Entry, Descent, & Landing Architecture for Unmanned Undersea Vehicle Mission to Titan

by

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The lakes of Titan are known to be rich in hydrocarbons, though their exact composition is unknown. These hydrocarbons could be conducive to some form of abiogenesis or the sustaining of simple alien life similar to the kind found in deep-sea thermal vents. The proposed "Hyperion" mission will consist of a submarine that will splash down in Kraken Mare, explore the depths of the lake, and transmit science data back to Earth. The UUV will take and analyze samples using a mass spectrometer and sampling system to elucidate the exact composition and rate of change of the composition of Kraken Mare. Depth measurements of the lake will be made using sonar, and temperature and pressure measurements will be taken and recorded at varying depths. A camera suite will be used to capture images of the sea floor, as well as any microscopic samples of interest. All collected data will be transmitted directly to Earth.

1.

Table 1: Level 1 Science and Mission Requirements
The payload shall safely and successfully splash down in Titan's polar liquid methane lake, Kraken Mare
The vehicle shall be capable of moving through the liquid methane lakes on the surface of Titan
The vehicle shall operate for two years
The vehicle shall take panoramic pictures of the lakebed
The vehicle shall investigate chemical composition and changes thereof under the surface of the liquid methane lakes through optical spectroscopy
The vehicle shall take temperature and pressure measurements during atmospheric entry and descent
The vehicle shall take microscopic pictures of collected samples
The vehicle shall take temperature and pressure measurements throughout the lake
The vehicle shall transmit collected science data directly back to Earth

2.

Table 2: Level 2 Entry System Requirements
The entry and descent capsule shall not exceed 5 g's during entry and descent flight
The EDL vehicle shall collect telemetry data during EDL flight
The EDL vehicle shall land directly in Kraken Mare
The submarine shall not experience more than 5 g's during splashdown.
The entry vehicle shall withstand 85.889 W/cm^2 during entry
The entry vehicle shall withstand 9891.293 J/cm^2 during entry
The submarine shall splashdown into Kraken Mare at no more than 27.013 m/s
The submarine shall supply the entry vehicle with 187 W of power to support telemetry data collection and entry Attitude Control System (ACS)

3. EDL Architecture

Hyperion will enter Titan's atmosphere from an elliptical orbit around Saturn. The EDL phase of the mission will consist of three stages, shown in Figure 1. The first phase will be hypersonic ballistic descent from an altitude of 1400 km to 135 km during which the spacecraft will experience peak heating and peak deceleration. Once the atmosphere has slowed the spacecraft down to below mach 5, it will enter the supersonic descent phase during which the aftbody will separate. Once the spacecraft has slowed down to below mach 1, it will enter the subsonic parachute terminal descent phase, during which the heat shield will separate and the parachute will deploy. The submarine will splash down into Kraken Mare nose first. Each phase of the EDL is described in detail in the following sections. Figure 1 shows the different stages of the EDL process from entry to splashdown.

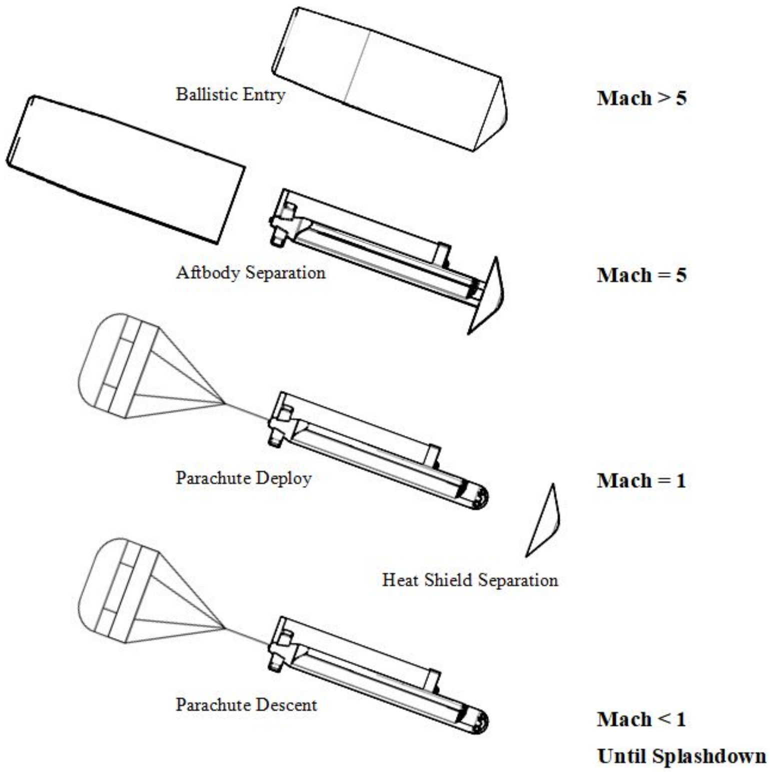

Figure 1: Hyperion EDL Process

4. Approach for Testing and Analyzing System Prior to Launch

In order to test the drag properties of our entry systems, we will be utilizing a hypersonic wind tunnel in combination with a scale model and pressurized air in order to match the Reynolds number and Mach number to what is expected on Titan. For thermal testing, we will be replicating the testing regiment outlined by Gentry Lee and his team in a report on Cassini/Huygens [1]. This would consist of placing thermocouples at various depths of the TPS and using an arcjet in order to test the temperature and recession rate at various heat fluxes, from 60 to 250 W/cm^2. A gaseous atmospheric composition as close to that of Titan would be used: 77% N_2, 20% Ar and 3% CH_4, by volume. Strain gauges and temperature sensors on various parts of the heat shield can measure the strain and thermal load on the entry vehicle. We will also push, budget allowing, for a splashdown test using a prototype launched from a sounding rocket. This would allow us to test the structural loads and shocks during splashdown, as well as allow us to test the deployment mechanism for our submarine. The submarine itself would need to undergo cryogenic testing and qualification as close to 93 Kelvin as possible. Finally, the attitude control system (ACS) will be tested using an air bearing to check for accurate attitude measurements, and with thrust test stands to understand the thrust response of each thruster to ensure accurate control actions.

5. Definition of Entry Conditions Including Entry Velocity and FPA

The most efficient way to reach Titan is to use gravity assist maneuvers. By flying close to a planet on a hyperbolic trajectory in the same direction as the planet's orbit, a spacecraft can extract some orbital energy from the planet, gaining velocity relative to the sun at no additional fuel cost. Cassini used two gravity assists around Venus and one around Earth to reach Jupiter, where it performed one more gravity assist around Jupiter to reach Saturn. Hyperion will likely follow a similar trajectory to Cassini [2].

