into an acceptable level and the risks were reevaluated. **Table 9.5-2** shows the risk likelihood-consequence matrix.

Table 9.5-1 Risk Classification Legend

Probability		Consequence	
A	Near Certainty	**1**	Catastrophic
B	Highly Likely	**2**	Critical
C	Likely	**3**	Significant
D	Low Likelihood	**4**	Moderate
E	Not Likely	**5**	Minimal

Table 9.5-2 Risk Cube

Probability	A5	A4	A3	A2	A1
	B5	B4	B3	B2	B1
	C5	C4	C3	C2	C1
	D5	D4	D3	D2	D1
	E5	E4	E3	E2	E1
	Consequence				

9.6 Risk Statements and Mitigations

Table 9.6-1 shows the risks and their corresponding mitigations, along with their before and after classifications, color-coded as per the risk cube.

Table 9.6-1 Risk Statements and Mitigations

Risk and Mitigation	Class	Adj. Class
If the design is not completed by May 2022 due to the team's inexperience with the design process, then the team could fail their senior design class and not graduate.	C1	E5
The team created a comprehensive schedule for all phases of design and institute regular progress checks and reports.		
If the spacecraft experiences loss of communication due to damage to the communications hardware during launch, then the spacecraft will not be able to transmit information back to Earth and the vehicle may be unable to execute the mission.	D2	E4
The team will select reliable hardware and the spacecraft and its components will undergo comprehensive stress and vibration testing		
If the spacecraft experiences loss of communication due to loss of power to the communications system, then the spacecraft will not be able to transmit	D2	E5

information back to Earth and the vehicle may be unable to execute the mission.		
The spacecraft will include redundant on-board power management that functions autonomously.		
If the spacecraft loses power due to faulty hardware that does not distribute power properly, then the spacecraft will be unable to perform the mission.	D3	E4
The team will select reliable hardware from proven manufacturers as much as possible and perform functional testing for all systems.		
If the instruments cannot gather reliable data due to the spacecraft being unable to function in Titan's environment, then the team will not gather accurate in-situ measurements.	C2	D4
The instrument selection will prioritize low operating temperatures. In addition, thermal design will consider cryogenic conditions and the testing campaign will involve functional testing in simulated environments.		
If the sample collection mechanisms cannot collect and store samples due to the mechanisms being unable to function in Titan's environment, then the team will not gather and return the required sample	B1	C5
The mechanism will prioritize low operating temperatures. In addition, thermal design will consider cryogenic conditions and the testing campaign will involve functional testing in simulated environments.		
If there are computer errors on the spacecraft due to unexpected anomalies and errors, then there could be power distribution and management errors, propulsion errors, and instrument and mechanism errors.	D1	E3
The spacecraft computers will be armed with autonomous fault detection and correction software and the team will implement hardware with overlapping functions.		
If the sample conditions are not controlled due to improper sample storage, then the spacecraft could be contaminated and would be unable to return to Earth.	A1	E1
The sample storage system will undergo functional and environmental testing before launch.		
If the sample is contaminated before it is returned due to changes the sample experiences during the flight back to Earth, then any information gathered from the returned sample will be unreliable and the science planned back on Earth would be rendered invalid.	A1	E1
The sample collection system will take measurements of the samples collected. In addition, the sample will be exposed to potential environmental changes in-situ to determine what properties observed when the sample returns to Earth are due to transport and what are scientifically significant.		

