Nano Icy Moons Propellant Harvester Final Report

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1.0 EXECUTIVE SUMMARY

As one of just a few bodies identified in the solar system with a liquid ocean, Europa has become a top priority in the search for life outside of Earth. However, cost estimates for exploring Europa have been prohibitively expensive, with estimates of a NASA Flagship class orbiter and lander approaching $5B. ExoTerra's NIMPH offers an affordable solution that can not only land, but return a sample from the surface to Earth. NIMPH combines solar electric propulsion (SEP) technologies being developed for the asteroid redirect mission and microsatellite electronics to reduce the cost of a full sample return mission below $500M.

A key to achieving this order-of-magnitude cost reduction is minimizing the initial mass of the system. The cost of any mission is directly proportional to its mass. By keeping the mission within the constraints of an Atlas V 551 launch vehicle versus an SLS, we can significantly reduce launch costs. To achieve this we reduce the landed mass of the sample return lander, which is the largest multiplier of mission mass, and shrink propellant mass through high-efficiency SEP and gravity assists.

The NIMPH project's first step in reducing landed mass focuses on development of a micro-In Situ Resource Utilization (µISRU) system. ISRU allows us to minimize landed mass of a sample return mission by converting local ice into propellants. The project reduces the ISRU system to a CubeSat-scale package that weighs just 1.74 kg and consumes just 242 W of power. We estimate that use of this ISRU vs. an identical micro-lander without ISRU reduces fuel mass by 45 kg. As the dry mass of the lander grows for larger missions, these savings scale exponentially.

Taking full advantage of the µISRU system requires the development of a micro LOx/LH_2 engine. The micro-LOx-LH_2 engine is tailored for the mission by scaling it to match the scale of the micro-lander and the low gravity of the target moon. We also tailor the engine for a near-stoichiometric mixture ratio of 7.5. Most high-performance LOx/LH_2 engines inject extra LH_2 to lower the average molecular weight of the exhaust, which improves Isp. However, this extra LH_2 requires additional power and processing time on the surface for the ISRU to create. This increases mission cost, and on missions within high radiation environments such as Europa, increases radiation shielding mass. The resulting engine weighs just 1.36 kg and produces 71.5 N of thrust at 364 s Isp.

Finally, the mission reduces landed mass by taking advantage of the SEP module's solar power to beam energy to the surface using a collimated laser. This allows us to replace an ~45 kg MMRTG with a 2.5 kg resonant array.

By using the combination of µISRU, a µLOx/LH_2 engine, and beamed power, we reduce the initial mass of the lander to just 51.5 kg. When combined with an SEP module to ferry the lander to Europa the initial mission mass is just 6397 kg - low enough to be placed on an Earth escape trajectory using an Atlas V 551 launch vehicle. By comparison, we estimate a duplicate lander using an MMRTG and semi-storable propellants such as LOx/methane would result in an order of magnitude increase in initial lander mass to 445 kg. Attempting to perform the trajectory with a 450 s LOx/LH_2 engine would increase initial mass to ~135,000 kg. Using an Atlas V $/kg rate to Earth escape value of $27.7k/kg, just the launch savings are over $3.5B.

2.0 SYSTEM DESIGN

2.1 MISSION OVERVIEW

The Nano-Icy Moons Propellant Harvester (NIMPH) project focuses on the development of a mission architecture which enables a Europa sample return while drastically reducing the cost versus traditional orbiter/lander architectures. After Earth and Mars, the liquid oceans beneath Europa's icy surface are arguably the most likely spot in the solar system to find life. Returning a sample from Europa will give scientists an opportunity to study the raw materials for signs of biological activity in ways a sensor operating remotely cannot match.

Traditionally, the cost of a mission is directly proportional to the initial mass of the satellite. Initial mass directly impacts the energy, and thus cost, needed to propel the satellite away from Earth. This cost often increases as a step function - as mass growth drives a mission from the capabilities of one launch vehicle to another, the cost steps with launch vehicle. For instance, large cost increases incur in a move from a Falcon 9 to an Atlas V to an SLS. The large Europa Clipper mission is currently scheduled to fly on an SLS[1], which is estimated to cost $1B[2] per launch. A key to reducing mission cost by an order of magnitude is stepping the mission down from SLS to a $180M Atlas V 551[3]. Based on the Atlas User's Guide we must keep the initial mission mass below 6500 kg to enable a launch to Earth escape.

While we could launch into Earth orbit and spiral out, we select Earth escape to avoid the operations, radiation exposure, eclipse cycling and time associated with the spiral out from Earth with our SEP system. As shown in Figure 2-1, to reduce propellant and development costs, we reuse a high efficiency solar electric propulsion (SEP) module being developed for NASA's Asteroid Redirect Mission. ExoTerra was one of 4 teams commissioned by NASA to study

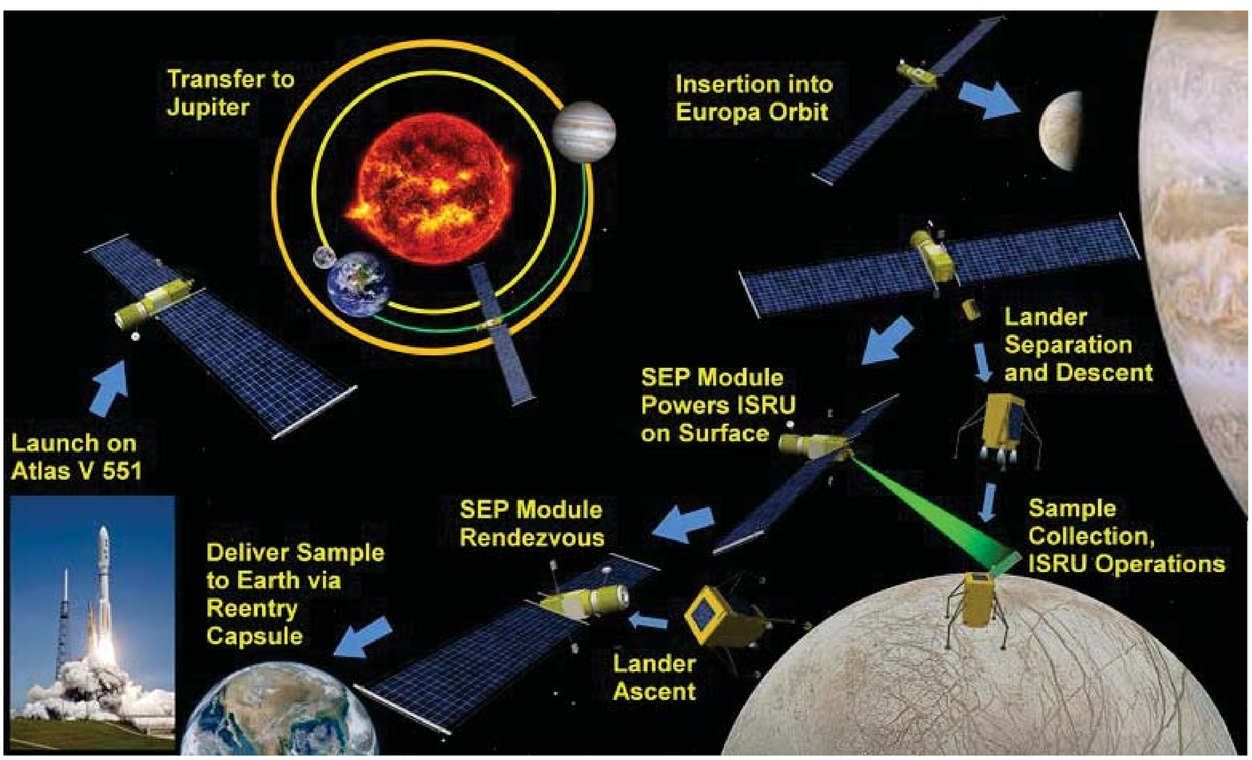

Figure 2-1: Design Reference Mission - ExoTerra's DRM Accomplishes a Europa Sample Return at 10% of Current Cost Estimates of the Europa Clipper

using commercial systems to perform the Asteroid Redirect Mission and to develop a conceptual ARM spacecraft design. ExoTerra's concept provided 57 kW of power at beginning of life (BOL) and had up to 3600 kg of xenon capacity. Hall thruster Isp reached 3150 s for transport of the asteroid. Initial dry mass (excluding the asteroid capture mechanism) was 2370 kg. In order to support potential commercial uses, ExoTerra's concept was also designed to be both modular and reusable, minimizing the cost of engineering and manufacturing for later missions. The SEP module connected to an externally mounted tank for refueling on one end, while the payload segment on the other end could be swapped out for different missions using a docking adapter. NASA implemented the modular concept into their eventual RFP, though the extent of the adoption will depend on the vendor ultimately selected. For the sake of the study, we have assumed that we will build-to-print a new SEP module for the mission based on the ARM design. Based on ExoTerra's ARM study, recurring cost for the 2nd module is <$170M.

The mission uses the SEP system to transfer from Earth to Jupiter orbit insertion and for the Jupiter to Earth return segments. Based on initial trajectory analysis (detailed in Section 2.2), 6.7 km/s of ΔV is required to transit from Earth to Jupiter and an additional 9.2 km/s are allocated for return. The analysis assumes a series of gravity assists to minimize the ΔV.