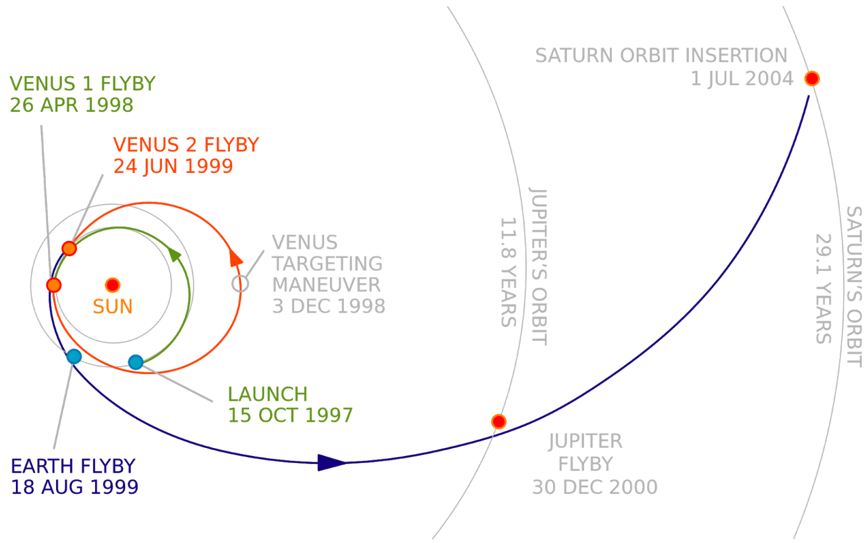

Figure 2: Cassini's Orbital Trajectory

After entering Saturn's sphere of influence, it will perform an orbital insertion maneuver to enter into an elliptical orbit around Saturn which intersects with Titan's orbit. The EDL stage will separate from the cruise stage and enter Titan's atmosphere at a speed of 6 km/s relative to Titan, with a flight path angle of 8.5 degrees. The entry velocity was chosen due to heritage from the Huygens probe, and the flight path angle was determined to be the maximum angle so that the maximum deceleration would not exceed 5 g's. Based upon the temperature data from Brown et al. [3], this mission was modelled at an entry altitude of 1,400km.

6. Entry Vehicle Subsystem Definitions and Sizing

a. Aeroshell Geometry Definition and Sizing

A 60 degree half-angle sphere-cone will be used as the forebody shape as was successfully flown in the Titan Huygens mission [2]. The coefficient of drag (Cd) as a function of mach number was incorporated into the calculations regarding each phase in the EDL. The Figure 3 shows data from Clark et al. [4] which was then extrapolated using a spline with 500 points to be accessed during the calculations.

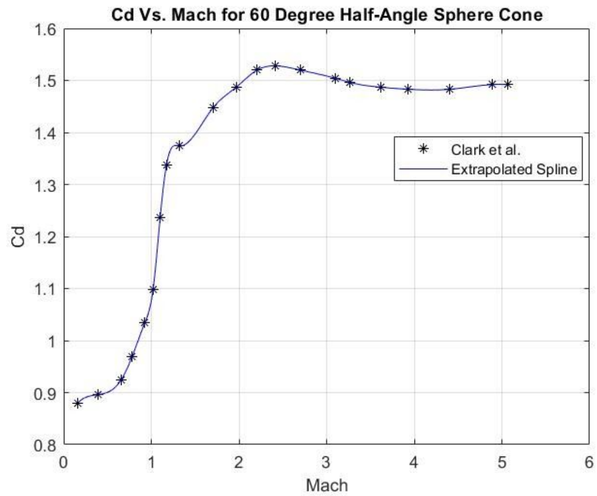

Figure 3: Drag Coefficient for 60 Degree Half-Angle Sphere-Cone

In the hypersonic regime, Cd was assumed to be a constant value equal to the Cd at mach 5. This assumption was made because the mission extends beyond the reported Cd data for 60 degree half-angle sphere cones.

The diameter of the aeroshell was driven by the packaging requirements of the submarine. The dimensions of the submarine were based on the design presented by Oleson et al. [5] because the geometry of their design accommodates the scientific instruments needed for the mission described in this paper. The submarine is capable of folding its antenna mast for stowage and is shown in both its fully operational configuration and stowed configuration in Figure 4.

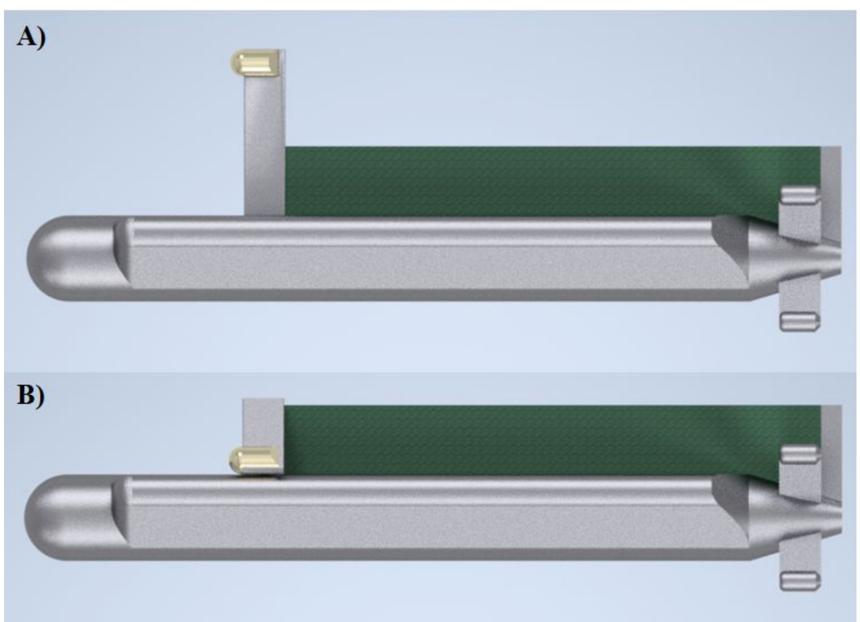

Figure 4: Hyperion Submarine. (A) Fully Operational Configuration. (B) Stowed Configuration.

Once in the stowed configuration, the maximum dimensions of the submarine fit within a 5.87m long, 1.75m diameter cylinder. The diameter of the aeroshell was chosen to be $d_{ev} = 2$m to add some packaging margin. The sphere-cone nose radius was defined as $R_n = \frac{d_{ev}}{4} = 0.5$m. The aftbody is a 5.5m long, 2m diameter chamfered cylinder made from aluminum. A cross-sectional view of the submarine in the stowed configuration encapsulated by the aeroshell is shown in Figure 5.