If the spacecraft structure fails due to excessive stresses experienced during the mission lifetime, then the trajectory maneuvers, landing, or Titan departure could destroy the spacecraft.	D1	E3
The spacecraft will be designed to survive maximum loads from launches, maneuvers, and landings and the team will conduct structural and environmental testing.		
If the spacecraft is destroyed during launch due to the launch vehicle aborting during launch, then the spacecraft will explode and the mission will be lost.	C1	E4
The chosen launch vehicle reliability will be at least 85%		
If the spacecraft does not land in the correct orientation due to ACS and GNC system failures, then the rover will not be able to deploy to collect the sample.	B1	E3
The ACS system will be autonomous, implement redundant sensors and thrusters, and undergo rigorous functional testing		
If the spacecraft crashes into the surface of Titan due to an excessive rate of descent, then the spacecraft structure could be compromised.	B1	E2
Propulsive and passive landing techniques will be implemented and undergo functional testing.		
If the KRUSTY nuclear reactor is improperly contained onboard the spacecraft due to poor technology development, then the spacecraft electronics may malfunction or the sample collected could be rendered unusable.	B1	D4
Radiation-hardened hardware will be used on the spacecraft and testing will be implemented on the KRUSTY containment. The sample will be stored in a radiation-safe container.		
If the chosen Launch Vehicle does is not equipped for a nuclear powered payload due to the rarity of the use of this technology, then the launch schedule could be pushed back and the program would incur extra costs	A1	C5
The chosen launch vehicle will either already be equipped to handle the KRUSTY reactor or will be retrofitted to accommodate it.		

10 Cost Analysis

One of the requirements from the Titan sample return request for proposal is a total mission cost not exceed $5 billion. Initially, the NASA Project Cost Estimating Capability (PCEC) was used to estimate the cost of the initial design. To get an accurate cost estimation, a multielement robotic spacecraft was used as the template when setting up the work breakdown structure (WBS) for the cost estimation. In this WBS, the three main sections of the spacecraft, the orbiter, rover, and LAV were used as subsections with each of their related subsystems listed underneath them. First using just the mass and power budgets the software ran with and produced an estimated cost of around $2.02 billion with a mass of 4,000 kg. As the team continued researching each subsystem and deciding on specific parts and materials the mass and power budgets became more accurate to the launch mass. One of the major costs came from the launch vehicle that the team selected which was the SLS, which has a cost of $2 billion. Using the updated masses and power the software estimated a mission cost of around $4.23 billion with a mass of 10,000 kg. A final cost estimation was done using the PCEC which resulted in a cost of $4.27 billion. The PCEC mass estimate also includes the breakdowns for each vehicle and their subsystems' estimated cost. These breakdowns can be seen in

Table *10-1*. The LAV with a total cost of $235.5 million, the orbiter with a cost of $152.5 million and the rover with a cost of $236.9 million.

Table 10-1 NASA PCEC Breakdown per vehicle

	LAV	Orbiter	Rover
Structures/Mechanisms	$24.2M	$17.6M	$15.7M
Thermal	$13.4M	$7.5M	$26.5M
Power	$33.5M	$33.9M	$24.3M
GN&C	$25.0M	$16.2M	$22.6M
Propulsion	$9.0M	$6.3M	$19.9M
Comms	$78.5M	$57.5M	$64.4M

C&DH	$38.4M	$13.4M	$21.4M
Entry, Descent & Landing	$13.6M	-	-
Total Cost	$235.5M	$152.5M	$236.9M

After running the PCEC another method of cost estimation was used to gauge the accuracy of the previous estimations. The second cost estimation method used was the USCM8 cost estimation tool for the New SMAD textbook. The comparison of the two cost estimations can be seen below in **Table 10-2**. The second cost estimation method produced a cost of $4.49 billion.

Table 10-2 Cost Breakdown Table with NASA PCEC and New SMAD USCM8

Subsystem	PCEC	USCM8
Orbiter	$152.5M	$509.2M
LAV	$235.8M	$497.6M
Rover	$236.9M	$299.7M
Launch Vehicle	$2.00 B	$2.00 B
Project Management	$189.2M	$110.8M
Systems Engineering	$344.2M	$194.4M
Science/Technology	$75.9M	$257.5M
Flight Systems	$900.1M	$47.9M
Mission Operations	$368.9M	$478.7M
System Integration/Testing	$205.2M	$91.5M
Total Cost	$4.27B	$4.49B

11 Compliance

By going through the RFP and design process, the team developed lists of requirements, constraints, and deliverables and tracked compliance throughout the process.

