INVESTIGATING THE COMPOSITION OF ENCELADUS VIA PRIMARY LANDER AND UNDERWATER MICROORGANISM EXPLORER (ICEPLUME)

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ABSTRACT

Investigating the Composition of Enceladus via Primary Lander and Underwater Microorganism Explorer (ICEPLUME) is a proposed mission to Enceladus originating from an undergraduate senior design course. ICEPLUME consists of an orbiter, lander, and ice-melting probe encapsulated within an aeroshell. The mission utilizes solar electric propulsion and an aero-gravity assist maneuver through Titan's atmosphere to maximize mass delivered to the final orbit. The ellipsled aeroshell was determined to be stable with a lift-to-drag ratio of 0.49. A composite payload deck is used as the orbiter's structure. The lander acts as a communication relay between the probe and orbiter once deployed. The probe contains two radioisotope thermoelectric generators, water jets, and science instruments to investigate the composition of the ice as it descends beneath the surface of Enceladus to the proposed subsurface ocean.

1. DESIGN CONSIDERATIONS

1.1 Scientific Motivation

Enceladus, the second moon of Saturn, has a diameter of roughly the length of Great Britain and a mean orbital distance from the sun of approximately 238,000 km [1]. Considering the satellite's size and distance from the sun, Enceladus should have cooled long ago, yet it continues to show geological and geothermal activity. The multitude of geological features on the surface indicates a wide range of events in its history. The southern region of the moon is of particular interest because of its expansive smooth areas, indicating recent resurfacing processes [1]. Located in this region are features known as the 'tiger stripes'. These cracks and fissures are warm enough to heat and eject ice-water plumes into space, continually refreshing the southern surface of the moon with a new supply of ice. Flybys by Cassini discovered a plume of gas and dust erupting from the South Pole, contributing to Saturn's E-ring. Enceladus is of interest to scientists and astronomers because of the possibility of life existing beneath the ice [2]. Extremophiles survive on Earth under a wide range of previously-thought uninhabitable environments. The presence of water as well as the fundamental four elements of life: carbon, hydrogen, oxygen, and nitrogen, provide strong evidence for the existence of astrobiology. Scientists now agree that Enceladus is emerging as the most habitable spot in the solar system besides Earth. "It has liquid water, organic carbon, nitrogen [in the form of ammonia], and an energy source," says Chris McKay, an astrobiologist at NASA's Ames Research Center. Besides Earth, he says, "there is no other environment in the Solar System where we can make all those claims" [3].

1.2 Science Goals

Based on the scientific motivation presented above, there are certain science goals that ICEPLUME will investigate. These goals are as follows:

1) What is the heat source of the moon?
2) What drives the large ice-water plume?
3) What is the plume production rate and does it vary?
4) What are the plume effects on the structure and composition of Enceladus?
5) Do the conditions of the plume source support pre-biotic chemistry and does microbial life exist?

This list of goals, developed with the help of a proposed NASA mission to Enceladus as a framework, drives the data collection as well as the instrumentation aboard the orbiter, lander, and probe [2].

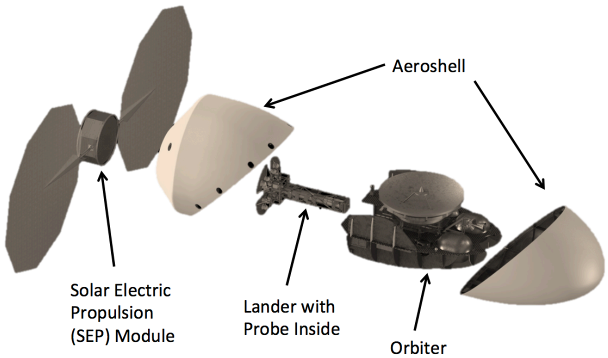

Fig. 1. Each of the ICEPLUME subsystems contribute to the success of the mission.

1.3 Design Constraints

Before the spacecraft could be designed, major design constraints were outlined. Two Quality Function Deployment (QFD) reports as well as a Product Design Specification (PDS) addressed these restrictions. Through the process of creating and evaluating design possibilities a final design emerged. The first QFD specified overall mission parameters such as propulsion methods, launch vehicles, science goals, and possible trajectories. The second QFD specified the instrumentation on the orbiter and lander to accomplish the desired scientific goals. The PDS constrained the final specifications for the spacecraft including restrictions based on Planetary Protection Act standards and the environment in which the orbiter and lander will be surviving; micrometeoroids, dust, corrosive gas and ice accumulation were all issues taken into consideration during the design of the orbiter and lander. The strictest constraints imposed on the design were the launch vehicle size and maximum mass allowance.

1.4 Mass Breakdown

A mass breakdown analysis determined the allowable payload mass allotted to each of the subsystems. The liquid propellant, Solar Electric Propulsion (SEP) module, SEP Propellant, science instruments, and orbiter structure mass estimates were based on Titan Saturn System Mission (TSSM), a similar proposed joint NASA/ESA mission to Titan and the Saturn System [4]. The propulsion system mass estimate was based on values used for the similar Cassini mission. The aeroshell, lander, and probe mass estimates were calculated from the current design. The power system mass is based on five radioisotope thermoelectric generators as described in the power section. The estimate for the communication system is based on the mass of the communication system used for the Cassini mission [5]. The mass breakdown is in Table 1, with a total mass from LEO of 7559 kg.

Table 1. Mass breakdown of the ICEPLUME mission.

Total Mission Mass	7559 kg
Communication System	1%
Power Systems	2%
Science Instruments	2%
Orbiter Structure	6%
Lander & Probe	7%
SEP Module	15%
Aeroshell	22%
Propulsion	45%

2. ORBITER SUBSYSTEMS

2.1 Orbiter Power System

Five Advanced Stirling Radioisotope Generators (ASRG) will provide the 500W necessary to power communications, data, and science instruments aboard the orbiter. ASRGs were chosen to maximize the power output for the least amount of mass. Solar panels were ruled out as the power source after a calculation for an output power of 500 watts with the best available efficiency of 33% equated to 101.8 square meters of solar panels.