While in the Jovian system there is insufficient sunlight to drive the EP system at full power. Instead we carry along a 350 s LOx/Methane stage for performing maneuvers within the Jovian system, and for Europa orbit insertion and departure. The total amount of ΔV required is a function of launch date and the location of the various moons at the time of arrival. Our trajectory analysis has shown we can reduce the ΔV within the Jovian system for flybys, Europa insertion, maintenance and departure to as low as 674 m/s using gravity assists. Given the uncertainty in launch date, we allocated a total of 1250 m/s for adjusting trajectory within the system for gravity assists from the Jovian moons. Total mass of the stage is 1991 kg.

While operating within the high radiation environment of Jupiter and its moons, we minimize total ionizing dose on the lander's electronics by storing it within a vault on the SEP module. To eliminate boiloff of lander propellants on the outward journey the LOx and LH2 required for landing is stored as water. This water is converted to LOx/LH2 prior to deployment of the lander at Europa, using a duplicate ISRU system mounted on the SEP module.

Once in orbit around Europa, the lander is deployed. To minimize landed mass, we replace the traditional RTG with a laser power beaming system. As discussed in Section 2.3, up to 1800 W of power can be beamed to the surface from the orbiter while in Europa orbit, providing up to 478 W of electrical power to the lander. This replaces the 45 kg RTG power system with a 2.5 kg array system, significantly reducing landed mass and avoiding the cost and complexity of using a radioisotope power source. We have allocated 50 kg on the orbiter for the laser power-delivery system.

During landing, we use LIDAR to select an icy surface for landing, ensuring the lander feet will be in direct contact with ice. Once on the surface, the lander collects a 1 kg sample of ice. The ISRU system then begins to process ice for propellant to return to the orbiter by sublimating the ice under the landing feet. Total return propellant is 21.95 kg, including a 10% ΔV margin and 5% unused propellant margin. The ISRU system operates during "laser daylight", i.e.: while the orbiter is in view of the lander. Water is processed at rate of 8.3 mg/s; 242 W is required to sublimate and electrolyze the water and then liquefy the propellants. This results in a total surface time for the mission of 148 days. Total wet mass of the lander is 52.5 kg when deployed.

Table 2-1: System Mass Summary	
Item	Mass (kg)
SEP Module Dry Mass	2371.0
Collimated Laser System	50.0
Misc Orbiter Sensor Payload	25.0
Lander Canister	5.0
Sample Return Capsule	6.2
Lander	52.5
LOx/Methane Stage	1991.6
Xenon Propellant	2131.4
Total	6396.8

The landing legs are left behind at liftoff, as we assume the foot pads will have become bound to the ice as a result of the water collection process.

Lander mass is further minimized by leaving off rendezvous systems. After the lander reaches orbit, the SEP module will rendezvous with and then capture the non-cooperative lander. The SEP module then leaves the Jovian system and returns the sample to Earth, where the lander is delivered to Earth's surface inside a miniaturized IRVE-derived inflatable reentry capsule. We estimate that the capsule weighs approximately 20% of the dry lander, or 6.2 kg.

Allowing for 25 kg of additional scientific payload instruments on the orbiter gives a total orbiter dry mass of 2487.3 kg. Using the SEP system for 6.7 km/s to and 9.2 km/s from Jupiter and the LOx/Methane system for 1250 m/s within the Jupiter system, we arrive at a total initial mass of 6446 kg.

2.2 TRAJECTORY ANALYSIS

Solar Electric Propulsion (SEP) vehicles have a wide range of trajectory solutions for interplanetary missions. The design space is a function of the low thrust, propulsion hardware, time of flight (TOF) requirements, radiation minimization, sunlight exposure, gravity assist maneuvers, and launch date opportunities to name a few. At this stage in the design, we have calculated the following solution as an example demonstrating the feasibility of an interplanetary trajectory with the NIMPH system. This data will be used in future work as a starting point for

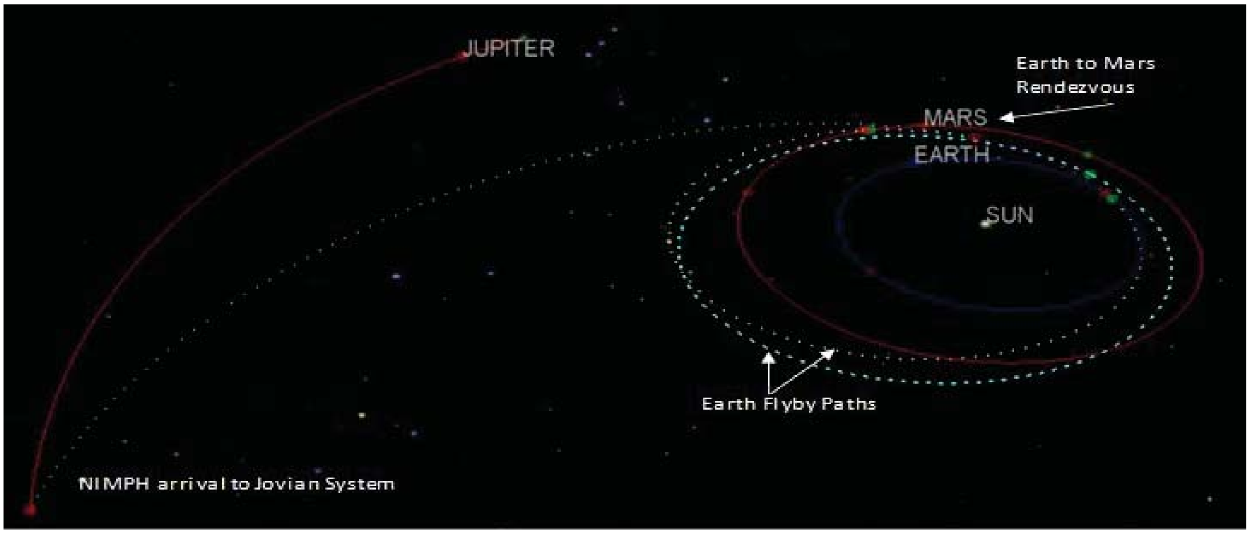

Figure 2-2: The MEEJ Gravity Assist sequence allows NIMPH to rendezvous with the Jupiter system in 6.9 years using 6.7 km/s ΔV from the propulsion system.

trade studies to determine several optimal solutions within the complex design space.

Based on the Earth/Jupiter synodic period, there is a direct trajectory from Earth to Jupiter every 13 months. However, to fit within an Atlas V 551, gravity assists must be used to minimize the ΔV that needs to be delivered from the propulsion system. There are many variations on the series of gravity assist paths. Multiple Venus flybys is often studied and may be desirable for a SEP system due to its proximity to the sun. But for that very same reason, additional thermal protection systems would be required, adding system weight, so we have opted to not use Venus in our initial study. Instead our path will include Mars and Earth only.

Using a patched conic approach and NASA's trajectory design tool, Copernicus, the analysis starts with a launch from an Atlas 551 resulting in a C3 of 31.1 km/s. NIMPH then proceeds on a Mars-Earth-Earth-Jupiter (MEEJ) path. The first gravity assist we encounter is from Mars 181 days after launch. For this solution, only 1.31 km/s were required from the propulsion system to achieve the rendezvous with Mars. Once we swing by Mars, a burn using 259 m/s ΔV was needed to continue with two flybys of Earth. After the second flyby of Earth, NIMPH has traveled a total of 4.5 years but now has the required energy to rendezvous with Jupiter. After a 2.3-year cruise to Jupiter the SEP system performs a 5.13 km/s maneuver for capture in to the Jupiter system. Figure 2-2 shows the overall path of this trajectory starting from Earth to rendezvous with Jupiter.

NIMPH arrives in the Jupiter system after 6.9 years having used only 6.7 km/s of ΔV from the propulsion system. We enter an elliptical orbit at 10.8 km/s and begin flybys of Jupiter's moons to begin decreasing our velocity for Europa orbit insertion. Figure 2-3 shows several revolutions around Callisto, Ganymede and Europa. The flybys give a reduction in ΔV ranging from 640 m/s to 1.08 km/s.

An important consideration in this phase of the trajectory is the amount of radiation NIMPH is experiencing. Therefore, in this solution, we do not perform flybys of Io to avoid the strong radiation exposure. We have also chosen not to include Callisto flybys for this example so as to reduce the flight time before Europa orbit insertion.