11.1 Design Compliance

Table 11.1-1 shows the compliance of the design with requirements and constraints with reference to the sections of the report that discuss those design aspects.

Table 11.1-1 Design Requirements and Compliance

Req #	Statement	Derived from	Comply	Section(s)
SYS 1.0	Spacecraft shall return to Earth a sample of at least 500 mL of surface liquid from Titan	RFP	Yes	4.1, 4.2, 7.2, 8.2
SYS 2.0	Launch vehicle shall launch by 2030	RFP	Yes	4.1, 9.1
SYS 3.0	Design shall address all required propulsive maneuvers and corrections to the orbit	RFP	Yes	4.1, 4.4, 4.5, 5.4, 6.3
SYS 4.0	In-situ measurements from Titan shall be returned to Earth by 2045	RFP	Yes	4.1, 4.4, 7.2, 8.2, 9.1
SYS 5.0	The sample and all science data shall be returned to Earth no later than December 2045	RFP	Yes	4.1, 4.4, 7.2, 8.2, 9.1
SYS 6.0	Total mission cost shall not exceed $5B (FY21)	RFP	Yes	10
M1.1-1	A Gantt chart shall be created and all major milestones shall be scheduled	SYS 6.0	Yes	9.1
M1.1-2	Conceptual and preliminary design phases shall be completed by the end of May 2022	SYS 6.0	Yes	9.1
M1.1-3	Spacecraft shall launch no earlier than 2026	SYS 6.0	Yes	4.1, 9.1
T2.0-1	Spacecraft should fit within chosen launch vehicle payload for launch	SYS 2.0	Yes	4.2, 4.5, 5.5, 6.4, 7.4
T2.2-1	The spacecraft components should experience accelerations less than 30 Gs	SYS 2.0	Yes	4.5, 5.5, 6.4, 7.4

T2.5-1	Spacecraft computers should include redundant or overlapping hardware and autonomous error detection and correction software	SYS 2.0	Yes	5.2, 5.6-5.10, 6.5-6.9, 7.5-7.9, 9.5
T2.1.1-1	Sample storage and in-situ measurement systems shall be operational in Titan surface environmental conditions	SYS 4.0	Yes	5.6, 6.5, 7.5, 8.3
T2.1-1	The liquid sample storage system should be capable of simulating the environment on the surface of Titan.	SYS 5.0	Yes	7.5, 8.3
T2.1-2	The rover shall be capable of traveling to the sample collection site from the landing site	SYS 1.0	Yes	7.6, 7.7
T2.4.3-1	Spacecraft ACS system should have 3-axis control	SYS 3.0	Yes	5.7, 6.6, 7.6
T2.4.3-2	Spacecraft ACS system should allow for 180 degree maneuverability	SYS 3.0	Yes	5.7, 6.6, 7.6
T2.3-1	ACS system should incorporate redundant sensors and thrusters	SYS 3.0	Yes	5.7, 6.6, 7.6
M3.1-1	The selected launch vehicle should have a flight success rate greater than 85%	SYS 2.0	Yes	4.5
M1.1-4	The probability that a planetary body will be contaminated by the spacecraft should be no more than 10^-3 for a minimum of 50 years	SYS 1.0	Yes	2.4, 8.3, 9.4
M1.1-5	The probability of impact on Titan by any part of the launch vehicle should be ≤ 1x10-4 for a time period of 50 years after launch	SYS 1.0	Yes	2.4, 8.3, 9.4
T2.1-3	The orbiter should prioritize autonomous functions	SYS 1.0	Yes	4.1, 4.4
T2.1-4	The LAV shall be an autonomous vehicle	SYS 1.0	Yes	6.6, 6.8, 6.9
T2.1-5	The rover shall be an autonomous vehicle	SYS 1.0	Yes	7.6, 7.8, 7.9

11.2 Deliverables Compliance

Table 11.2-1 shows the compliance of the report with deliverables listed within the RFP alongside reference to the relevant sections of the report.