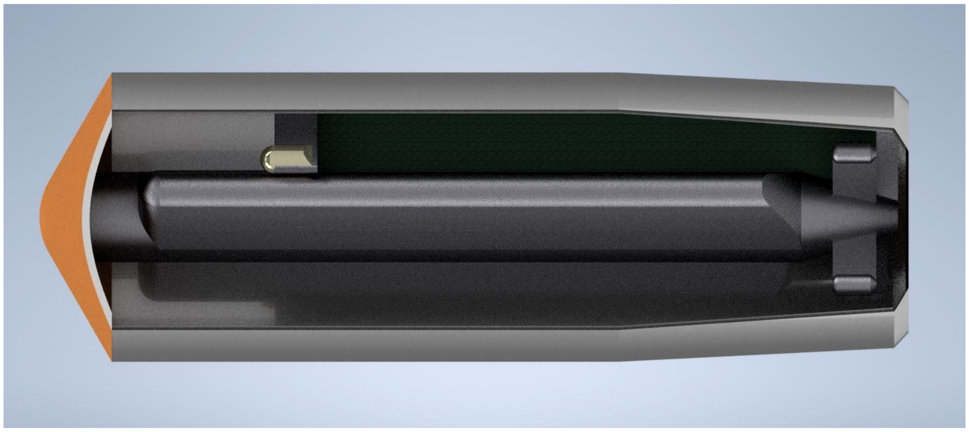

Figure 5: Aeroshell Cross-Section with Payload.

With this geometry, the aeroshell is capable of fitting within a Falcon 9 or Falcon Heavy fairing with significant diametral and length margin. Figure 6 shows an accurately scaled CAD rendering of the aeroshell within a SpaceX Falcon fairing. The SpaceX Falcon fairing has an internal diameter of 4.6m and an internal height of 11m [6].

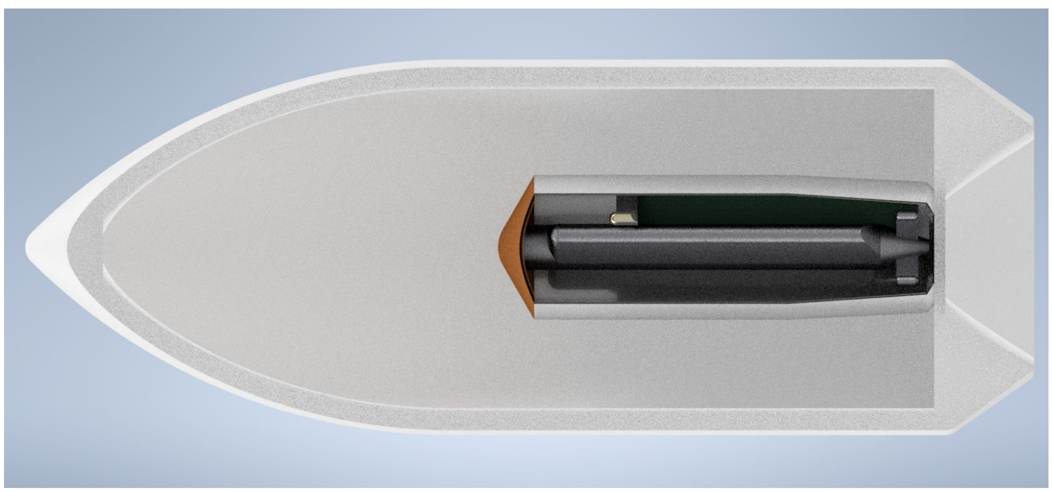

Figure 6: Aeroshell Within SpaceX Falcon Fairing.

b. Heat shield TPS Sizing Based Upon Peak Heating and Total Integrated Heat Load

An ablative heat shield will be used because its function will be single use upon entry into Titan's atmosphere. Therefore, the use of reusable thermal protection systems (TPS) is outside of the scope of the mission. The heat shield material was selected based off of the peak heat rate, which was found to be 85.889 W/cm^2. Lockheed Martin's MI-15 elastomeric silicone ablator was chosen, as it has been tested to an upper heat flux limit of 150 W/cm^2 with an associated heat of ablation of 6,276 kJ/kg [7]. The total integrated heat load was calculated to be 9891.293 J/cm^2. The required thickness of the heat shield was calculated based on this value of total integrated heat load. First the mass of the heat shield was calculated using equation 1, then the area of the heat shield was calculated using equation 2, and then equation 3 was used to find the required thickness.

$$m_{TPS} = \frac{Q_{tot} S_{HS}}{HoA} \tag{1}$$

Where Q_{tot} is the total integrated heat load, HoA is the heat of ablation, and S_{HS} is the surface area of the heat shield. Since the heat shield geometry is part sphere and part cone, S_{HS} is the sum of the sphere cap area and the lateral area of the conical frustum. The formula for S_{HS} was derived from Figure 7, which represents the geometry of a sphere-cone heat shield.

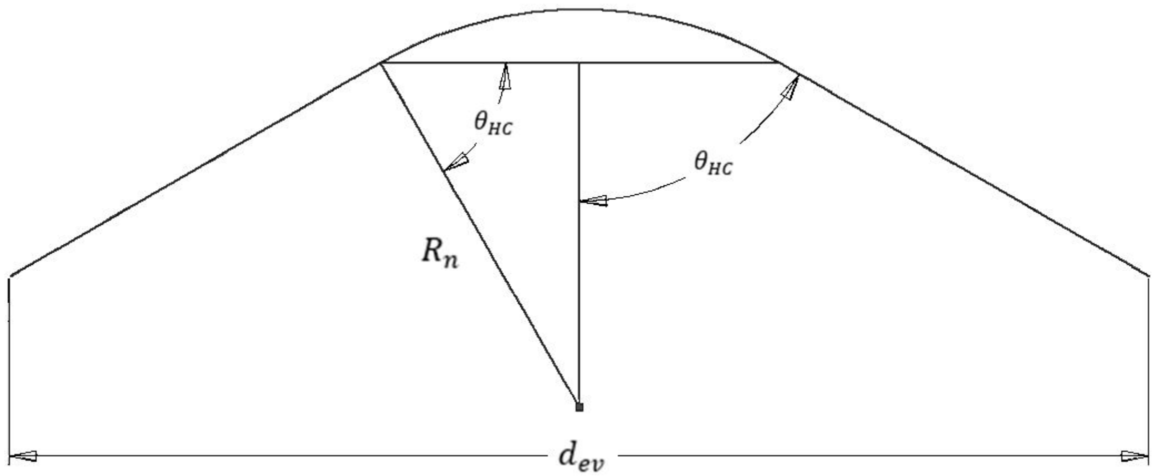

Figure 7: Sphere-Cone Heat Shield Geometry

The total heat shield area can be calculated entirely from the half-cone angle, nose radius, and forebody diameter as shown in equation 2.