After 745 days including four flybys around Ganymede and another four flybys around Europa, we have decreased our energy enough to only need 261 m/s from the LOx/Methane system to perform the orbit insertion maneuver into a 500-km altitude orbit. Any propulsion ΔV required during the Ganymede and Europa flybys was shown to be negligible for this case. We estimate 113 m/s for orbit maintenance during the 170 days of lander operations. Once surface operations are complete and the lander has returned the ice sample to the SEP spacecraft

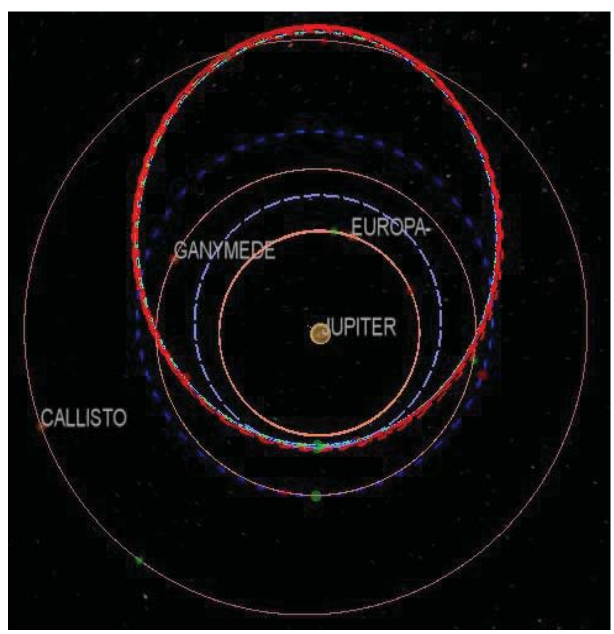

Figure 2-3: Multiple Gravity Assists around Ganymede and Europa result in a 261 m/s EOI.

Table 2-2: Propulsion ΔV and TOF Summary		
Earth Departure	**ΔV (km/s)**	**Time (days)**
Earth to Mars	1.31	181
Post Mars Flyby	.259	587
Earth to Earth Flyby (x2)	~	900
Earth to Jupiter	5.13	854
Total	6.7	2522 (6.9y)
Jovian System		
Ganymede Flyby (x4)	~	304.8
Europa Flyby (x4)	~	440
Europa Orbit Insertion	.261	1.5
Orbit Maintenance	.113	170
Europa Orbit Departure	.3	2.2
Total	.674	918.5
Earth Return		
Direct	9.2	378
Gravity Assist Option	3.4-5.8	>900
Total	9.2	378
Mission Total	16.6	3818.5 (10.5y)
Reserve	3.4	

in orbit, we'll need approximately 300 m/s to depart Europa and return to Jupiter orbit.

Departing Jupiter for a direct return to Earth will take the SEP system 9.2 km/s and a little over a year. If the ΔV budget allows this large of a burn, this may be desirable due to the scientific importance of the sample. However, we may need to perform gravity assists to return to Earth with a more reasonable propulsion ΔV requirement. If so, the ΔV requirement could be in the range of 3.4 – 5.8 km/s for one flyby of either Mars or Earth and even lower if multiple flybys were chosen. However, this will increase the time of flight by three to four years. Table 2-2 shows a summary of the ΔV and TOF for this data set. We also maintain ΔV margin for any additional deep space maneuvers for corrections due to launch date changes.

This solution is just one example. The flexibility offered from using a low thrust SEP system may result in more than one optimal solution for each launch date opportunity. As stated previously, once a target launch date is set, many trades need to be performed using a range of launch dates to optimize on the large design space while satisfying the constraints of ΔV, fuel and mission duration. The trades may include the following:

Gravity Assists - The number of GA's, and the planet sequence and timing of the maneuvers will be analyzed to minimize the overall time of flight and the time the spacecraft is in shadow while on the Earth departure and return phase of the mission. While in the Jupiter system, the numerous sequences for moon gravity assists will be considered for both the arrival and departing phase to optimize the TOF, ΔV requirements and radiation effects and Europa sample location.

Radiation Effects - The spacecraft will spend several years within the heavy magnetic field of Jupiter and minimizing those effects will be necessary to ensure the health of the spacecraft systems. Factors will include timing of the JOI, as well as choosing the optimal sequence during the Jupiter moons flyby phase to avoid the regimes of maximum radiation. Another possible consideration according to Astrobiology Magazine article, 'Hiding from Jupiter's Radiation' at astrobio.net, is that the sample site may need to be on the leading hemisphere of Europa in an area that has been protected from constant irradiation which would kill many organic molecules. These effects may all contribute to the optimal choice for landing site and in turn, the orbit for the SEP spacecraft.

Europa's Orbit Stability – The stability of the orbit around Europa is a function of the dynamics from the other Jovian moons as well as Jupiter's large gravitational field. Optimizing the orbit for minimal orbit maintenance maneuvers and/or finding stable manifold within the low Europa orbit range, will decrease the overall ΔV needed for the propulsion system.

2.3 POWER DELIVERY SYSTEM

At Jupiter, we anticipate that 2.03 kW of power will still be available to the orbiter due to the large arrays. 200 W of power is reserved for the orbiter, leaving 1.83 kW of power to support lander operations. To deliver this power from the spacecraft to the ground, we use a collimated laser system. The laser system is based on an nLight diode pumped laser developed under the DARPA SHEDs program[4]. The system starts with a 75% efficient fiber laser. This is then pumped with an 85% efficient diode laser to reach 1.83 kW. Using optics, the beam is collimated into a .35 m beam waist laser. Assuming a further 10% degradation for losses through the optics, we see a net conversion of 57.4% of the input electrical power to a 1052 W output beam.

Assuming a Gaussian beam, divergence of the beam is governed by the Raleigh Range. At a wavelength of 1030 nm and beam waist of .35 m, the collimated laser has a Raleigh Range of 3.74×10^5 km. At an orbit altitude of 500 km, the maximum distance between the satellite and lander is 1346 km when the satellite is on the horizon. Due to the high Raleigh Range, we see negligible beam divergence at this distance and the resulting spot size from the laser is .38 m^2. The flux at the lander is 2734 W/m^2, roughly 2x the flux of sunlight in Earth orbit. This concentrated power allows for a reduction in the size of the solar array on the surface.

A driving requirement for the system is the pointing accuracy of the laser. Most star trackers have accuracies on the order of 1 arcsecond. This error in attitude knowledge alone results in an error on the ground of 8 m, which could miss the lander altogether. To lock onto the lander we require a feedback loop. We envision the satellite will perform a scan with the beam as it comes over the horizon. Once the array is pinged, the lander sends a signal back to confirm the beam is on target. From there, the array can sense the position of the beam on the array. Feeding this information to the spacecraft allows the spacecraft to center the beam on the array. To achieve this fine precision requires a fine resolution on the spacecraft laser gimbal. Current high end gimbals such as the Moog ultrafine rotary actuator have a step angle of .001 deg, or 3.6 arcseconds – which is worse than the attitude knowledge. To further improve the resolution, we use the rotary actuator for coarse pointing and implement a secondary piezo-electric actuator along the edge of the collimated laser. The piezo allows for step sizes as small as 20 nm. At a radius of .35 m, we achieve an angular step size of .01 arcseconds, which results in a worst case step on the ground of .077 m. Since the spacecraft and Europa are moving, we must project forward where the array will be located when we point the array. We allocate an additional error of .01 arcseconds to account for errors in projections between feedback signals. This results in oversizing the array radius by .153 m to provide some margin for error.

Because we use a laser beam with a single frequency, we can tune the photovoltaic cells to convert the energy more efficiently than typical solar power conversion which much convert energy from a wide spectrum. Theoretical efficiencies are as high as 78%[5], though we assume a more conservative 60% conversion rate. In addition to conversion losses, we assume an additional 24% loss due to packing factor, diode losses, temperature losses, radiation degradation and manufacturing imperfections. This results in a total output power of 478 W at the surface.

Companies such as LaserMotive have demonstrated receiver specific power of 500 W/kg.[6] As another reference, ATK advertises a 150 W/kg solar array specific power (BOL) for their UltraFlex[7] arrays that are designed for operation in a gravity field. Adjusting the ATK value for a flux that is 2x that of solar power in Earth orbit and for 60% conversion efficiency vs. the 30% UltraFlex cells results in a specific power of 600 W/kg. Assuming 24% degradation results in 478 W/kg at Europa. For the actively powered section of the array, we estimate a mass of 1.05 kg. Since the array is oversized to .8 m^2 vs. the .38 m^2 requirement to account for pointing errors, total array mass is estimated at 2.18 kg.

2.4 EUROPA ORBIT OPTIMIZATION

The orbit selection for the host satellite plays a large role in mission optimization. Reducing the orbit altitude reduces the total ΔV needed for the lander, and the resulting propellant that must be produced. It also reduces the distance to the lander, improving pointing errors and reducing the size of the solar arrays. Conversely, a lower altitude decreases the total fraction of time the satellite is in view of the lander to provide power via laser. This increases the eclipse time and lander battery mass required, and increases the time required to process the propellant. A longer time on the surface in turn increases the radiation vault mass, and the longer mission duration adds to mission operations costs.