Table 11.2-1 Deliverables Compliance

RFP Para	Statement	Comply	Section(s)
5.01	Payload configurations	Yes	4.2
5.01	Spacecraft configurations	Yes	4.2
5.01	Mission design	Yes	3
5.02	Launch window in available/projected launch vehicle	Yes	4.5
5.03	Specify extreme operating conditions in environments (temp, radiation exposure, etc)	Yes	5.6, 6.5, 7.5
5.03	Specify systems for command, control, communication, and data storage and transmission	Yes	5.7, 5.9, 5.10, 6.5, 6.6, 6.8, 6.9, 7.5, 7.6, 7.8, 7.9
5.03	Specify requirements for power system, TPS, tracking and positioning systems, any unique systems, etc	Yes	5.1, 6.1, 7.1, 8.1, 11.1
5.03	Indicate needs for Planetary Protection Protocols and their influence on design	Yes	2.1, 2.4, 9.2
5.04	3-view layouts	Yes	4.2
5.04	Configurations and placements of all systems	Yes	4.2
5.04	Masses for structures, piping, and cabling	Yes	5.3, 6.2, 7.3
5.04	Tank capacity	Yes	5.4.2, 6.3
5.04	Total dry and wet masses where applicable	Yes	5.3, 5.10, 6.2, 7.3
5.04	Mass and power statements	Yes	5.2, 5.3, 5.8.5, 5.10, 6.2, 6.8, 6.9, 7.2, 7.3, 7.7, 7.8, 7.9
5.04	Power requirements for all subsystems	Yes	5.8.1, 6.8, 6.9, 7.7, 7.8, 7.9
5.05	Cost analysis from initial design, fabrication, testing, integration, launch, launch ops, in space ops, flight, data return, disposal/decommissioning	Yes	10, Appx B
5.06	Cost analysis for hardware, flight and ground systems software, PPP, data archiving	Yes	10, Appx B

5.06	Cost estimation for interplanetary cruise ops and science ops	Yes	10, Appx B
5.06	Cost estimation for science acquisition add-ons	Yes	10, Appx B
5.07	Goals and objectives of the mission	Yes	2.2
5.08	Mission, system, and subsystem level requirements for science, mission, and design	Yes	2.3, 2.4, 5.1, 6.1, 7.1, 8.1, 11.1
5.09	Trade studies for mission architecture and mission operations	Yes	3.4
5.10	Details concept of operations for all mission phases, including execution and accomplishment of science requirements	Yes	4.1, 8.2, 8.3
5.11	Discussion of structures, mechanism, thermal, ACS, telemetry, tracking, command, power, propulsion, scientific payload, and sensor design	Yes	2.1, 2.2, 4.1, 4.2, 4.2.4, 5.2, 5.5-5.8, 6.5-6.9, 7.2, 7.5-7.7, 7.9, 9.2
5.11	Discussion of LV interfacing	Yes	4.5
5.11	Discussion of ConOps design	Yes	4.1, 8.2, 8.3
5.11	Mass and power budgets, broken down by subsystem and including margins	Yes	5.2, 5.3, 5.8.5, 5.10, 6.2, 6.8, 6.9, 7.2, 7.3, 7.7, 7.8, 7.9
5.11	Proposed ground system for operations	Yes	3.2, 4.1, 4.2.4, 6, 7, 8
5.11	Mass, power, and other resource summary table	Yes	5.2, 5.3, 5.8.5, 5.10, 6.2, 6.8, 6.9, 7.2, 7.3, 7.7, 7.8, 7.9
5.12	Top-level cost estimate covering life cycle for all cost elements	Yes	10, Appx B
5.12	WBS capturing each cost element	Yes	Appx B
5.12	Cost summary table for all WBS elements	Yes	Appx B
5.13	Mission development and ops schedule	Yes	4.1, 9.1
5.13	Schedule margin applied to appropriate areas	Yes	9.1
5.13	Funded schedule reserve included in cost estimate	Yes	Appx B