2.2 Orbiter Thermal Control

Radioisotope heater units (RHU) keep science instruments and electronics warm once at Enceladus where the temperature can get as low as -200 degrees Celsius. Each produces 1 thermal watt of power and is only 3.3 cm tall by 2.5 cm in diameter. Their small mass of 40 grams does not significantly change the center of mass and each unit powers themselves, making them negligible in most of the calculations. However, the heat produced by RHUs and the power system while in the aeroshell during interplanetary flight will be significant. An active cooling system is used to prevent the instruments from overheating or damaging parts of the spacecraft designed to survive cold temperatures. MLI (multi-layer insulation) blankets on the outside of the science instruments and around the critical electronic component housings are used to make sure cold temperatures do not damage the instruments.

2.3 Communication Radio Frequency Subsystem

For communication, the radio frequency subsystem produces an X-band carrier at 8.4 Ghz [6]. The radio frequency subsystem then modulates the X-band carrier with the data received from the data subsystem and amplifies the power of the X-band carrier to produce 20 watts from the Traveling Wave Tube Amplifiers and delivers the signal to the antenna subsystem [6]. There are three antennas on the orbiter, a High-Gain Antenna and two Low-Gain Antennas. The High-Gain Antenna is used primarily to communicate with Earth. The High-

Gain Antenna is a 3.5-meter parabolic primary reflector similar to that used on the Cassini spacecraft [6]. At distances up to 10.6 AU from Earth, radio signals will take up to 84 minutes to communicate from Earth to the orbiter [6]. The model of the High Gain Antenna is shown in Fig. 2.

2.4 Data Subsystem

The data subsystem on the orbiter processes and stores data from the science instruments and other subsystems. This is the same system used to provide the commands to the science instruments and subsystems. All of the orbiter components are connected through one orbiter bus system as in the Cassini mission [7]. The data subsystem is equipped with a solid-state drive to record data from the science instruments [7]. This system also is used to store flight programs. The space available on this solid-state drive is 4 gigabytes [7]. In order to ensure that there is enough space available, data from the science instruments will be periodically sent to Earth through the communication subsystem described above.

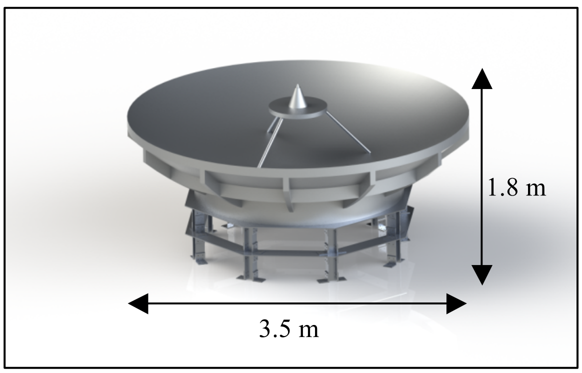

Fig. 2. The High Gain Antenna enables communication between the orbiter and Earth.

3. AEROSHELL DESIGN

When coupled with solar electric propulsion (SEP), aerocapture permits up to 2.4 times more mass delivered to the final destination [8]. With ICEPLUME's complicated trajectory, there is a high demand for total required ΔV, so maximizing the mass of payload delivered is vital. Using aerodynamic drag generated by plunging into Titan's atmosphere, the spacecraft's fuel requirement is reduced by 1 km/s [2]. However, because the spacecraft would be traveling at around 6.5 km/s at the point of atmospheric entry, intense heat loads are experienced from the drag and dissociation of molecules. To protect the payload from the heat, an aeroshell forms the thermal protection system (TPS). Even though the aeroshell adds considerable weight to the overall system, the ΔV fuel mass savings of aerocapture is significant enough that the added complexity is worthwhile.

The major constraint for the ICEPLUME project is that the payload must be configured such that it completely fits inside the aeroshell with a very precise weight distribution for stability purposes. No aeroshell has ever been designed to encapsulate an entire orbiter, so the volumetric demand alone is a major challenge. Additionally, the system must be autonomously controlled during the aero-assist maneuver, perform trim/stabilization adjustments with additional reaction control system (RCS) thrusters, and carry a heavy heat shield for thermal protection. Lastly, due to stability issues with an elongated shell body, a much more complicated geometrical shape is required than that of the heritage sphere-cone aeroshell. The resulting ICEPLUME aeroshell, shown in Fig. 3, is an ellipsled shape, which is based on a proposed NASA mission to Neptune [9].

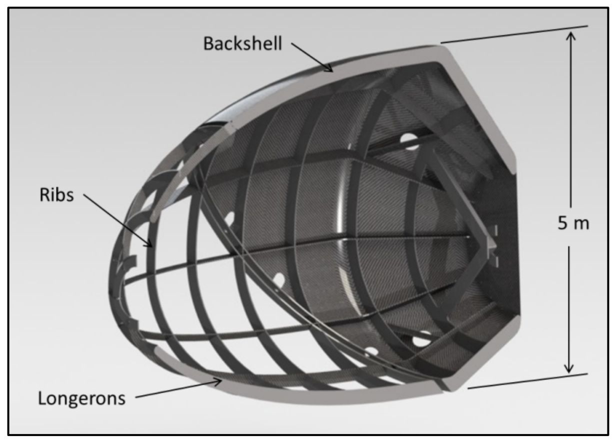

Fig. 3. The aeroshell is strengthened with an internal framework of ribs and longerons.

3.1 Aeroshell Structure and Material Selection

The structure of the aeroshell shown in Fig. 3 includes an internal framework of ribs and longerons for load distribution. Based on a proposed Titan aerocapture mission, the peak thermal and structural loads expected on ICEPLUME's heat shield are 700 W/cm^2 and 3150 Pa, respectively [10]. Heat concentration is minimized by blunting the overall shape, while loads are distributed across and through the shield via the internal framework to the backshell. Under such harsh conditions, appropriate material selection is crucial.