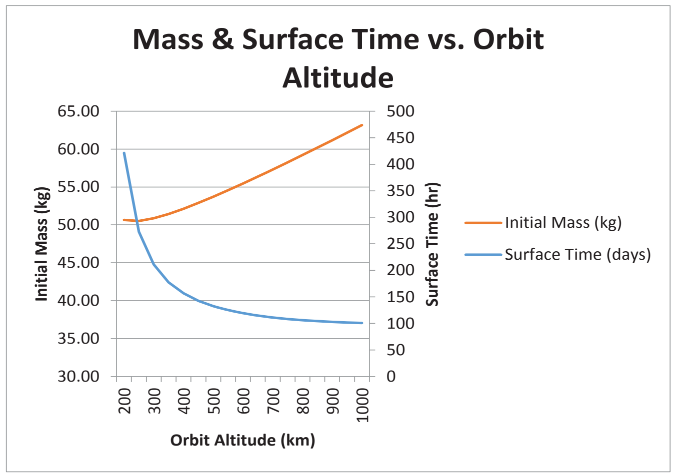

Figure 2-4: Orbit Altitude Optimization

ExoTerra has built a system optimization program to evaluate the impact of changing the orbit altitude. Our primary goal is to minimize the mission cost. These costs are assumed to be driven by the initial mass and operations costs. Figure 2-4 provides the initial mass and surface time as a function of orbit altitude. Based on the cost and performance of the Atlas V 551, our target launch vehicle, we find that the launch vehicle costs \$27.7k/kg to Earth escape velocity. Assuming we use the SEP Module, we also find that each kg of dry mass on the surface results in 6.1 kg growth to the mass at Earth escape. This results in an incremental cost of \$169k/kg on the surface. We have also estimated \$7,470/day to operate the mission. These values were used with the total surface time and initial mass to estimate the

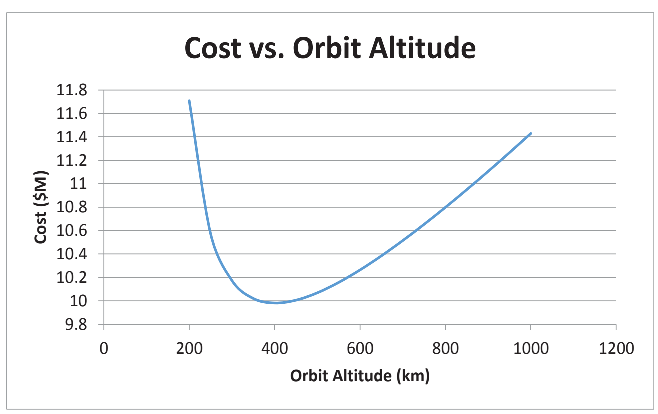

Figure 2-5: Relative Cost vs. Orbit Altitude

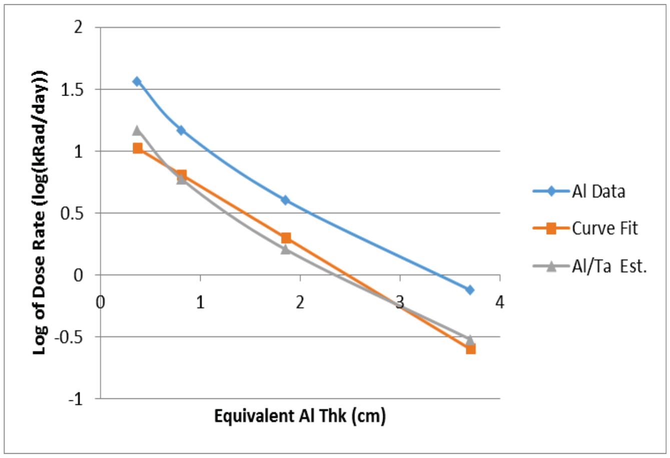

Figure 2-6: Radiation Dose v. Shield Thickness Approximation

mission cost for each orbit altitude. We find the minimum cost point at 400 km (see Figure 2-5).

However, since the mission time begins to grow quickly below 500 km, and the cost difference is small between the points, we have elected to use 500 km to keep our mission time on the surface shorter. This minimizes the amount of time available for Murphy's law to operate.

To calculate the vault mass we assumed that the electronics were manufactured with a radiation tolerance of 1 MRad. This is higher than typical cubesat electronics at 30-100 kRad, but within the realm of the industry. We derated the exposure limit to 800 krad to provide 25% margin. In addition, we assumed 1 krad of exposure during the journey and 8 krad of exposure during orbit capture maneuvers based on an estimate from M. Podzollo.[8]

To improve vault mass, we assume a combination aluminum/tantalum wall. Based on data from QinetiQ[9], use of Al/Ta shielding can reduce the total dose by 70-90% vs. straight aluminum. This is accomplished by taking advantage of the different densities to break up the radiation and block the different levels. The fraction varies with the shield thickness. We have conservatively estimated a 60% reduction for all thicknesses.

The data is roughly linear with the log of the dosage rate. To quickly estimate the required thickness as the surface time varied, we developed a curve fit of the Al/Ta estimated dose rate. This curve is also shown in Figure 2-6.

The vault is assumed to be 1U in size, resulting in 6 10 cm x 10 cm panels. We multiply this surface area by the required equivalent aluminum thickness to determine the total vault mass. At 500 km, this is 1.26 kg.

2.5 THRUSTER OPTIMIZATION

The thruster sizing is integral to the overall lander performance. As such, ExoTerra integrated the thruster optimization within our system design tool to gauge the impact of various variables on the system design. Key variables we explored included mixture ratio, Throat/Nozzle area ratio, and chamber pressure. We used the NASA Chemical Equilibrium with Applications (CEA) online tool to generate a database of performance data for the various options. This included varying mixture ratio from 4:1 to 8:1, area ratios of

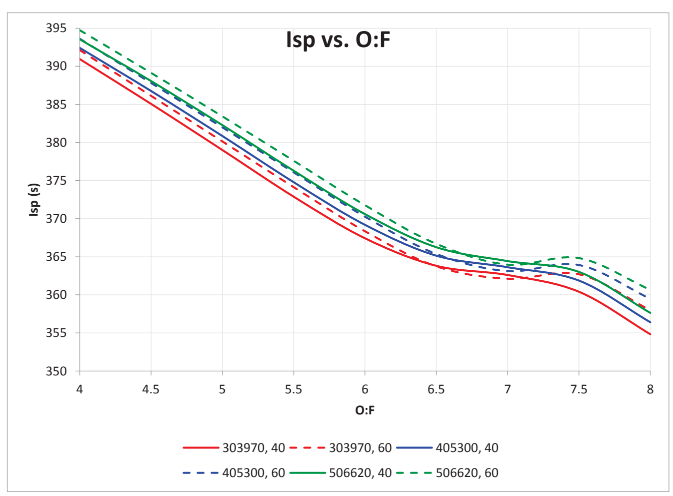

Figure 2-7: Isp v. Mixture Ratio

40 and 60, and chamber pressures of 304 kPa, 405 kPA and 506 kPa. The program provided data on thrust, Isp, fluid densities and temperatures that were used to evaluate the mission performance.

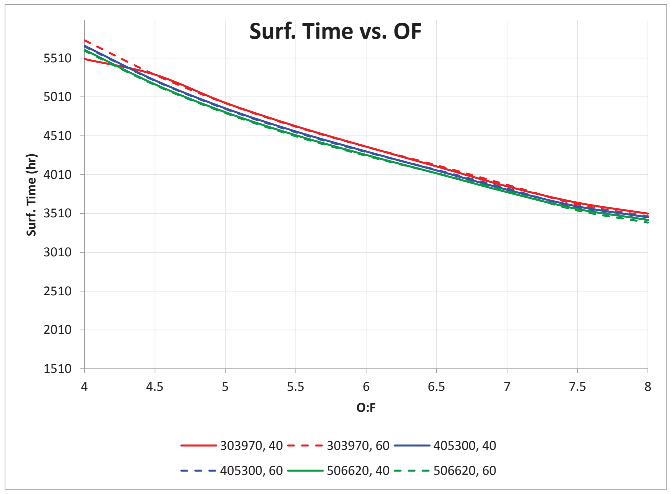

Figure 2-8: Surface Time v. Mixture Ratio

The mixture ratio had a direct impact on the performance of the engine, the mission time, the lander mass and radiation shielding. Typical LOx/LH2 engines such as the RL10 run with an oxidizer to fuel mass ratio of ~5.5:1 vs. a stoichiometric ratio of 8:1. By flowing extra hydrogen through the engine, it reduces the average molecular weight of the rocket exhaust. This lower molecular weight improves the Isp of the engine, reducing the total mass of propellant needed for launch. Figure 2-7 shows the decrease in Isp with increasing O:F ratio for each of the 6 combinations of area ratio and chamber pressure. While higher Isp is desired, this results in an increase in the time needed to process propellant since we need to process extra hydrogen, throwing away the associated oxygen. This is significant at Europa as any increase in mission time results in increased radiation exposure. ExoTerra's model accounts for this by increasing shielding mass as processing time increases to maintain a total ionizing dose of 1 MRad on the components within the vault. Figure 2-8 shows the processing time decreases as the O:F ratio approaches stoichiometric. We see that the propellant reductions from higher Isp never result in reductions in surface time since the time is driven by the total hydrogen need.