12 Conclusion

Emrys Space Systems has developed and refined a design solution to the request for proposal from the Caltech Space Challenge 2022. Initially developing two designs to return a sample from the surface of Titan by December 2045. Each design showed a unique solution with in-situ propellant collection and use of orbital refueling respectively. Leveraging a trade study which weighed technological maturity and cost, the design using orbital refueling was selected as at the final architecture. The design was then refined with a significant mass savings and increased fidelity in the analysis of various subsystems.

The winning architecture features three major vehicles, an orbiter, landing and ascent vehicle, and an autonomous rover. The orbiter utilizes the KRUSTY nuclear reactor to power two NEXT ion propulsion engines which will propel the vehicle from Earth to Titan, where it will conduct an orbiting science mission with the UVIS and VIMS instruments. Upon arrival to Titan the LAV will descent to Titan and deploy the autonomous rover. Over the span of the next month, the rover will carry out its mission to collect in-situ measurements and the samples from our target lake of Bolsena Lacus. Once the samples are collected, the rover will return to the LAV and transfer the samples. After launching from Titan and a refueling stop with the orbiter, the LAV will return to Earth. Arriving at Earth, the sample capsule will be deployed and the remaining portions of the LAV will be placed in a heliocentric orbit to ensure compliance with planetary protection protocols.

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Appendix A: Science Traceability Matrix

Titan Sample Return Caltech Space Challenge 2022	**Objective 1:** Retrieve samples from liquid lakes on Titan to determine if methane-based life forms exist **Objective 2:** Analyze atmospheric conditions of Titan and preliminary sample analysis to find conditions viable for biological life
Mission Objectives	Find habitable conditions for life outside of Earth by retrieving data from the suspected liquid bodies of methane located on one of Saturn's moons Titan, along with taking atmospheric sample readings of Titan

Science Objectives	**Measurement Objectives**	**Measurement Requirements**	**Instruments**	**Instrument Requirements**	**Data Products**
Environmental Analysis					
Monitor wind speeds at ground level during the mission	While on the surface of Titan, measure the intensity and direction of the wind to discover more information on Titan's conditions	Withstand wind speeds of 400 km/hr	Anemometer (TEDA)	Able to change orientation	Digital transmission in km/hr
Measure atmospheric	To ensure that the conditions of the sample are maintained during	Withstand pressure of 1.5 atm	Barometer	Thermal resistance Low	

pressure at sample retrieval site	the retrieval and containment for return flight		(TEDA)	power consumption	Digital transmission in atm
Temperature and its variation during the duration of the mission	Gain insight into the environmental cycle on Titan	Withstand temperature of 80 K	Thermometer (TEDA)	Low power consumption	Digital transmission in Kelvin
Environmental mapping during Lander descent to Titan surface	Gain insight into the environment around the lander for insight into Titan environment in addition to spotting potential hazards	Camera must be able to see through Methane clouds to ensure visibility	Altimeter, Gyro, Comms, Camera	High rate of comms, High camera resolution	Digital transmission, Camera footage
Atmosphere chemical composition at surface level	Precise Hydrogen, Acetylene, Methane, Ethane and Carbon concentrations for the duration of the mission	Sensor accuracy within <1%	Gas Chromatograph Mass Spectrometer	Thermal resistance Sealed instruments	Digital transmission of atmosphere composition
Sample Analysis					
Retrieve sample from liquid bodies located on Titan	Retrieve a minimum of 500 mL of liquid sample from Kraken Mare	Maintain pressure and temperature of sample based on measured environmental data	Mechanical retrieval	Thermal resistance Sealed container	Physical sample to be further analyzed
Measure temperature of	To ensure that the conditions of the sample are maintained during	Withstand temperature of 80 K	Thermometer	Low power consumption	