$$S_{HS} = \frac{\pi(r_2^2 - r_1^2)}{sin(\theta_{HC})} + 2\pi R_n^2[1 - sin(\theta_{HC})] \tag{2}$$

Where r_1 and r_2 are defined as

$$r_1 = R_n cos(\theta_{HC})$$
$$r_2 = \frac{d_{ev}}{2}$$

Where θ_{HC} is the half-cone angle, R_n is the nose radius, and d_{ev} is the entry vehicle forebody diameter. The thickness of the ablative heat shield can then be calculated using equation 3.

$$t_{TPS} = \frac{m_{TPS}}{\rho_{TPS} S_{HS}} \tag{3}$$

Where ρ_{TPS} is the density of the ablator material. The required no-margin mass of the heat shield was calculated to be 52.643 kg. The corresponding required no-margin thickness of the heat shield was calculated to be 3.283 cm. The calculations in the other sections of this paper assume a safety factor of 1.5 is multiplied to the heat shield mass to account for design margins. The heat shield mass with a 1.5 safety margin comes out to 78.964 kg.

c. Structure Material, Thickness, and Mass Based Upon Peak Deceleration Load

The material, thickness, and resulting mass of the structure was driven by the peak deceleration load. The peak deceleration experienced during the mission occurs during ballistic entry. The geometry of the structure was approximated to take the shape of a cylindrical shell which encapsulates the submarine. To determine if a proposed shell thickness would cause the structure to buckle, the longitudinal stress resulting from the peak deceleration load was compared with the critical buckling stress of the shell geometry given material parameters. The longitudinal and critical buckling stress is shown in equations 4 and 5 respectively.

$$\sigma_{Long} = \frac{F_{peak} R_{ev}}{2t} \tag{4}$$

Where

$$F_{Peak} = m_{sub+fore} a_{peak}$$

$$\sigma_{cr} = 0.6\gamma \frac{Et}{R_{ev}} \tag{5}$$

Where

$$\gamma = 1 - 0.901(1 - e^{-\phi})$$

$$\phi = \frac{1}{16}\sqrt{\frac{R_{ev}}{t}}$$

Where F_{peak} is the peak deceleration load, calculated by multiplying the mass of the sub and heat shield by the peak deceleration, a_{peak} is the peak deceleration, E is the elastic modulus of the material of choice, t is the thickness of the structure shell, and R_{ev} is the radius of the aeroshell. The peak deceleration load was calculated to be 50.820 kN. Aluminum was first evaluated as it is the same material that was used on the Huygens probe aftbody structure [2]. It was calculated that an aluminum shell of t = 3.1 mm thickness would provide a safety factor of 1.1. To find the mass of the structure, the 3.1 mm shell thickness and aluminum density properties were applied to a computer aided design (CAD) model of the aftbody structure that was created to accommodate the submarine's geometry. Figure 8 shows a rendering of the aftbody structure with the proper shell thickness and material properties applied.

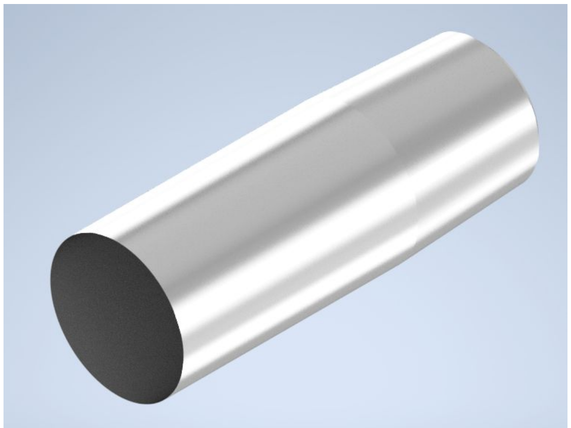

Figure 8: Aluminum Aftbody Structure

By taking the required buckling thickness with 1.1 safety factor, aluminum material properties, and submarine packaging factors into account, the mass of the aftbody structure was determined to be 302.128 kg.

d. Parachute Type, Size, Peak Load, Terminal Velocity, and Mass

The entry, descent, and landing architecture of this mission requires the use of a parachute to descend the payload in the subsonic regime until splashdown. A Disk-Gap-Band (DGB) parachute was selected because of its prior success with the Titan Huygens probe descent [2] and Mars Viking lander descent [8]. At mach 1, the DGB parachute will deploy simultaneously with the separation of the heat shield from the payload. Therefore, from parachute deployment to splashdown, the submarine will be the only mass suspended by the parachute. The diameter of the parachute was chosen to be 12m as to provide a >1.5 factor of ballistic coefficient separation between the submarine and heat shield to avoid collision post-separation. The opening load of the parachute was calculated using equation 6.

$$D_{p,0} = \frac{1}{2} C_x C_{Dp} S_p \rho V^2 \tag{6}$$

Where C_x is the opening load factor, C_{Dp} is the parachute drag coefficient, S_p is the area of the parachute, ρ is the exponential model for atmospheric density, and V is velocity. The peak load was calculated to be 42.2 kN. Figure 9 shows a plot of the drag force over time after the parachute is deployed at mach 1.

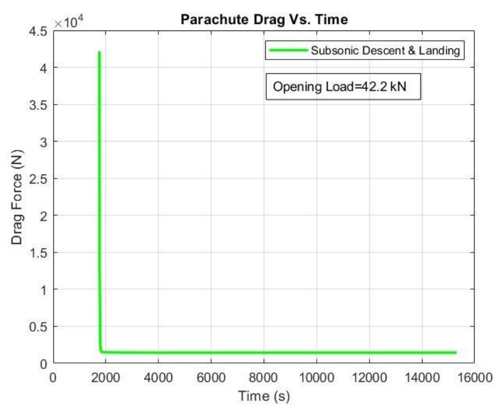

Figure 9: Parachute Drag Force Vs. Time.

The terminal velocity under the parachute at an altitude of 100 m was calculated using equation 7.

$$V_{Term} = \sqrt{\frac{2 m_{sub} g}{\rho (C_{Dp} S_p + C_{Dsub} S_{sub})}} \tag{7}$$

Where m_{sub} is the mass of the submarine, g is Titan's gravity, C_{Dsub} is the submarine drag coefficient, and S_{sub} is the frontal area of the submarine. The submarine drag coefficient was based on the drag coefficient of a rounded-nose section from White [9]. At an altitude of 100 m, the terminal velocity of the submarine under the parachute was calculated to be 2.8 m/s.