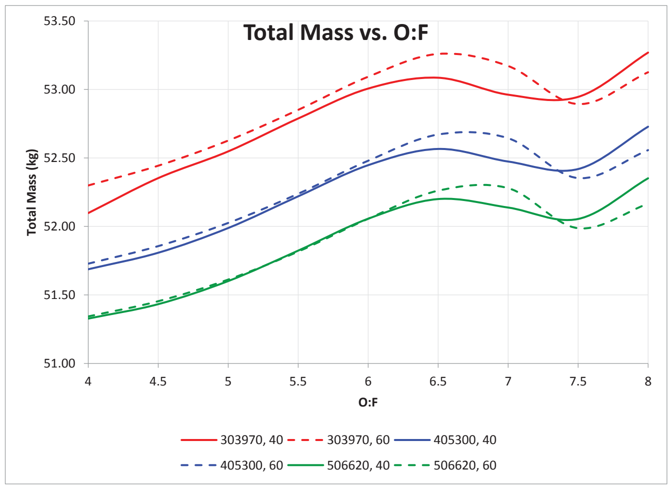

Figure 2-9: Initial Mass v. Mixture Ratio

To sort the conflicting performance variables, we evaluated each of the thruster design points within the system model to find the optimum design point for the mission. Figure 2-9 provides the total initial mass versus the O:F ratio. We see there is a local minimum at an O:F ratio of 7.5 at 506 kPa and a 60 area ratio.

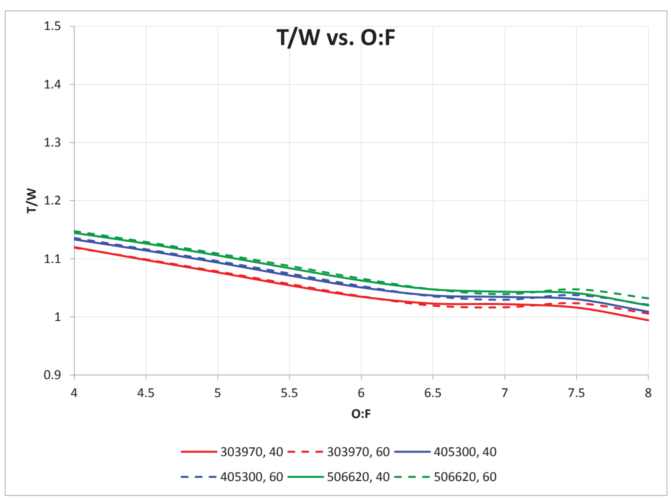

Figure 2-10: T/W v. Mixture Ratio

In addition to reductions in Isp, the thruster also has a reduced thrust as O:F ratio increases (see Figure 2-10). To ensure we maintain a positive Thrust/Weight ratio for liftoff, we check thrust to weight ratio vs O:F ratio as well. The weight calculation assumes local gravity of Ganymede vs. Europa to ensure the system could be used on any icy moon. We find that at 7.5:1 we have a positive thrust to weight ratio of ~1.05. We would prefer to be above 1.1 to reduce gravity losses, so we may adjust the final design for slightly higher thrust during Phase II work.

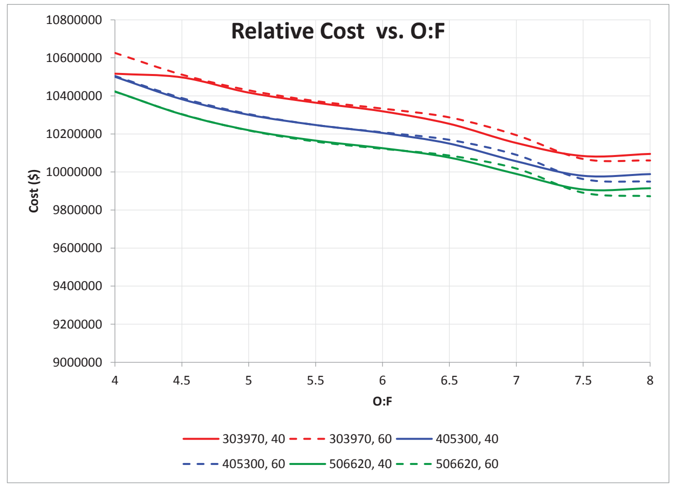

Figure 2-11: Relative Cost v. Mixture Ratio

Lastly, we evaluated relative cost vs. O:F ratio for each design point. We estimated the cost of operations as $7475/day which covers DSN costs plus a staff of 6 monitoring the satellite 8 hr/day. The cost of the mass was derived from the launch costs and gear ratio for the system. At a cost of $180M and capacity of 6500 kg to Earth Escape, we have an incremental cost of $27,700 per kg. Based on our model, each kg of dry mass on the surface results in 6.1 kg needed at Earth escape. Thus we estimate each kg on the surface of Europa cost $169,000. Using these rates, we find that the minimum cost is in the 7.5 to 8.0 O:F area of the graph shown in Figure 2-11. To maintain a positive T/W ratio, we select the ratio 7.5:1.

3.0 ISRU DESIGN

The ISRU system is composed of three branches: water collection, oxygen processing, and

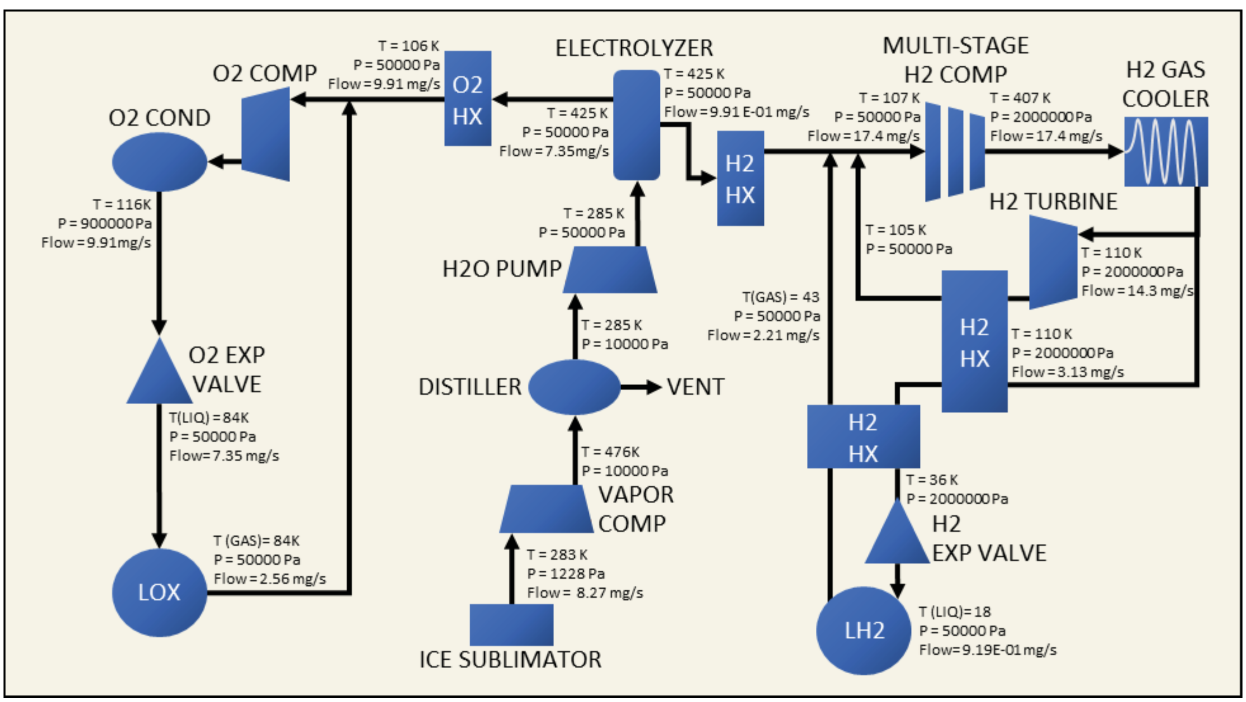

Figure 3-1: ISRU Block Diagram

hydrogen processing. Water harvesting starts with the sublimation of ice to vapor by the heaters embedded in each lander foot. The vapor is gathered by a compressor and converted to liquid in a distilling condenser. The liquid water is fed into the electrolyzer, where it is split into its constituent oxygen and hydrogen.

From there, the oxygen passes through a compressor, a condenser, and an expansion valve, which convert it to liquid form for storage in the oxidizer tank. Gaseous oxygen from tank boiloff and incomplete liquification feeds back into the compressor inlet for further processing. Hydrogen follows a similar sequence, using additional heat exchangers and a turbine to attain the very low temperatures needed to liquefy the fuel. Figure 3-1 is a block diagram of the system.