Based on the proposed Neptune ellipsled, the primary structure of the ICEPLUME aeroshell is made from a graphite polycyanate honeycomb sandwich structure, with modified RS-9 adhesive. This composite option offers a 14% improvement on aerial density compared to the typical aluminum honeycomb with HT-242 adhesive [11]. The honeycomb is 26 mm thick and is sandwiched between two 2 mm thick, graphite polycyanate, molded face sheets. Likewise, the ribs and longerons are made from a 35 mm isogrid sandwich structure. All composite parts are done with the lay-up technique using bias weave fiber and cyanate ester resin with vacuum bag curing in an autoclave.

If higher peak thermal loads are expected after further analysis, the heat shield would need to be constructed from 11-layer MLI (multi-layer insulation) with a high-temperature coating. This would yield a 31% improvement in aerial density when compared to the heritage design for extreme thermal loading honeycomb [11].

The aeroshell also requires an ablative material coating, which burns up in the atmosphere allowing the dissipation of heat loads away from the structure. One phenolic-carbon and three silicone-refined ablative materials (SRAM's) are used to accommodate different thermal zones. They are all produced by ARA Ablatives Laboratory and are as follows: PhenCarb-20, SRAM-20, SRAM-17, and SRAM-14. The ablator is packed and cured into large-cell Flex-Core composite honeycomb that is adhered to the outer shell surface. Once cured, a solid model-based CNC mill is used to grind the shell to its final shape.

The shell's back plate, which interfaces with the SEP module, is covered with a TPS honeycomb plate packed with Acusil II. This silicone-foam material, selected for its capability of transmitting radio waves from a low-gain antenna located on the inside of the aeroshell, is purchased from ITT Aerotherm.

The ICEPLUME aeroshell mass will be minimized with these reduced-density materials, but is still 1700 kg. At first, a design similar to the Mars Science Laboratory (MSL) was considered, and the mass was similar to the MSL aeroshell. However, fitting the orbiter inside the aeroshell required a longer shell length, increased volumetric demand, and therefore more surface area/mass for the ICEPLUME aeroshell altering the stability of the design. There is a 15% mass growth allowance on the aeroshell.

3.2 Aerodynamic Considerations

The aerodynamics of the aeroshell depend on geometry and is an iterative process that requires optimization of overall shape to meet the requirements of the mission. Aerodynamic stability and performance are governed by several parameters shown in Table 3, including drag area (C_DA), lift-to-drag ratio (L/D), and the stability parameter ($(C_MAl)\alpha$). Other calculated values include the ballistic coefficient (β) and volumetric efficiency (η_V).

Table 3. Aeroshell performance parameters.

Parameter	Definition	Specified Value
Lift-to-drag Ratio	L/D	0.20 to 0.50
Stability Parameter	$(C_MAl)_\alpha$	0.0 m³/rad MAX
Drag Area	$C_D\mathrm{A}$	10 m² MIN
Ballistic Coefficient	$\beta = \frac{m}{C_D\mathrm{A}}$	500 kg/ m² MAX
Volumetric Efficiency	$\eta_V = 6\sqrt{\pi}\frac{V}{S^{\frac{3}{2}}}$	0.80 to 0.98

Optimization of geometry is required between volumetric efficiency, drag area, and stability, where each parameter affects of the other two. Since stability is an absolute necessity, there is a bias towards minimizing $(C_MAl)_\alpha$, which is a measure of how quickly the pitching moment is neutralized under perturbations.

Aeroshells are typically conical (or bi or tri-conical) in shape and include a blunt nose to reduce thermal loading. However, the heritage sphere-cone shape could not be used for ICEPLUME due to the required length of the backshell. A modified sphere-cone for the ICEPLUME aeroshell would be unstable for nearly every position of the center of gravity. To accommodate the volumetric demand of the mission while maintaining stability, the ellipsled shape was adopted.

Seven iterations of the ellipsled were analyzed in SolidWorks Flow Simulation to identify optimization trends. The seventh shape was tentatively selected knowing that a years-worth of optimization would be required prior to launch. Due to limitations in the student software licensing, aerodynamic performance was only determined for velocities under Mach 4.

As seen in Fig. 4, the ellipsled design is much larger than the largest aeroshell ever flown; the MSL aeroshell. In fact, ICEPLUME's aeroshell demands a nearly 400% volumetric increase. But with improved materials and the disproportional 156% increase in surface area, ICEPLUME is expected to weigh less than twice that of the MSL design.

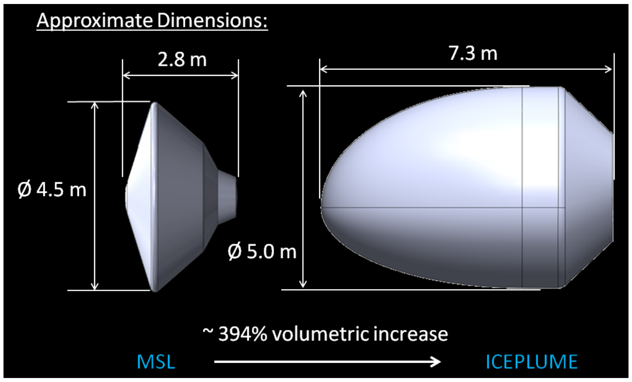

Fig. 4. ICEPLUME's aeroshell demands a large volumetric increase when compared to the MSL aeroshell, which is the largest flown aeroshell.

3.3 Orbiter Deployment

After the aero-assist maneuver, the heat shield and backshell must be jettisoned before the heat loads can transfer inward to the payload. Heritage designs feature one separation plane at the base of the heat shield. ICEPLUME's aeroshell will separate similarly, but at a diagonal, as shown in Fig. 5. The separation will be achieved by jettisoning the heat shield with ten exploding bolt separation mechanisms. One such mechanism is shown in Fig. 6.