liquid sample at Titan	the retrieval and containment for return flight				Digital transmission in Kelvin
Analyze liquid sample for signs of biological molecules	Perform analysis of liquid sample from Titan to find Vinyl Cyanide	Sensor accuracy within <1%	SHERLOC Camera	Low power consumption	Digital transmission of results
Analyze liquid sample for hydrocarbons obtained on Titan surface	Measure the amount of Acetylene, Methane and Ethane within the liquid methane sample	Sensor accuracy within <1%	PIXL Camera	Sealed instruments Low power consumption	Digital transmission of results
Mission Documentation					
Obtain photos of the landing site upon entry	To gain insight into the geography of the landing sight and the overall impact of the system on the environment	Resolution of 1024x1024	Camera	360-degree view Thermally Resistant	Digital transmission of photos
Obtain video of Titan surface during periods of exploration	Gain additional insight into the geography of Titan in addition to verifying sample mission success	Resolution of 20 Mp	Video camera	Less than 1 kg Thermally resistant 360-degree view	Physical video hard-drive retrieved from Rover

Appendix B: Cost Breakdown WBS

WBS #	Level	Line Item Name/Description	Non-Recurring	Recurring Production	Non-Allocated	Operations	TOTAL
0	**1**	**System Name**	**$2,624.60**	**$1,367.80**	**$75.90**	**$200.00**	**$4,268.30**
1	2	Project Management	**$75.10**	**$114.20**	**$ -**	**$ -**	**$189.20**
2	2	Systems Engineering	**$104.00**	**$240.20**	**$ -**	**$ -**	**$344.20**
3	2	Safety and Mission Assurance	**$32.80**	**$57.00**	**$ -**	**$ -**	**$89.80**
4	2	Science/Technology	**$ -**	**$ -**	**$75.90**	**$ -**	**$75.90**
5	2	Payload(s)	**$35.80**	**$59.20**	**$ -**	**$ -**	**$95.00**
5.01	3	Payload Management	**$9.80**	**$15.00**	**$ -**	**$ -**	**$24.80**
5.02	3	Payload System Engineering	**$7.80**	**$18.10**	**$ -**	**$ -**	**$25.90**
5.03	3	Payload Product Assurance	**$0.70**	**$1.30**	**$ -**	**$ -**	**$2.00**
5.1	3	Instruments - EMPTY ROLLUP	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
5.x	3	Payload I&T	**$17.40**	**$24.90**	**$ -**	**$ -**	**$42.30**
6	2	Flight System \ Spacecraft	**$339.40**	**$560.70**	**$ -**	**$ -**	**$900.10**
6.01	3	Flight System Project Management	**$3.90**	**$5.90**	**$ -**	**$ -**	**$9.80**
6.02	3	Flight System Systems Engineering	**$17.60**	**$40.60**	**$ -**	**$ -**	**$58.20**
6.03	3	Flight System Product Assurance	**$9.70**	**$16.80**	**$ -**	**$ -**	**$26.50**
6.1	3	Orbiter	**$49.50**	**$103.00**	**$ -**	**$ -**	**$152.50**
--	4	Elt1 - Management	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt1 - Systems Engineering	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt1 - Product Assurance	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt1 - Structures & Mechanisms	**$6.60**	**$11.10**	**$ -**	**$ -**	**$17.60**
--	4	Elt1 - Thermal Control	**$2.10**	**$5.40**	**$ -**	**$ -**	**$7.50**
--	4	Elt1 - Electrical Power & Distribution	**$11.30**	**$22.50**	**$ -**	**$ -**	**$33.90**
--	4	Elt1 - GN&C	**$5.90**	**$10.30**	**$ -**	**$ -**	**$16.20**
--	4	Elt1 - Propulsion	**$2.70**	**$3.60**	**$ -**	**$ -**	**$6.30**