3.1 PERFORMANCE SUMMARY

The NIMPH In-Situ Resource Utilization (ISRU) system harvests water from the surface of Europa and converts it into hydrogen and oxygen propellants. Figure 3-2 shows the arrangement of the principal components within the lander. The ISRU system processes 8.27 mg/sec of water through an electrolyzer, producing 7.35 mg/sec of oxygen and 0.92 mg/sec of hydrogen. The

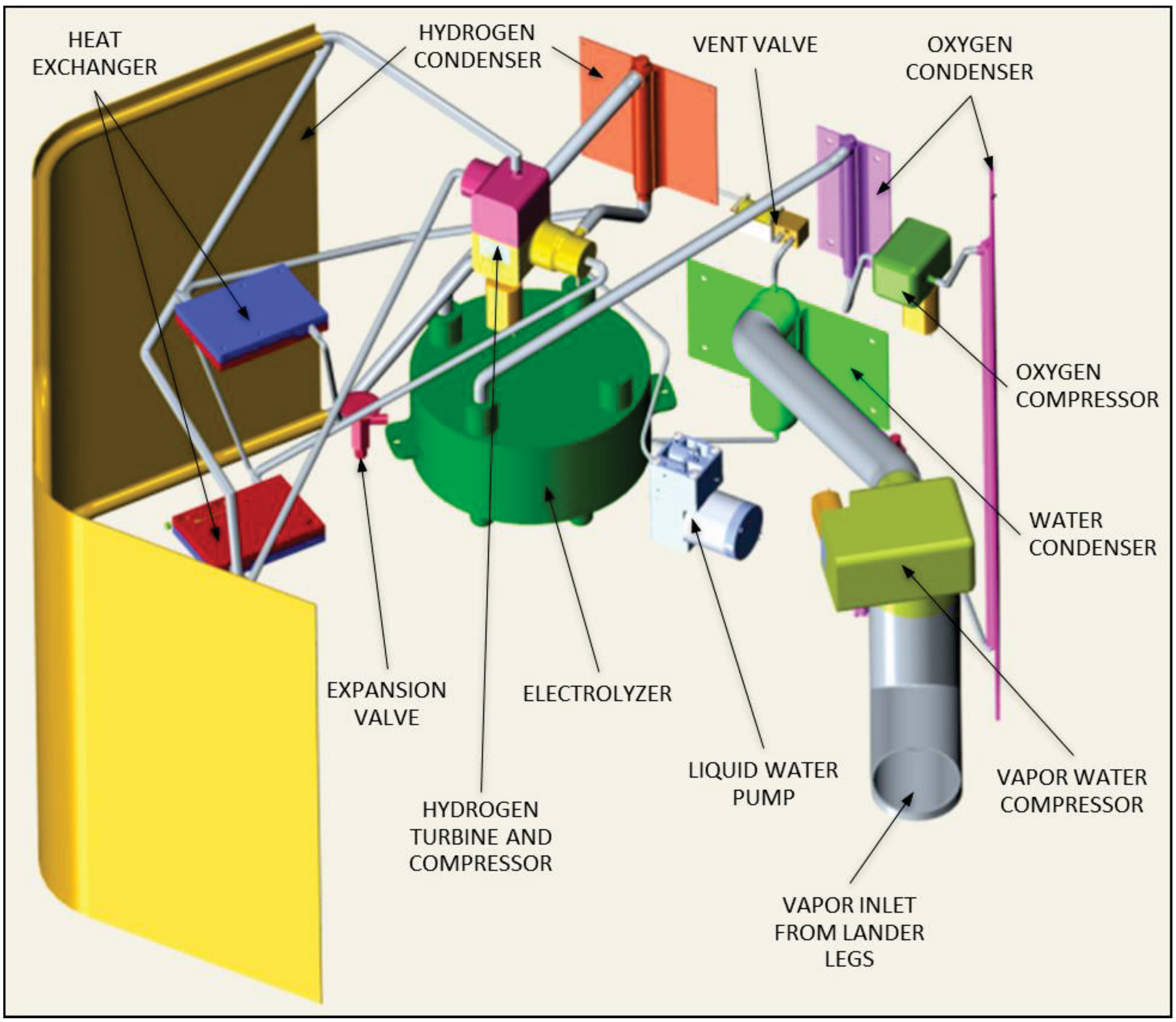

Figure 3-2: ISRU Water Processing System Layout

estimated mass of the entire system is 1.74 kg, with an estimated power consumption of 242 W. Table 3-1 provides a summary of the mass and power.

Table 3-1: μISRU Mass & Power Summary

Item	Mass (g)	Power (W)
Inlet	170	24.7
H2O Compressor	118	3
H2O Distiller	42	
H2O Pump	100	.1
Electrolyzer	400	154
O2 Radiative Cooler	16	
O2 Compressor	46	1.2
O2 Condensoer	24	
O2 Expansion Valve	2	
H2 Radiative Cooler	24	
H2 Compressor	114	72
H2 Condenser	70	
H2 Expansion Valve	2	
H2 Turbine	102	-12.4
H2 Hx	400	
Tubing	106	
Total	**1740**	**242.7**

3.2 WATER COLLECTION

The process for producing water from the icy surface of Europa begins at the feet of the lander.

The surface temperature of Europa never rises above 110 K, and the atmospheric pressure is a negligible 0.1 μPa (compared to 101.3 kPa on Earth). Under such conditions, ice on the surface will not melt into a liquid when heat is added; it will sublimate into a vapor. The lander takes advantage of this phase shift by embedding heating coils into the bottom of each lander foot, as shown in Figure 3-3. These coils heat the surface of the foot, sublimating the ice on which it rests. Using work performed by Andreas[10] regarding sublimation of ice in space, we find we can sublimate 8.7 mg/s by heating the ice to 211.5 K with 24.1 W of power. The ~ 1 Pa vapor passes through holes between the turns of the heating coils and into the hollow tubular lander leg. We heat the vapor to increase the pressure to 1228 Pa using .6 W of power.

By harvesting only vapor, the only potential contaminants will be sublimated gases captured with the vapor. All other impurities, such as dirt and rocks, remain on the surface, simplifying our collection and purification system. The heated footpads form tight seals with the surface of Europa as they sublimate ice and smooth the contact surface beneath them, improving compressor performance while minimizing vapor escape to the environment. Because the footpads will end up frozen into the icy surface, the lander jettisons the legs at takeoff.

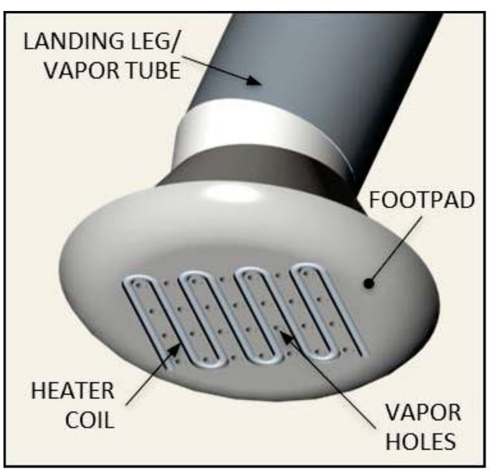

Figure 3-3: Lander Foot Detail

A 3 W, 118 g micro-compressor increases the pressure and temperature of the vapor to 10 kPa and 476 K and then passes it on to the condenser shown in Figure 3-4. The condenser uses the cold ambient temperature of Europa to cool the vapor. At 10 kPa and 285 K, the water condenses into a liquid. We then vent any other gasses, leaving 8.3 mg/s of water. Since the ambient temperature of Europa is so cold, an insulative material is placed between the condenser and the exterior of the lander to prevent freezing. Other gases collected from the surface, such as carbon dioxide, will not condense and will be vented from the top of the unit. The condenser mass is estimated at 42 g. Liquid water then passes through a pump to the electrolyzer,

3.3 ELECTROLYZER

An electrolyzer uses electrical current to separate liquid water into hydrogen and oxygen gases. The µISRU uses a Polymer Electrolyte Membrane (PEM) electrolyzer, which is compact, lightweight, and works at lower temperatures than those using other electrolysis methods (see Figure 3-5). The electrolyzer splits water into oxygen and hydrogen and draws them to separate sides of the system, automatically separating the gases. From here the gases are delivered to their respective liquification loops for further processing.