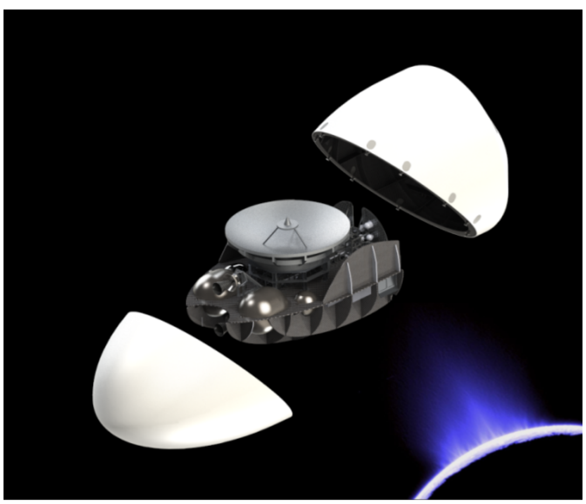

Fig. 5. The orbiter is deployed immediately after leaving Titan's atmosphere.

The separation mechanisms are modeled after the MSL design to provide torque-free, compression-enabling connections. Under peak loading, the heat shield will deflect, imparting a large moment on the connection, which would yield the components if a standard rigidly-connected exploding bolt were used. By incorporating a spherical bearing and compression spring, the load is alleviated from the connection by rotation and compression to transfer the loads to the aeroshell's backshell. The compression spring forces the heat shield and backshell to split from the orbiter, and eventually fall to Titan's surface.

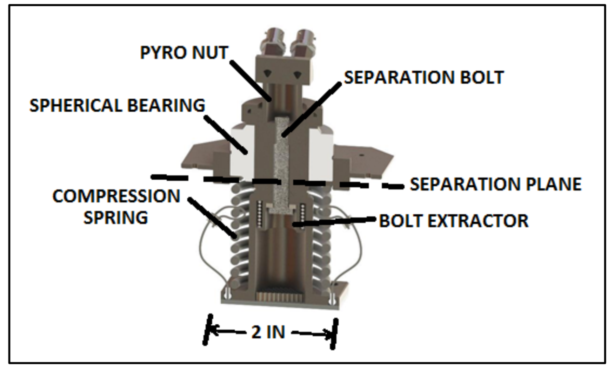

Fig. 6. Ten separation mechanisms split the aeroshell after leaving Titan's atmosphere.

4. PROPULSION

Three different propulsion systems are used at different stages of the mission to control the spacecraft, while a separate propulsion system has been designed for the lander. For clarity, these propulsion systems are introduced in the sequence that they are used.

4.1 Solar Electric Propulsion

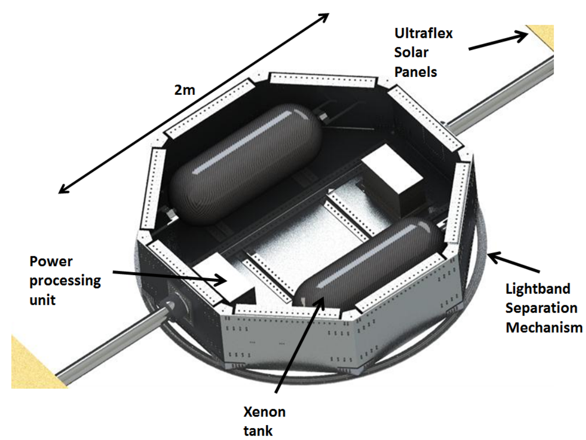

Fig. 7. The interior of the solar electric propulsion module contains xenon tanks and power processing units.

The solar electric propulsion (SEP) module propels the spacecraft from low earth orbit on its trajectory towards Saturn. Combined with multiple flybys of Earth and Venus, depending on the launch date, the SEP will provide 2.7 km/s of ΔV. The SEP is made up of two Ultraflex Solar Arrays with a collecting area of 61.2 m^2. This provides 27.6 kW of power at 1 AU. Applying a 10% degradation to the solar panels capacity at 2 AU the solar panels can produce 6.9 kW; enough to power only one thruster at maximum output. These solar arrays power four NASA Evolutionary Xenon Thrusters

(NEXT). There will be five total with one as a cold spare. The four thrusters operating at a maximum specific impulse of 4170 seconds require 450 kg of xenon to achieve the 2.7 km/s change in velocity. The tank is designed to hold 500 kg of xenon since the thrusters will not always be working at maximum capacity or efficiency. The SEP connects to the payload fairing inside the launch vehicle and to the aeroshell using Lightband separation mechanisms. This allows for an easy disconnect after the initial thrust stage thereby reducing the mass of the system.

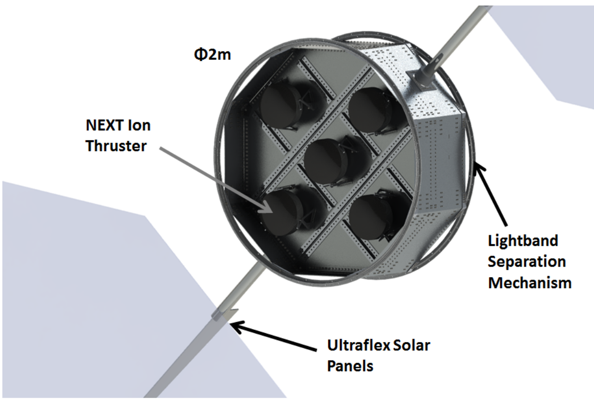

Fig. 8. The solar electric propulsion module contains five NEXT ion thrusters and two Ultraflex solar panels.

4.2 Orbiter Bipropellant System

Post aeroshell separation, the spacecraft requires further reduction in velocity in order to achieve an orbit about Enceladus. This velocity decrease is generated by gimballing and firing one of the two R-4D – 490 N bipropellant rocket engines in the direction opposite the orbiter's velocity and vectored through the orbiters center of mass. The two R-4D rocket engines (one main engine and one additional engine for redundancy) are the end points of a highly complex bipropellant system. This system is strongly based on the Cassini spacecraft design.

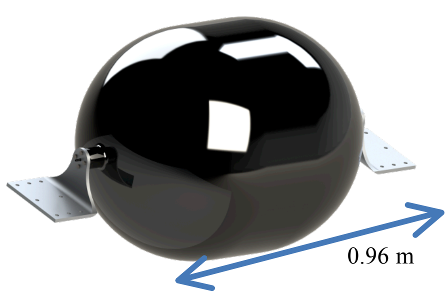

Fig. 9. This is a rendered image of the Helium Backfill Tank Assembly.