--	4	Elt1 - Communications	**$17.00**	**$40.50**	**$ -**	**$ -**	**$57.50**
--	4	Elt1 - C&DH	**$3.80**	**$9.60**	**$ -**	**$ -**	**$13.40**
--	4	Elt1 - I&T	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
6.2	3	LAV	**$83.20**	**$152.70**	**$ -**	**$ -**	**$235.80**
--	4	Elt2 - Management	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt2 - Systems Engineering	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt2 - Product Assurance	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt2 - Structures & Mechanisms	**$5.20**	**$19.20**	**$ -**	**$ -**	**$24.40**
--	4	Elt2 - Thermal Control	**$2.10**	**$11.40**	**$ -**	**$ -**	**$13.40**
--	4	Elt2 - Electrical Power & Distribution	**$13.50**	**$20.00**	**$ -**	**$ -**	**$33.50**
--	4	Elt2 - GN&C	**$6.90**	**$18.10**	**$ -**	**$ -**	**$25.00**
--	4	Elt2 - Propulsion	**$4.30**	**$4.80**	**$ -**	**$ -**	**$9.00**
--	4	Elt2 - Communications	**$23.30**	**$55.20**	**$ -**	**$ -**	**$78.50**
--	4	Elt2 - C&DH	**$25.90**	**$12.50**	**$ -**	**$ -**	**$38.40**
--	4	Elt2 - Entry, Descent, and Landing	**$2.00**	**$11.50**	**$ -**	**$ -**	**$13.60**
--	5	Elt2 - Parachutes	**$0.40**	**$4.00**	**$ -**	**$ -**	**$4.40**
--	5	Elt2 - Thermal Protection System	**$1.70**	**$7.50**	**$ -**	**$ -**	**$9.20**
--	4	Elt2 - I&T	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
6.3	3	Rover	**$101.50**	**$135.40**	**$ -**	**$ -**	**$236.90**
--	4	Rover	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt3 - Systems Engineering	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt3 - Product Assurance	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt3 - Structures & Mechanisms	**$4.20**	**$11.50**	**$ -**	**$ -**	**$15.70**
--	4	Elt3 - Thermal Control	**$3.00**	**$23.60**	**$ -**	**$ -**	**$26.50**
--	4	Elt3 - Electrical Power & Distribution	**$11.30**	**$12.90**	**$ -**	**$ -**	**$24.30**
--	4	Elt3 - GN&C	**$6.30**	**$16.30**	**$ -**	**$ -**	**$22.60**
--	4	Elt3 - Propulsion	**$2.80**	**$16.50**	**$ -**	**$ -**	**$19.30**

--	4	Elt3 - Communications	**$19.40**	**$45.00**	**$ -**	**$ -**	**$64.40**
--	4	Elt3 - C&DH	**$11.80**	**$9.60**	**$ -**	**$ -**	**$21.40**
--	4	Communications (SSPA), Planetary		**$ -**	**$ -**	**$ -**	**$ -**
--	4	Elt3 - I&T	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
6.x	3	Flight System I&T		**$106.30**	**$ -**	**$ -**	**$106.30**
7	2	Mission Operations System (MOS)	**$37.60**	**$131.30**	**$ -**	**$200.00**	**$368.90**
--	3	MOS/GDS Development (Phase B-D)	**$37.60**	**$131.30**	**$ -**	**$ -**	**$168.90**
--	3	Mission Ops & Data Analysis (Phase E)	**$ -**	**$ -**	**$ -**	**$200.00**	**$200.00**
8	2	Launch Vehicle/Services	**$2,000.00**	**$ -**		**$ -**	**$2,000.00**
9	2	Ground Data System (GDS)	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
10	2	System Integration, Assembly, Test & Check Out		**$205.20**	**$ -**	**$ -**	**$205.20**
11	2	Education & Public Outreach	**$ -**	**$ -**	**$ -**	**$ -**	**$ -**
					Total w/ Fees, Burdens, & Reserves		**$4,268.32**