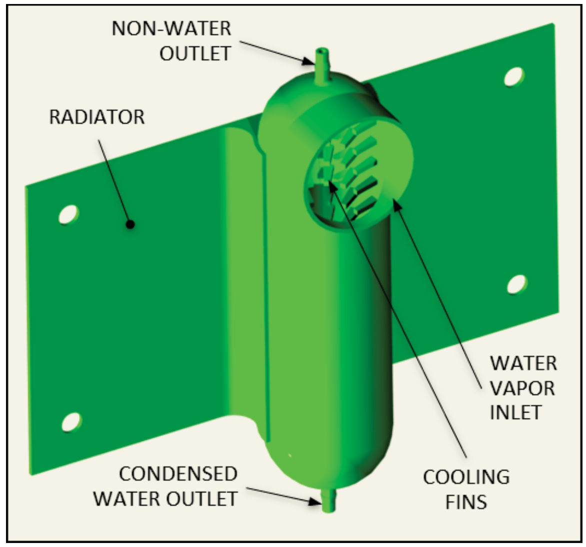

Figure 3-4: Conceptual Condenser Design

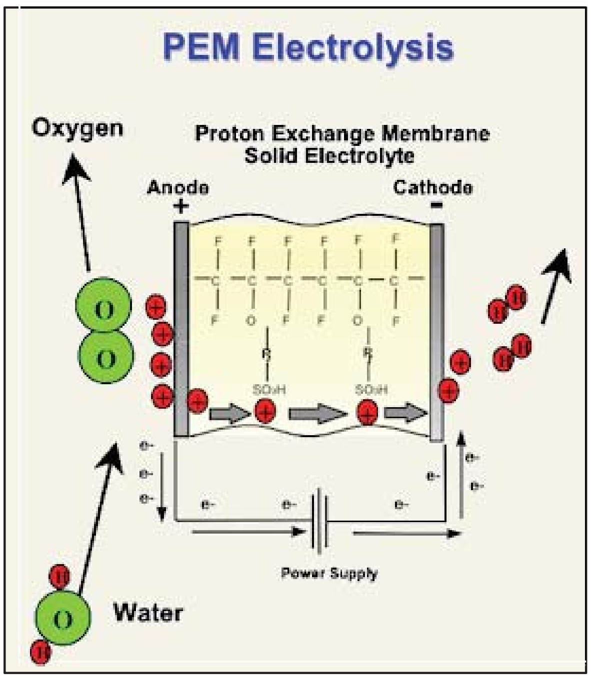

Figure 3-5: PEM Electrolysis

During Phase I, the electrolyzer setup was prototyped using a commercially available system, as shown in Figure 3-6. For Ph II, ExoTerra is developing this component with Giner Inc., a leading expert in electrolyzer design and manufacture in the United States, with experience in space applications of this technology. Giner has demonstrated electrolyzers with 88% efficiency and 1000 W/kg specific mass. This allows us to decrease the size of the electrolyzer within the bounds of a CubeSat scale. The total power required to electrolyze the water is 154 W. According to the Giner design, total mass for our application is expected to be .4 kg. The electrolyzer has a diameter of 13.5 cm and is 9.2 cm tall, occupying just over 1U. As an additional benefit, the design is capable of operating in 0g, allowing potential follow-on missions to asteroids.

3.4 HEAT EXCHANGERS

In the hydrogen loop, heat exchangers transfer heat from one stage of the process to another in order to reach the much lower temperatures required for hydrogen liquification. As shown in Figure 3-2, there will be two counterflow heat exchangers: one between the line from the hydrogen turbine and the line to the expansion valve, and the other between the lines to and

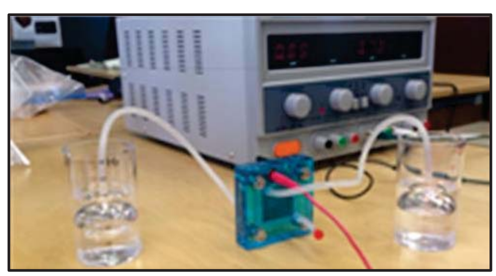

Figure 3-6: Electrolyzer Prototype

from the hydrogen tank. We've estimated the mass of the two heat exchangers at 200 g each. Detailed design of the heat exchangers will be performed in Ph II.

3.5 HYDROGEN LIQUEFACTION

We calculate that waste heat from the electrolyzer will result in an average output temperature of 425 K. The LH_2 production process begins by cooling the .91 mg/s of gaseous H_2 from the electrolyzer using a radiative cooler and the ambient conditions of Europa's surface to 116 K. The cold gas is combined with 2.21 mg/s of gas vented from the tank and 14.3 mg/s of recirculated flow from the turbine/heat exchanger and sent through a multistage compressor. The micro-compressor increases pressure to 2 MPa and 407 K using 72 W. Initial sizing of the compressor estimates the mass at 46 g.

The temperature is then reduced by an inline gas cooler to radiatively cool the fluid to 110 K. From there the fluid splits between a path to the turbine and to the Joule-Thompson expander. 82% of the fluid is sent to the turbine where it expands to 50 kPa, cooling to 27 K and providing 12.4 W of power to help power the compressor. The bypassed fluid passes through a heat exchanger to transfer heat to the now 27 K fluid. This drops the bypassed flow to 44 K. While the flow from the turbine line is sent back to the compressor, the bypassed flow passes through another heat exchanger coming up from the tank return at 18 K. This reduces the temperature to 36 K. Finally, the chilled H_2 gas flows through an expansion valve and ~29% liquefies at 18 K and 50 kPa. The liquid H2 is collected in the fuel tank. Saturated gas from incomplete liquification and tank boiloff is recirculated to the compressor inlet.

The resulting design requires custom compressors based on the cryogenic temperatures, low flow and high pressure/temp gain. Research and communications with pump manufacturer Barber-Nichols shows that modification to existing impeller designs is possible to make them suitable for our application. However, purpose-designed positive displacement pumps may more efficiently and cost-effectively meet the flow, temperature, and compression ratio conditions in the ISRU system. The detail design of these items will be performed in Ph II.

3.6 OXYGEN LIQUEFACTION

The LOx system follows a simpler sequence than the LH_2 system due to its higher boiling point. 7.35 mg/s of gaseous O2 from the electrolyzer is cooled to 116 K using a radiative cooler. The cold gas is combined with 2.56 mg/s of gas drawn from the oxidizer tank and then compressed to 900 kPa and 246 K. The compressor power is 1.2 W and it weighs an estimated 46 g. The gas is then cooled 116 K in a condenser prior to passing through the expansion valve to 50 kPa. 74% of the fluid liquefies, and is stored in the oxidizer tank. The remaining gaseous oxygen from boiloff and incomplete liquification is recirculated to the compressor.

Brushless DC motors will be used in both systems to increase reliability and decrease the risk of sparking. Compressor startup needs further consideration with both systems due to the recycled flow required for nominal operating conditions. Where necessary magnetic couplings, extended shafts, and linkages fabricated from composite materials will thermally isolate motors, actuators, control electronics, and cryogenic components from each other.

3.7 SYSTEM MONITORING

The ISRU process is monitored and regulated using a combination of temperature and pressure transducers, illustrated in Figure 3-7. The pressure of the system is measured using two type of pressure transducers. The first is a NOVA Sensor NPC-410 Series Medium Pressure Sensor, selected for its small size. The temperature range for this sensor is -40°C to 125°C, which covers the range for the water extraction and condensing portion of the ISRU process. The liquid hydrogen portions of the system use an OMEGA Thin-Film Cryogenic Pressure Transducer, with a temperature operating range of -196 to 149°C.

Figure 3-7: NOVA Pressure Sensor, OMEGA Cryogenic Pressure Transducer, Omega Cryogenic Temperature Sensor

Unfortunately, this sensor is much bigger than the NOVA sensor. There is lack of availability of pressure sensors that operate at the low temperatures required to condense hydrogen, especially in a compact package. Even this OMEGA sensor operating range does not reach coldest temperature we are expecting in the ISRU process, -230°C. Further discussions with pressure transducer companies are required to explore the possibilities of pressure sensors that operate at these low temperatures.

OMEGA Cryogenic Temperature Sensors, with an operating range of -271.7°C to 226.8°C (1.4K to 500K), measure temperatures throughout the ISRU system.

4.0 PROPULSION DESIGN

The NIMPH propulsion system shown in Figure 4-1 is a pump-fed LO_2-LH_2 design which uses propellants produced by the on-board ISRU equipment. During propellant production, shutoff valves isolate the tankage from the propulsion system. Prior to operation, upstream shutoff

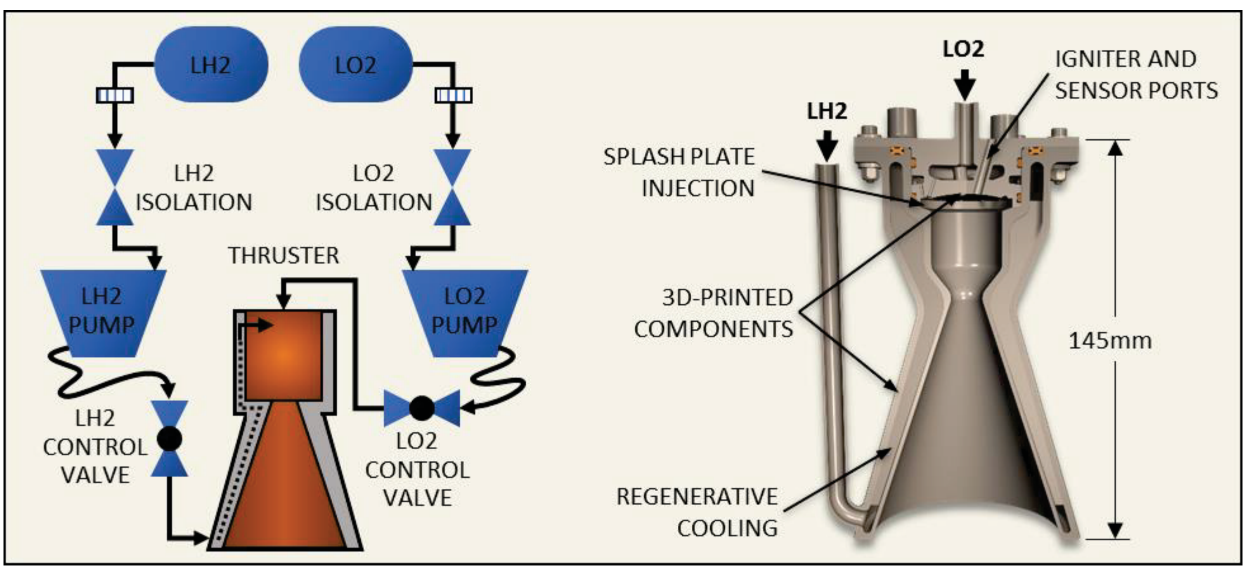

Figure 4-1: Propulsion Block Diagram and NIMPH Miniature LH2-LO2 Engine

valves close to isolate the ISRU fueling system from the tankage, and the isolation valves open to allow propellants to flow into the feedlines. Pumps draw in propellants from the feedlines and deliver them downstream at the required pressures and flow rates. To accommodate engine gimbaling for thrust vector control, the pumps connect via cryogenic flexlines to control valves on the engine head. These valves regulate propellant flow to the engine inlets to optimize combustion and throttle the engine as needed.