Following the design of the Huygens DGB parachute, the Hyperion DGB parachute will have kevlar lines and the canopy will be made from nylon. The total mass of the parachute and deployment device were sized based on the Huygens Descent Control Subsystem (DCSS) and scaled by the ratio of the diameters of the Hyperion parachute to the Huygens main parachute as shown in equation 8.

$$m_{Hypchute} = \frac{d_{Hypchute}}{d_{Huygchute}} m_{DCSS} \tag{8}$$

Where $m_{Hypchute}$ is the mass of the Hyperion parachute and deployment device, $d_{Hypchute}$ is the diameter of the Hyperion parachute, $d_{Huygchute}$ is the diameter of the Huygens main parachute, and m_{DCSS} is the mass of the Huygens DCSS. The mass of the combined Hyperion parachute and deployment device was calculated to be 17.5 kg.

e. Orbital and ACS Pre-EDL Propulsion: Engine, Fuel, and Mass

The EDL architecture of this mission will not use any propulsive descent methods. This section acts purely as a conceptual first glance into the possible engines that could be used in the orbital phase prior to entry as well as the small attitude control system (ACS) thrusters that could be used to maintain proper attitude of the vehicle so it does not tumble upon descent.

First, a conceptual first glance was done regarding the use of a propulsion system to perform orbital maneuvers in orbit, prior to entry. Since the operating regime of this engine system is in space, a monopropellant engine would be the logical choice, since it doesn't require the vehicle to carry oxidizer, while still providing adequate thrust and specific impulse. NASA has used Aerojet Rocketdyne monopropellant engines for past Mars landers, and they are very well-respected in the space industry. Our engines of choice will be Aerojet's MR-104H Hydrazine engines, and we will use three of these engines in tandem in order to increase our thrust and delta v. Having three engines will also provide stability by allowing the three engines to balance any induced rotation. Each engine provides 510 Newtons of thrust, and a specific impulse of 223 seconds. They operate at a mass consumption rate that ranges from 0.2495 kg/s to 0.0908 kg/s [10].

For the ACS, a set of MRE 5.0 monopropellant thrusters will be used. [11] In general, satellites are required to correct for an estimated 75 m/s per year for attitude control, and since the EDL phase of this mission will take less than a day, that number was converted to 0.21 m/s. To calculate the required ACS fuel, an intermediate all-up vehicle mass of 1100 kg was used; since the specific impulse of hydrazine is 232 seconds, the ACS fuel mass came out to be 0.1 kg. Using a density of 1021 kg/m^3, the volume of the fuel becomes 0.099 liters, which leads to the choice of a 3.8 liter NG-80568-1 tank from Northrop Grumman, which is their smallest available tank [12].

f. Landing attenuation Method and Peak Load

Since the submarine will be landing in Kraken Mare via splashdown, it was critical to determine the maximum splashdown velocity such that the submarine does not exceed the 5g deceleration requirement. The splashdown velocity equation given by Weaver [13] is shown in equation 9.

$$V_{Splash} = \sqrt{\frac{2 m_{sub} a_{max}}{C_p \rho_{CH4} S_{sub}}} \tag{9}$$

Where $a_{max} = 5*9.81 \frac{m}{s^2}$ is the maximum splashdown acceleration, $C_p = 1$ is the impact drag coefficient based on the submarine's nose geometry and data provided by Weaver [13], ρ_{CH4} is the density of liquid methane, and S_{sub} is the frontal area of the submarine. From this equation it was calculated that the submarine must impact Kraken Mare below a velocity of $V_{Splash} = 27.013\ \frac{m}{s}$ to stay beneath the 5g deceleration limit.

The terminal velocity of the submarine under parachute just before splashdown was calculated to be 2.8 $\frac{m}{s}$. Since this terminal velocity is much lower than the splashdown velocity, it was determined that no landing attenuation devices will be needed. Using this terminal velocity value it was possible to calculate the acceleration experienced during splashdown by rearranging equation 9 into equation 10.

$$a_{Splash} = \frac{\frac{1}{2} C_p S_{sub} \rho_{CH4} V_{Term}^2}{m_{sub}} \tag{10}$$

It was calculated that the acceleration experienced during splashdown was 0.543 $\frac{m}{s^2}$. By multiplying this value for a_{Splash} by m_{sub}, the splashdown load was calculated to be 523.588 N.

g. Telecommunications Approach, Integration Into the Entry System, Required Power, and Link Budget

The telecommunications approach is to have direct communication between Titan and Earth during the entry, descent, and landing (EDL) phases and during surface operations. During the EDL phases of the mission, we plan to have the system autonomously navigate through the atmosphere of Titan for a safe landing on Kraken Mare. After landing, the UUV would establish a communication link Direct-to-Earth (DTE) and report the recorded telemetry through the EDL phases for analysis of Titan's atmosphere. Once telemetry is completely reported, the UUV is ready to accept commands from the mission operators on Earth. These commands would be to select one of the pre-loaded mission plans to start autonomous exploration through Kraken Mare. The UUV will explore through Kraken Mare for 8 hours at a time and surface for 16 hours [5]. During the 8 hours, the UUV will collect sensor data and information about Kraken Mare. During the 16 hours on the surface, the UUV will send back telemetry and sensor data. Once completed, the UUV will then again be ready to accept a command for a different mission plan and autonomously navigate through Kraken Mare. This process is repeated for the duration of the UUV's lifetime.

During surface operations, the UUV will communicate back to Earth via the Deep Space Network (DSN) operating in the X-band region (9 GHz). The X-band region was selected for compatibility with DSN. For our mission, the desired data rate to support mission data being transmitted back to Earth is 5 kbps. With the 5 kbps data rate, this amounts to a total of 288 Mb per day (based off of being on the surface for 16 hours). Our communication system was designed to support this data rate, which assumed a 4-meter diameter transmitting antenna dish with an Traveling Wave Tube Amplifier (TWTA) operating at 90 W. Our design assumes that our phased array can support the necessary transmitting antenna gain as calculated from the equivalent 4-meter diameter antenna dish. The 34-meter diameter dish was assumed for the DSN antenna [14]. Based off of the Cassini Orbiter telecommunications radio frequency subsystem (RFS), our communication system will draw 106.25 W of power [15].

A link budget (Table 3) was created to ensure that our assumptions still provide an adequate Signal-to-Noise ratio or Eb/No, which is calculated to be around ~3.1 dB. This gives us the option to use the QPSK modulation format with a ⅚ code rate (Eb/No = 2.99 dB). By standard practices, a required link margin of 3 dB was imposed onto our communication system. All losses in the link budget were estimated. Below is equation 11, which was used for calculating Eb/No:

$$\frac{E_b}{N_o} = \frac{(P * L_l * G_t) * L_s * L_a * G_r}{k * T_s * R} \tag{11}$$

Where:

- Eb/No - energy per bit / noise power spectral density ratio
- P = Transmitted Power
- L_l = Line Loss
- G_t = Transmitting Antenna Gain
- L_s = Space Loss
- L_a = Attenuation Loss
- G_r = Receiving Antenna Gain
- k = Boltzmann Constant (1.38 E-23 J/K)
- T_s = Temperature Noise
- R = Data Rate
- Notes: Atmospheric Loss, Required Link Margin, and Uncertainty Link Margin is not listed in equation 11, however they were included in the calculation for Eb/No in Table 3).