The bipropellant system uses a helium backfill method to maintain a pressure of 689.5 kPa (100 psi) in the oxidizer and propellant tanks. The helium pressurization assembly's main pressure vessel, purchased from Lincoln Composites, utilizes a composite overwrapped Ti 6AI-4V shell as shown in Fig. 9. At launch this tank holds 8.6 kg of helium at a pressure of 23.7 MPa (3440 psi). Downstream of this tank the helium passes through the redundant Pressure Control Assemblies (PCA1 and PCA2) which regulate pressure and flow to the oxidizer and bipropellant tanks. The helium then forces out oxidizer (nitrogen tetroxide) and propellant (monomethylhydrazine) into the Propellant Isolation Assembly (PIA). After passing through the PIA the propellant and oxidizer meet in the rocket engines. Each engine is mounted to a gimbal that allows for rotations of 34 degrees off of its nominal position. This rocket engine/gimbal assembly is shown in Fig. 10.

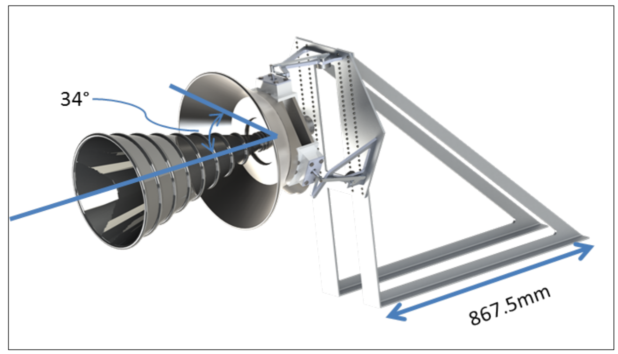

Fig. 10. This rendered image demonstrates the gimballing functionality of the orbiter's gimbal/booster assembly.

4.3 Orbiter Monopropellant System

The monopropellant system is used to perform attitude adjustment maneuvers in conjunction with 100 N-m-s reaction wheels purchased from Honeywell. This propellant system uses a blowdown approach. Initially the monopropellant tank will be pressurized to 2.34 MPa (339.39 psi) and as propellant is used the pressure in the tank decreases; in order to increase the useful life of the monopropellant system, a single-time helium recharge tank is used to repressurize the propellant tank when the pressure drops below a propellant efficient level (approximately when 33 kg of the hydrazine has been used). The helium recharge tank is purchased from Ardé and is constructed of 6AI-4V Titanium with a volume of 0.0085 m^3 (0.3 ft^3).

Moving downstream, the next major component is the monopropellant tank assembly (MTA). The MTA is a 6AI-4V Titanium sphere with an AF-E-332 ATK standard diaphragm. This tank's maximum operating pressure is designed to be 2.90 MPa (420 psi) when at a temperature of 45° C. Advantages of the AF-E-332 ATK standard diaphragm inside the tank include increasing the damping of the diaphragm on fluid motion and limiting the shifting the center of gravity of the propellant. Such characteristics lead to repeatability of propellant expulsion. The MTA is designed to hold the 132 kg of hydrazine (N2H4) required to complete the mission.

Downstream of the MTA, the fuel line breaks into two separate branches, A and B. Branch B exists for redundancy. Each branch has a filter (to prevent the flow of potential debris due to the use of pyro-valves from damaging the thrusters) followed by a latching valve with an inlet filter. Both of these units are purchased from Moog. When a given latch filter is opened, propellant will be allowed to flow to the single seat thruster valve assemblies with inlet filters. These 1 N "thruster clusters" are fixed to the far corners of the orbiter payload deck on the top and bottom surface to allow control and redundancy of pitch, roll, and yaw. Eight of these thruster clusters will be utilized (four fixed to the top of the payload deck and four on the bottom).

Before concluding the section on the orbiter propulsion systems, it should be noted that all of these sub-assemblies are connected via 6AI-4V Titanium piping and are fixed together via welding to minimize the opportunity for leaking failures. Resistance heating elements and insulation is wrapped around piping and tanks to prevent the propellant from freezing and causing pipe burst failure.

4.4 Lander Monopropellant System

The propulsion system for the Lander is similar to that of the orbiter; however, due to space and weight restrictions the monopropellant and helium are distributed among many tanks instead of one equivalent tank. Fig. 11 shows these tanks and their respective locations on the lander. The lander uses twelve, 195 mm (7.70 in) diameter, Ti 6AI-4V, spherical, monopropellant tanks to store 32.3 kg of hydrazine. Like the orbiter, each tank is initially pressurized with helium to 2.34 MPa (339.39 psi). Similarly, the lander utilizes a blowdown monopropellant method; however, the lander helium recharge tanks proportionally carry twice as much helium as the orbiter's helium recharge tank (the lander carries 0.196 kg of He). The additional helium is carried so that the monopropellant system is able to recharge the propellant tanks twice during the descent, thus providing longer periods of time that the lander is descending with full thrust capacity. The descent rate is controlled by feeding the monopropellant to four 22 N thrusters while controlling attitude with 16 pairs of 1 N thrusters.

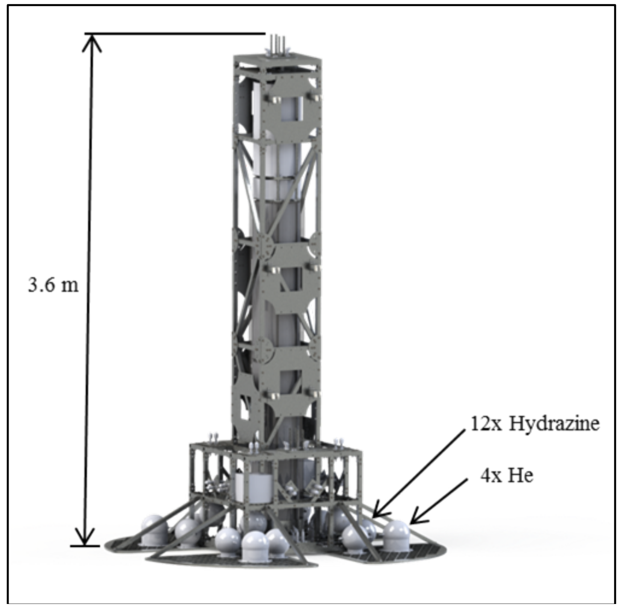

Fig. 11. This figure shows a side view of the lander and indicates the positions of the helium and monopropellant tanks.