Table 3: Link Budget for Surface Operations

Titan Sub to Earth DSN during Surface Operation		
Parameter	**Linear Value**	**dB Value**
Transmit		
Transmitting Antenna Diameter	4 m	
Efficiency of Transmitting Antenna	0.7	
Selected Frequency	9 GHz	
Transmitting Antenna Gain	99.5 kW	50.0 dB
Transmitted Power	90 W	19.5 dB
Line Loss	0.8	1.0 dB
EIRP	7,163 kW	70.5 dB
Range and Atmospheric		
Range	1.27E+9 km	
Space Loss		293.6 dB
Atmospheric Loss		2 dB
Receive		
Receiving Antenna Diameter	34 m	
Efficiency of Receiving Antenna	0.7	
Selected Frequency	9 GHz	
Receiving Antenna Gain	7,188 kW	68.6 dB
Attenuation Loss		3.0 dB
Temperature Noise of Receiving Antenna	200 K	23.0 dB
Data Rate	5 kbps	37.0 dB
Required Link Margin		3.0 dB
Uncertainty Link Margin		1.0 dB
Eb/No		3.1 dB

h. Payload Mass, Data Rate, Thermal Limits Peak Load, and Other Considerations

The submarine is the driving payload of the EDL mission. This design is based on the mission design by Oleson et al., with some changes in instrumentation based on science mission goals; most notably, a microscope camera was added to take pictures and videos of microscopic samples, and a panoramic camera similar to the one used by the Mars rovers. After taking all scientific, thermal, communication, and diving instrumentation and structure into account, the mass of the submarine comes out to be 964.41 kg. It's maximum dimensions fit within a 5.87 m long, 1.75 m diameter cylinder. Though this payload is unusually large, it will fit in existing launch vehicles, such as SpaceX's Falcon 9 and Falcon Heavy fairings if mounted vertically. Vertical mounting is a driver for the EDL vehicle's aeroshell shape, since it would have to fit into the fairing but have a wider area than the fairing if mounted horizontally. The design of the mission's telecommunication system is comparable in data rate and required power to the telecom system of the mission design by Oleson, et al. Therefore, their telecommunication subsystem mass served as a good estimate for this mission mass calculation. The submarine will use two Stirling Radioisotope Generators (SRGs) to generate the required power, weighing a total of 130 kg. These were chosen for their high energy density and because they would provide additional excess heat which could be used to keep the onboard instruments warm. Excess heat from the SRGs will be transported to heat the electronics via a pump system weighing a total of 100 kg. The submarine will use fore and aft titanium ballast tanks to control depth, weighing a total of 15 kg. The structure holding the instrumentation and supporting the submarine pressure vessel is designed to withstand 5 g's, with a safety factor of 1.4, during launch. This is a key design driver for the entry vehicle. Such a heavy payload would require a decelerator with a large area to slow down the vehicle with a low enough peak deceleration.

Calculations:

7. Velocity Versus Altitude Indicating the Different EDL Mission Phases

The atmospheric EDL phases in this mission include ballistic hypersonic entry, supersonic descent, and subsonic descent and landing. The Allan and Eggers [16] model was used in the hypersonic regime where velocity was calculated using equation 12.

$$V = V_e \exp\left(\frac{\rho_o}{2\beta A \sin(\gamma)} e^{-Ah}\right) \tag{12}$$

Where ρ_0 is the surface atmospheric density, A is the inverse scale height, h is the altitude, γ is the flight path angle, and β is the ballistic coefficient which is calculated using equation 13.

$$\beta = \frac{m}{C_{Dev} S_{Dev}} \tag{13}$$

Where m is the mass of the entry vehicle, including the aftbody, submarine, and heat shield, C_{Dev} is the entry vehicle drag coefficient, and S_{ev} is the frontal area of the entry vehicle.

At Mach 5, the aftbody separates from the entry vehicle, leaving the submarine and heat shield to descend as one unit until Mach 1. The equation of motion for the free body diagram in Figure 10 was used to calculate velocity for the supersonic regime.

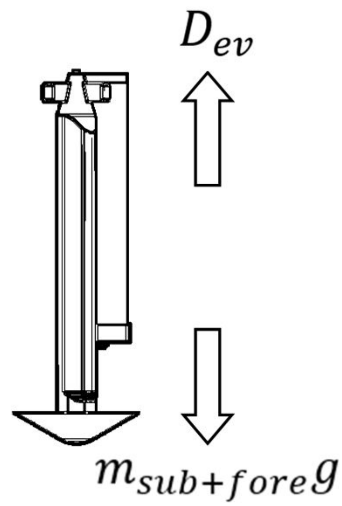

Figure 10: FBD of Entry Vehicle from Mach 5 to Mach 1

As seen in Figure 10, the forces acting on the entry vehicle in the supersonic regime are D_{ev} , drag on the entry vehicle, and $m_{sub+fore}g$, mass of the submarine plus mass of the heat shield multiplied by Titan's gravity. D_{ev} is defined by equation 14.

$$D_{ev} = \frac{1}{2}\rho V^2 C_{D_{ev}} S_{ev} \tag{14}$$

Where ρ is the exponential model of atmospheric density, and V is velocity. The equation of motion for the FBD in Figure 10 was employed using finite differencing as represented by equation 15.

$$V_k = V_{k-1} + [g - \frac{1}{2m_{sub+fore}}\rho_0 exp(-Ah_{k-1})V_{k-1}^2[C_{D_{ev}}S_{ev}]\Delta t \tag{15}$$

Where Δt is the time increment and k is the current time step.

At Mach 1, the parachute is deployed at the same time the heat shield is separated from the submarine. Therefore, the FBD in the subsonic regime consists of the submarine descending under a parachute. Figure 11 shows the FBD of the submarine-parachute system in the subsonic regime.