5. ORBITER SCIENCE INSTRUMENTS

Each science instrument on the orbiter will be custom designed to maximize the science accomplishments. Five main science instruments flown on similar missions were picked to satisfy these science goals.

The high resolution camera will be used to map the surface of Enceladus in order to learn more about the geological history of the moon and also the nature and composition of the moon [12]. A UV-IR imaging spectrometer will be used to make spectroscopic measurements of the structure and the composition of the atmosphere of Enceladus [13]. The radar subsystem on the orbiter is used make observations using imaging, altimetry, backscatter, and radiometry [5]. This mission uses a Dual Technique Magnetometer (MAG) to determine the magnetic fields of the moon [14]. Finally, the orbiter is equipped with an Ion and Neutral Mass Spectrometer (INMS) that is used to determine the composition and structure of positive and neutral particles in the atmosphere of Enceladus [15].

Each science instrument is within a mass range and given a corresponding dimensional "envelope". The ICEPLUME mission utilizes a universal mounting system to allow the greatest versatility of possible science instrument configurations. Each of the science instruments is listed in Table 4, along with a table of design guidelines, Table 5.

Table 4: Science Instruments for Orbiter

	Mass Allowance [kg]	Power Allowance [W]	Similar to
High resolution camera	60	60	Cassini
UV-IR imaging spectroscopy	18	12	Cassini
Ion and Neutral Mass Spectrometer	10	28	Cassini
Radar or laser altimeter	42	109	Cassini

Table 5: Science Instrument Design Guidelines

	Up to 25kg	Up to 50kg	Up to 75kg
Power	< 25 W at peak	< 50 W at peak	< 75 W at peak
Data Rate	< 10 Mbps	< 10 Mbps	< 10 Mbps
Electrical Ground	The instrument should be electrically grounded to a single point on the spacecraft.	The instrument should be electrically grounded to a single point on the spacecraft.	The instrument should be electrically grounded to a single point on the spacecraft.
Thermal to Spacecraft	Conductive < 4W Radiative < 3W	Conductive <4W Radiative <3W	Conductive <4W Radiative <3W
Volume	< 200 x 350 x 600 mm	< 300 x 400 x 700 mm	< 300 x 500 x 730 mm
Center of Mass	< 3.5 cm radially from z-axis < 10 cm above interface plane along z-axis	< 4 cm radially from z-axis < 15 cm above interface plane along z-axis	< 5 cm radially from z-axis < 15 cm above interface plane along z-axis

In Table 5, the 50kg mass range is based on the Common Instrument Interface Project document; everything else is based on that architecture [16]. The spacecraft provider is responsible for the mounting interface and the specified power and data available. The instrument provider is responsible for kinematic mounting and degrees of freedom. Instruments should be self-contained assemblies with on-board electronics and thermal control. Instruments must fit within the above constraints for the appropriate mass margin.

6. ORBITER STRUCTURE

A flat payload deck is used for the main spacecraft structure. The payload deck consists of an isogrid structure with two face sheets on either side of the isogrid. Much like the aeroshell structure, the composite face sheets and the isogrid structure are made of graphite polycyanate. The basis of this payload deck design comes from a proposed NASA mission to Neptune [9]. The isogrid design is well suited for mounting components and has sufficient space to run cables and piping to science instruments and other subsystems [9]. The model of the payload deck is shown below in Fig. 12 with cutouts for the lander and propellant tanks. Fig. 13 shows the underside of the deck with the interlocking support system.

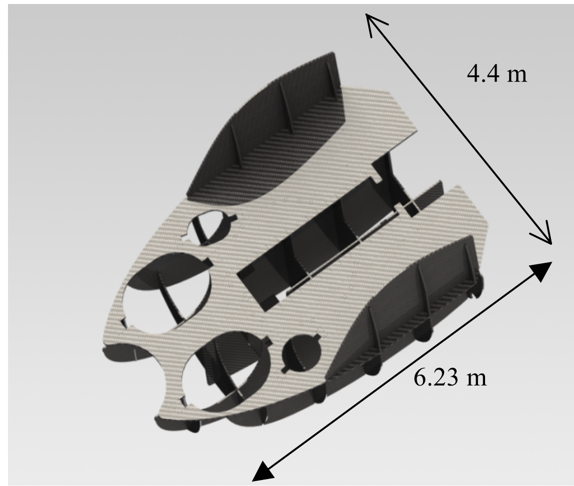

Fig. 12. The payload deck isogrid is covered by two face sheets.

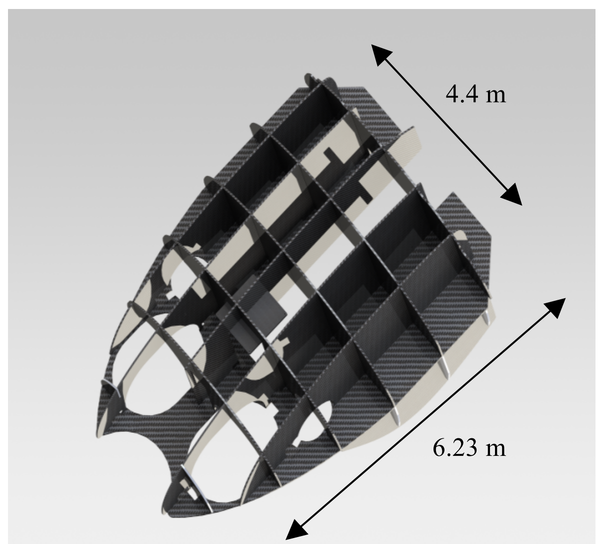

Fig. 13. Lateral and longitudinal ribs support the payload deck.

The High Gain Antenna requires an additional support structure attached to the main payload deck. This structure is designed to fit within the volume of the aeroshell and allow the probe and lander to fit underneath the High Gain Antenna. This structure is made of 6061-T6 aluminum I-beams and angle brackets with 7075-T73 aluminum sheet. The top of the structure is attached using solid countersunk rivets in order to have a flat surface for the antenna to mount. The rest of the structure uses standard round head solid rivets. All the rivets will be 7075-T73 aluminum.

7. LANDER

Once the orbiter has collected the necessary information to determine a safe landing zone, the lander is deployed from the orbiter towards the surface of Enceladus. Attitude and descent thrusters guide the lander to the surface, where it deploys the exploration probe.

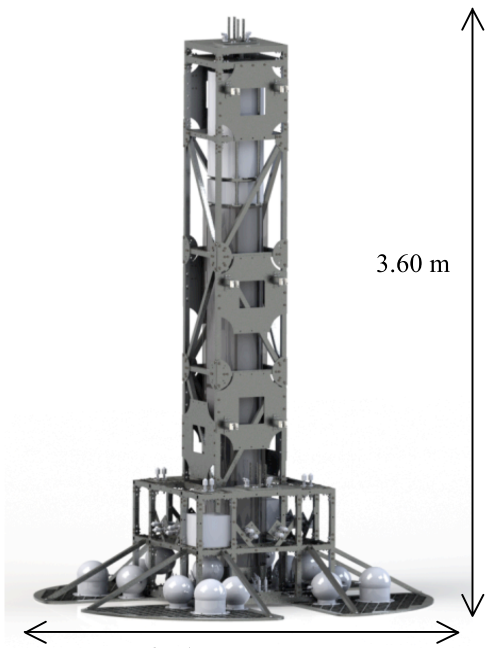

Fig. 14. The lander attaches to the orbiter via deployment attachments at the top.

The lander fits in the back of the orbiter, with the top probe-holding section sliding into the orbiter along its minor axis of rotation. The lander is attached to the orbiter through pyronuts, which compress two steel springs made of 302 stainless steel per AMS 5516. Each spring is compressed 7 cm, and the spring constant is 25-30 N/m. This spring system is redundant following NASA-STD-5017. The pyronuts used are a simplified version of those used on the aeroshell. Similar to Cassini-Huygens, the lander is deployed by exploding the pyronuts to break the hard connection to the orbiter, which allows the spring system to expand and push out the lander from the back of the orbiter. The lander contains 12 linear bearings which run along four 12 mm diameter, 2.54 m long guide rails attached to the orbiter deck. Given this push, the lander runs out along these rails and has a lower orbital velocity (less than 0.13 km/s) than the orbiter causing it to head toward the surface.

The payload bay located just above the feet and centered about the vertical axis of the lander, houses attitude and descent thrusters as well as science instruments. Sixteen groups of redundant attitude thruster cluster sets are located on each of the six faces of the payload bay. The four descent thrusters are located at the bottom of the payload bay. Each attitude thruster cluster contains two thrusters, each with 1 N of thrust force capacity. Both of these thruster types are used to control the lander during descent. Four 22 N descent thrusters are used to decelerate the lander.

All of the propellant is housed in tanks which are attached to the feet near the base of the lander. The mass of the lander is 550 kg, and given initial orbital velocities and the gravitational field of Enceladus, the lander contains 32.3 kg of propellant, which is 8.3 kg greater than estimates for safe fuel requirements. The science instruments are also located in the payload bay and include a descent camera, accelerometer, tiltmeter, seismometer, radar, and low gain antenna. The antenna will be used to communicate with the orbiter. The accelerometer, tiltmeter, and radar will be used to help control the lander during descent. The seismometer will measure surface activity after landing.

Enceladus has a trace atmosphere, with a surface pressure estimated at 1 x 10^{-11} bar. To determine whether or not some form of an aeroshell is needed on the lander, energy calculations were performed. These calculations show a flux value of 2 x 10^{-8} W/cm^2, many orders of magnitude lower than that of the aeroshell with the lowest heat flux value that has ever flown (24 W/cm^2). This referenced aeroshell was flown to Mars at a velocity of 4 km/s (20 times faster than this plan) and into a significant atmosphere. These factors lead us to the conclusion that no separate aeroshell is necessary for the lander.

Four quarter-circular feet of 1.00 m radius provide the platform for the lander to land and stand on. These feet are designed for the lander to land in different environments. Due to a landing target in the southern hemisphere where the continuously active plumes are located, there is a strong chance of landing on an unknown depth of powdered snow. These large feet provide a “snow-shoe” effect to prevent the lander from falling through the snow and burying the communications systems and science instruments. The four feet also incorporate an isogrid pattern to reduce its weight and to minimize the bending stresses on the feet due to large launch forces.

In case of a landing on a solid ice surface, enough thruster propellant is provided to slow the lander to zero vertical velocity for a soft landing. This soft landing is aided by the low gravity on Enceladus, which is about 1/100th of that on Earth (0.113 m/s^2).

Protruding from the payload is the container for the probe. The probe is internally fixed to the lander through the use of exploding bolts similar to the ones used by the lander for lander deployment from the orbiter. After landing on Enceladus, the probe is dropped from the lander through an open hole in the bottom of the payload bay by firing the pyronuts.

The structure is comprised of aluminum 7075-T73 and 7075-T7351, per ASTM B209M for sheets and plates and ASTM B221M for bars and rods. The structure is mainly comprised of interconnected square hollow bars made of this aluminum, with a 30 mm by 30 mm cross section and wall thickness of 4 mm. These materials were chosen because of their high strength to weight ratios and stress-corrosion resistance. Variously sized hexagonal bolts per MA3347, self-locking nuts per MA4177, and helical coil inserts per MA3329 connect the parts.