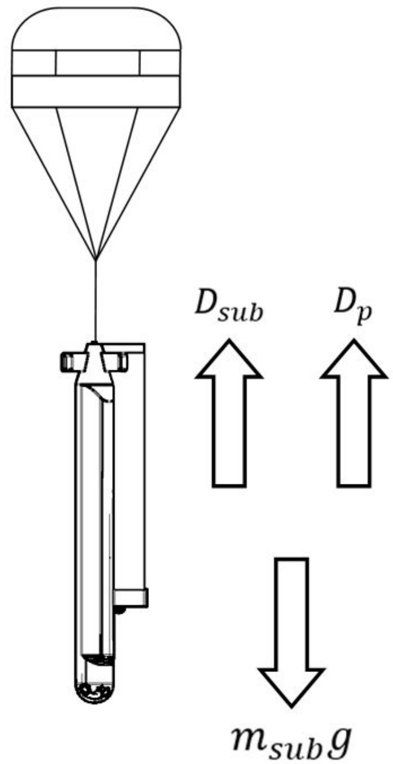

Figure 11: FBD of Submarine-Parachute System from Mach 1 to Splashdown

As seen in Figure 11, the forces acting on the submarine-parachute system in the subsonic regime are D_{sub} , drag on the submarine, D_p , drag on the parachute, and $m_{sub}g$, mass of the submarine multiplied by Titan's gravity. D_{sub} is defined by equation 16.

$$D_{sub} = \frac{1}{2}\rho V^2 C_{Dsub} S_{sub} \tag{16}$$

The equation of motion for the FBD in Figure 11 was employed using finite differencing as represented by equation 17.

$$V_k = V_{k-1} + [g - \frac{1}{2m_{sub}}\rho_0 exp(-Ah_{k-1})V_{k-1}^2[C_{Dev}S_{ev} + C_{Dp}S_p]\Delta t \tag{17}$$

The altitude from entry until splashdown was calculated at each time step as represented in equation 18.

$$h_k = h_{k-1} - V_{k-1}\Delta t \tag{18}$$

Figure 12 shows the velocity at each phase in the EDL plotted against altitude using equation 12 for the hypersonic regime, equation 15 for the supersonic regime, and equation 17 for the subsonic regime.

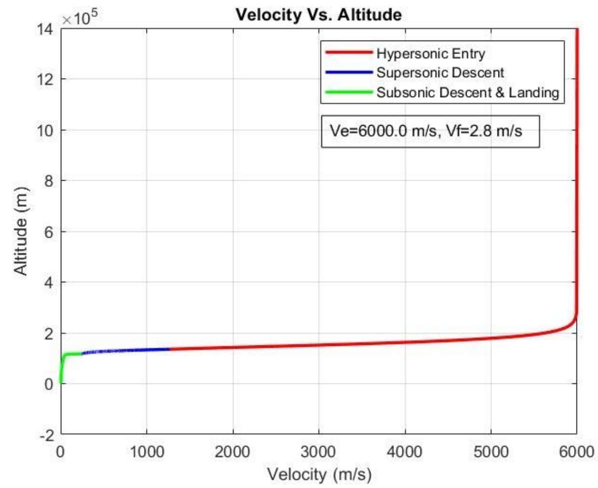

Figure 12: Velocity Vs. Altitude

8. Downrange Distance Versus Altitude

The downrange distance in the hypersonic, supersonic, and subsonic regimes was calculated using finite differencing with equation 19.

$$DS_k = DS_{k-1} + V_k \cos(\gamma) \Delta t \tag{19}$$

Where γ is the flight path angle. In the hypersonic regime, flight path angle is limited by the max deceleration requirement of 5 g's defined in the entry system requirements. The flight path angle is 8.5 degrees in the hypersonic and supersonic regimes. In the subsonic regime, the flight path angle is defined as $\gamma = \theta_{trim} - \pi/2$, where θ_{trim} is the constant parachute trim angle. A constant parachute trim angle of 10 degrees was chosen from parachute deploy until splashdown. Figure 13 shows the downrange distance plotted against altitude for each EDL phase.

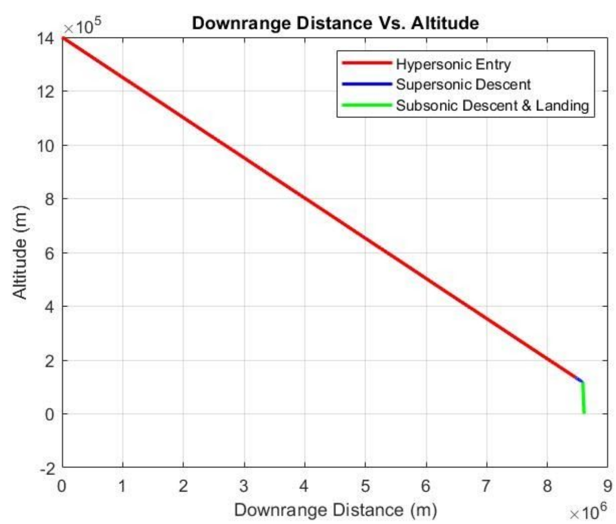

Figure 13: Downrange Distance Vs. Altitude

9. Heat Rate Versus Altitude

The convective heat rate was plotted as a function of altitude using the Sutton-Graves formula shown in equation 20 [17].

$$q_{conv}(h) = C_{SG}\sqrt{\frac{\rho(h)}{R_n}}V(h)^3 \tag{20}$$

Where C_{SG} is the Sutton-Graves constant and R_n is the nose radius. Figure 14 shows a plot of the heat rate as a function of altitude during the different EDL phases.

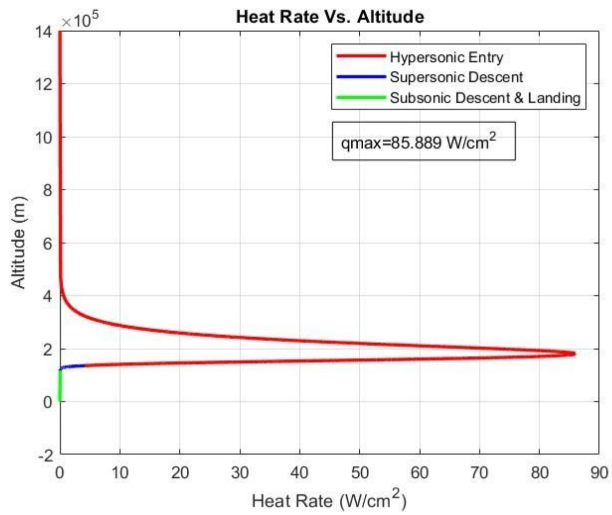

Figure 14: Heat Rate Vs. Altitude

10. Peak Deceleration and Altitude of Peak Deceleration

The peak deceleration and altitude of peak deceleration was calculated using equations 21 and 22, respectively.

$$n_{\max} = \frac{V_{atm}^2 A \sin\gamma}{2 g_E \exp(1)} \tag{21}$$

$$h_{n,\max} = \frac{1}{A}\ln(-2C) \tag{22}$$

Where

$$C = \frac{\rho_o}{2\beta A \sin\gamma}$$

The magnitude of peak deceleration was calculated as 5 g's and occurred at an altitude of 158.16 km.