8. PROBE

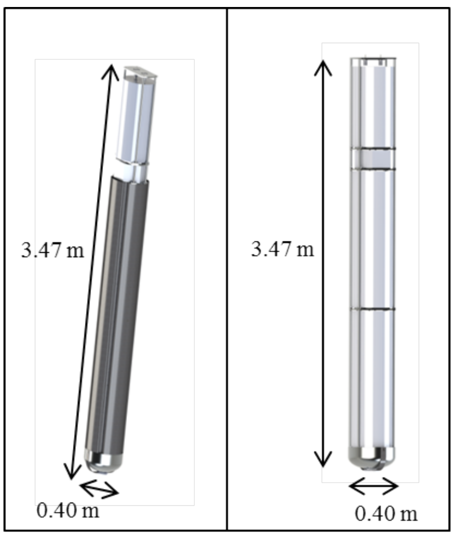

Fig. 15. The rendered image of the probe on the left shows the probe covered; the image on the right shows it uncovered.

After being dropped from the lander, the duel RTG-powered probe will melt through the glacial surface of Enceladus, on its way toward an underground ocean. Water jets and pumps push the melted ice water to maneuver the probe down through the ice. It also uses the jets to push warm water forward to melt the ice faster and to control its direction.

The probe is in the shape of a long bullet, with a length of 3.47 m and an outer diameter of 0.404 m. The rounded front end contains the aforementioned jets, along with an accelerometer and tiltmeter to determine its velocity and orientation. On top of these jets are the two General Purpose Heat Source Radioisotope Thermalelectric Generators (GPHS-RTGs). The GPHS-RTGs power the science instruments on board with 100 W-electric while providing enough excess heat through 8600 W-thermal power to melt through the ice. Each GPHS-RTG is 1.14 m long with an outer diameter of 0.35 m. A nuclear reactor was considered for this purpose, but no reactors are currently available which are light enough for this mission.

Above the GPHS-RTGs is a 0.145 m long cavity reserved for science instruments. These instruments include a descent camera to take images, a geochemistry package and water-volatile experiment to make chemical observations, a thermal/heat flow sensing package to measure thermal properties of the moon, a permittivity module and magnetometer for magnetic field measurements, a mineralogy-astrobiology camera for glacial compositions and astrobiology detections, and an x-ray fluorescence spectrometer for astrobiology detection as well. All of these instruments have flown in previous missions and will be constructed specifically for this mission. Specific volume and mounting specifications are given to each contractor for all of the science instruments used in ICEPLUME to ensure compatibility with the design. Due to the instruments need to access the environment outside the probe, everything above the GPHS-RTGs is uncovered.

Above the science instruments is a 0.685 m tall cavity for the tether bay. In the tether bay, a tether is unspooled from the back end of the probe, while being connected at the other end to the lander. The tether is not for physical support, but for data relay. The ESA/SCC 3901 019 tether is 1.98 mm in diameter and 16 km long. It operates in temperatures between -200 and 200 degrees Celsius. The length of this tether limits the depth the probe can reach if the lander and probe live beyond the minimum lifetime stated in the PDS, which frequently occurs in space missions. The PDS plans a 1.5 year minimum lifetime of the lander and probe, and in that time the probe will reach 10.5 km of depth. If no ocean is reached, the probe will end up melting a local pool of water for the probe to float in since the buoyant force is greater than the weight of the probe on Enceladus. If an ocean is found, the jets can be used for further exploration underwater.

The final section of the probe is the separation mechanism between the probe and lander. As mentioned previously, pyronuts are used for separation, but gravity pulls the probe out of the lander. Two 6.4 mm thick crushable honeycomb panels are used at the leading edge of the probe to provide a safer and softer impact with the surface when deployed.

The probe and GPHS-RTGs need to operate in temperatures as cold as -198 degrees Celsius (surface temperature) and as warm as freezing (0 degrees Celsius) or higher if an ocean is found. Calculations were made to determine what heat output is needed to sufficiently melt the surrounding ice to move the probe 10.5 km in 1.5 years. The outer diameter is the major factor in limiting the speed of the probe, and the outer diameter is most heavily influenced by the GPHS-RTG size. The glacial surface could be as thick as 48 km, but scientists have shown it is thinner near the south pole (where the probe will land) and thicknesses at specific locations are unknown. Also, the probe may travel through and melt frozen ocean water that is pushed up towards the surface.

The main structure of the probe is comprised of aluminum 7075-T73 and 7075-T7351, per ASTM B209M and ASTM B221M. Carbon Fiber – Epoxy T-300 3k/934 Plain Weave Fabric is used to cover the bottom portion of the probe, but aluminum 7075-T73 is used on the nose covering for increased protection and strength. These materials were chosen because of their high strength to weight ratios and stress-corrosion resistance. Each section of the probe is mounted onto its own aluminum mounting plate of 0.175 m side length and 1 cm thickness. Each mounting plate is attached to a support plate of 2 cm thickness which connects to the column rods. Four sets of four 2 cm diameter column rods surround each major section of the probe (except the two end sections) and attach each section to each other. Variously sized hexagonal bolts per MA3347, self-locking nuts per MA4177, and helical coil inserts per MA3329 connect the parts together.

9. PROJECT COST

ICEPLUME is an interplanetary mission, with one surface landing unit. It is designated as a Flagship Program, meaning it is determined to be in the most expensive class of missions. Only one launch is required and the total spacecraft mass is estimated at 7559 kg. Given these factors and the high degree of difficulty associated with the multiple components, the cost of ICEPLUME is estimated at $10 billion.

10. CONCLUSION

The ICEPLUME mission utilizes a variety of innovative technologies to fulfill the identified science goals. The architecture consists of an orbiter, lander, and ice-penetrating probe that will perform scientific observations and measurements around and on the Enceladan surface. The entire system will be encapsulated in an aeroshell for an aero-gravity assist maneuver through Titan's atmosphere. Solar electric propulsion coupled with gravity assists is the primary method of propulsion to escape Earth's sphere of influence and reach the Saturnian sphere of influence. The details outlined in this report show that a mission of this sort is not only feasible, but also practical in the search for astrobiology in the galaxy.

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