11. Peak Heat Flux and Altitude of Peak Heating

The peak heat rate and altitude of peak heating was calculated using equations 23 and 24, respectively.

$$q_{max} = C_{SG}\sqrt{\frac{-1}{R_n} * \frac{\beta\ \mathrm{A}\sin(\gamma)}{3}}(0.6055\ V_e^3) \tag{23}$$

$$h_{qmax} = \frac{\ln(-6C)}{A} \tag{24}$$

Where C_{SG} is the Sutton-Graves constant. The max heating rate experienced during entry was calculated to be 85.889 W/cm^2 and occurred at an altitude of 180.29 km. This max heat rate drove the material choice for the ablative heat shield.

12. Total Integrated Heat Load

The peak heat load was calculated using equation 25.

$$Q_{total} = C_{SG}\sqrt{\frac{-\beta\pi}{R_n A\sin(\gamma)}}V_e^2 \tag{25}$$

The total heat load was calculated to be 9891.293 J/cm^2. This value drove the thickness of the ablative heat shield.

13. MEL of the Entry System Including Subsystem Estimates

The requirements of the scientific mission are similar to the project by Oleson, et al. There are notable differences between required scientific equipment, such as the microscope camera and intake system this project will use that Oleson, et al. did not use. However, to design the EDL system, their structural mass serves as an accurate estimate, putting the structural mass of the submarine at 446 kg. A GP:50 MODEL 7720 pressure sensor weighs 0.2 kg, and 8 will be used for the ACS system, putting their weight at 1.6 kg [18]. An Omega CY7 temperature sensor weighs 1.4 grams, allowing for many to be used for redundancy and temperature gradient measurement. A set of 10 will weigh 14 grams [19]. The required compressor for this system will be similar to the one used in the report by Oleson, et al, and will weigh 16.5 kg [5]. A microscope camera such as a Sony IMX222 weighs 9 kg if including structural addition for mounting and sample containment [20]. The link budget requires a telecommunication system that is estimated to weigh similar to the one used by Oleson, et al, at 30.5 kg. A panoramic camera like the one used on the Mars rovers weighs 0.3 kg. Using additional sensors used for attitude and position determination, a similar propulsion system, ACS, and C&DH subsystems as the report by Oleson, et al adds 118, 20.6, 32.9, and 44 kg respectively. The total mass of the submarine and scientific equipment and payload, as well as the entry vehicle computer comes out to be 964.4 kg.

Table 4: Science Payload and Payload Telecom MEL

Item	Mass (kg)
Structure	446
Pressure Sensors (8 units)	1.6
Temperature Sensors (10 units)	0.014
Power System	130
Thermal Control	100
Ballast Tanks	15
Compressor	16.5
Microscopic Imager	9
Submarine Telecom	30.5
Panoramic Cameras	0.3
Attitude and Position Sensors	118
Propulsion	20.6
Command and Data Handling (C&DH)	44
Attitude Control System	32.9
Total	**964.4**

The largest contributors to the EV mass were the aftbody structural mass and the heat shield. The separation mechanism will weigh 11.4, as described in the mission by [2]. The weight of the attitude control system, including fuel, tank, and four monopropellant thrusters, is 9.5 kg. Other contributors include actuation mechanisms for aftbody separation, heat shield separation, and parachute release. The total EV mass comes out to be 421.1 kg.

Table 5: Aeroshell and Entry Vehicle MEL

Item	Mass (kg)
Heat Shield	79.0
Separation Mechanism (Including Power System)	11.4
Pressure Sensors	1.6
Temperature Sensors	0.014
ACS Thrusters	8
ACS Fuel	0.1
ACS Fuel Tank	1.4
Aftbody Structural Mass	302.1
Parachute and Mortars	17.5
Total	**421.1**

14. PEL of the Entry System Including Subsystem Estimates

The submarine will require 723 W of power to complete the mission, which will be supplied by two Stirling radioisotope generators (SRG). The propulsion system will draw 440 W to move when submerged [5]. Attitude control and command and data handling will require 123 W. The microscope camera and spectrometer will require a combined 5 W; for reference, a Sony IMX222 microscope camera requires 2.5 W, and Lovibond TR500 spectrometer requires 2.5 W. The sample intake system is estimated to require 22 W; this is a conservative estimate that is equal to a fifth of the Mars rover power, which had its own sample intake system [21]. The power required for the telecommunication subsystem was calculated in detail in the telecommunication section above; 30.5 Watts will be required for it. The panoramic camera will require 6.5 W [22]. Finally, the thermal control system of the submarine will require 20 W.

During entry, telemetry data will be recorded by temperature and pressure sensors on the outside of the EV. Their data will be processed and recorded by the submarine computer, and the control inputs will be sent from the computer to the ACS thrusters. Thus, no additional electric power is required for this subsystem.

The aeroshell will be separated from the submarine via a system nearly identical to that used by the Huygens probe [2]. The system includes: three springs to provide separation force, three helical guide rails to ensure a controlled ejection and spin trajectory, and three pyronuts which connect the aeroshell to the submarine. Since this only requires a signal from the submarine computer, there is no additional power requirement for this mechanism.

The parachute will be released from a mortar. The system includes the following mechanisms: a cylinder to contain the parachute pointed aft of the submarine, the packaged 12 m diameter DGB parachute, and a small explosive device to accelerate a sabot along guide rails to eject the parachute out of the cylinder. Like the aeroshell separation, this only requires a signal from the submarine computer and will not require additional power.

Table 6: Submarine (Payload) PEL

Item	**Power (W)**
Pancam	6.5
Sample Intake System	22
Microscope Camera	2.5
Spectroscope	2.5
ACS and C&DH	123
Thermal Control System	20
Submarine Propulsion	440
Telecommunications	106.3
Total	723

Table 7: Entry Vehicle PEL

Item	Power (W)
EDL Telemetry Data Recording System	123
EDL ACS	63
Aeroshell Separation System	1
Parachute Release System	1
Total	188

15. Unique Environments and Entry System Constraints

The most unique environmental obstacle that the payload (the submarine) must be able to withstand is the immense cold of the cryogenic methane lake, Kraken Mare. The lake is composed of hydrocarbons such as methane which are around 93 Kelvin in temperature [23]. These temperatures present a unique challenge in terms of materials selection, structural considerations (temperature affects yield strength favorably, but affects brittleness unfavorably), and preventing ice formation that would hinder mechanical motion or electrical operation. This will be addressed by utilizing the submarine's innate heat generation as a byproduct of its power systems. Additionally, the exact composition of Kraken Mare is unknown, so the density of the lake will have to be estimated, and any potential corrosive or reactive properties anticipated. Finally, the submarine is designed to withstand a maximum load of 5 Earth g's, which further constrains the design of the entry vehicle, as described in previous sections.

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