On arrival in Europa's orbit, the orbiting spacecraft supplies water and power to process into liquid oxygen and hydrogen using duplicate ISRU equipment on the orbiter. When fuel production is complete, the lander departs the orbiter and the NIMPH propulsion system deorbits and lands the vehicle at a selected site on the surface of Europa using this initial fuel load. Following a successful landing, the tank isolation valves are closed again, and the system is vented to vacuum until it is needed for ascent.

After sample collection and propellant production phases of the surface mission are complete, the NIMPH propulsion system returns the lander to orbit and rendezvous with the orbiter for retrieval and subsequent return to Earth.

Feedlines are made from 316 stainless steel for LH_2 and 18-8 stainless steel for LOx for material and cryogenic temperature compatibility.

4.1 PERFORMANCE SUMMARY

The NIMPH thruster design produces 71.5 N of thrust at 364 s Isp, yielding up to 1.81 km/s ΔV with the current propulsion system configuration.

Mass of the engine is 1.34 kg. Overall the engine is 145 mm long with an 85mm outside diameter at the nozzle exit plane.

4.2 ENGINE DESIGN

NIMPH uses an original-design 71.5 N LH_2-LO_2 bipropellant engine, whose key characteristics are shown in Figure 4-2. The engine consists of two major components, a nozzle body and injector head, which are 3-D printed to reduce design complexity, manufacturing and assembly costs, and mass. A miniaturized ignitor provides multiple restart capability.

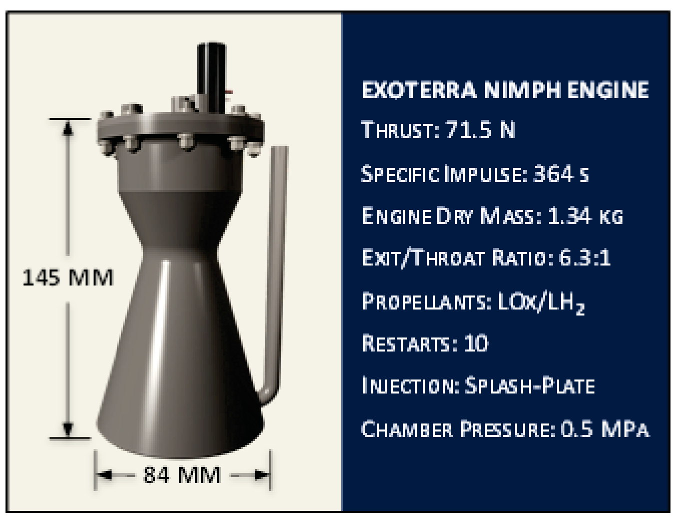

Figure 4-2: Details of the NIMPH Main Engine

4.2.1 NOZZLE BODY

The nozzle body combines three principal elements of the engine – nozzle, thrust chamber, and regenerative cooling jacket - into a single part 3-D printed from Inconel 625.

The nozzle is 88.9 mm long from throat to exit plane, with an 70.7 mm interior diameter (ID) at the exit plane and 11.2 mm ID at the throat. The shape is a simple cone of 18.7° half-angle. Above the throat is the thrust chamber. The thrust chamber is 47.2 mm long and 25 mm in internal diameter (including the injector head dome) and operates at 506 kPa internal pressure.

Surrounding the nozzle and thrust chamber is a regenerative cooling jacket. Liquid hydrogen fuel enters the jacket near the exit plane of the nozzle and is distributed around the nozzle's periphery. From there it is forced upward through 24 coolant passages, which are parallel to the axis of the engine and integrally printed with the nozzle body. At the top of the thrust chamber, flow combines into a single circumferential collector plenum from which it exits through twelve ports into the distribution plenum in the injector head. LH_2 enters the jacket at sufficient pressure to reach the combustion chamber at 506 kPa when allowing for pressure losses inside the flow passages en route to the injector.

This component weighs 1.05 kg.

4.2.2 Injector Head

The injector head combines propellant distribution and injection and includes accommodation for pressure and temperature sensors. It is 20 mm long and 37.8 mm in diameter (excluding the bolt flange and mounting features).

Liquid oxygen feeds into the injector head along the axis of the engine. Inside, a circular distribution plenum supplies 12 uniformly-spaced injection ports leading into the thrust chamber. Liquid hydrogen enters the injector head through the twelve distribution ports in the top of the nozzle body, into an annular distribution plenum, and out through matching injection ports.

The paired injection ports are angled to impinge at a point on an inclined/conical surface or (splash plate) inside the thrust chamber. This dual impingement (fuel with oxidizer, both propellants with the splash plate surface) assures thorough mixing and atomization of the liquid propellants as they enter the combustion zone.

Four 3 mm dia. threaded ports for thrust chamber pressure and temperature sensors are provided.

This component weighs 0.25 kg.

4.2.3 Igniter

The NIMPH propulsion system is required to self-start during two mission phases: descent and landing on Europa, and ascent and rendezvous with the orbiting satellite. With such a small engine and limited space and mass, the usual approaches (such as hypergolic injection and exploding bridgewires) were not feasible.

Instead, the NIMPH ignitor (Figure 4-3) lights the engine on command by triggering a spark through the splash plate injection zone. The simple self-regulating mechanism works like a carbon arc-lamp to compensate for any

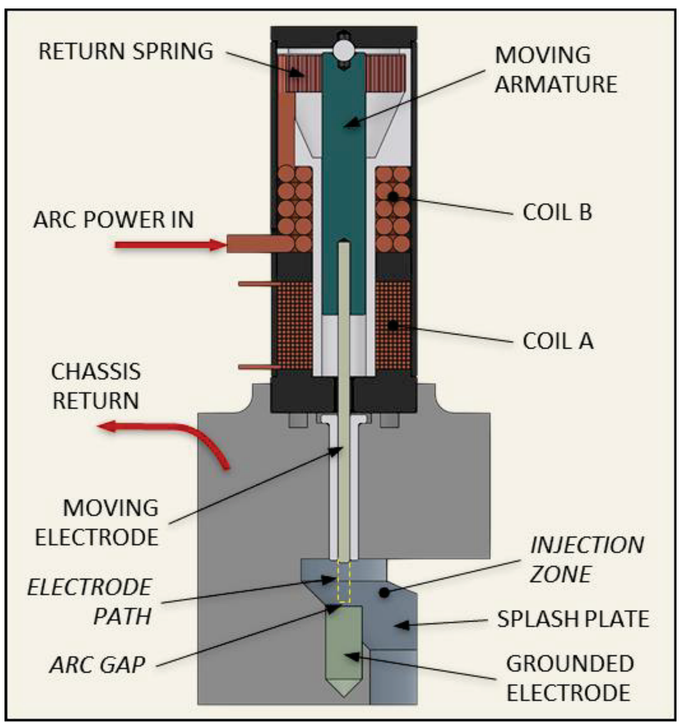

Figure 4-3: NIMPH Miniature Ignitor

erosion from the tip of the needle during startup and operation, supporting multiple restarts during testing and operations.

When the ignitor is activated, current passing through solenoid coil A pulls the moving armature downward, pushing the needle-like moving electrode through the injection zone and into contact with the grounded electrode. Current rises rapidly through solenoid coil B, wired in series with the needle, creating an upward pull on the armature. As the electrodes draw apart an arc is struck, and as the arc gap widens the current flow through the electrodes is reduced. The current through coil B decreases as a result, allowing coil A to pull the needle down again, and the sequence repeats.

Electrically the arc can be thought of as a variable resistor, with coil B working to increase the resistance and coil A to decrease the resistance. The arc gap widens if the current increases and narrows if the current decreases. The push-pull interplay of the two solenoid coils results in a self-regulating movement of the electrode as it maintains the arc gap required for the set operating current. Electrical energy is forced into the arc as heat sufficient to ignite the surrounding mixture of atomized propellants.

When ignition is successful and power is removed from the circuit, the return spring draws the needle back into its insulating sleeve. The return spring also conducts the operating current to the moving needle, and allows the needle to extend further into the injection area at startup to account for tip erosion. The needle can erode up to 2 mm and still reach the grounded electrode, after which the ignitor has reached the end of its useful life.

The ignitor body is 25 mm long and 13 mm in diameter, and is designed to have low power draw (5 A at 28 V) and a minimum of ten restarts of the engine for test firing and mission operations.

4.3 PUMPS

In Phase I, we explored options for cryogenic pumps fitting the needs of the NIMPH propulsion system. The unusual combination of high compression ratio (10-1) and low mass flow rates (2.5 g/s for LH_2, 20 g/s for LO_2) makes the selection of off-the-shelf components difficult. Cryogen pumps on this scale are typically impeller-type designs, from which it is difficult to attain the combination of pressure ratio and low mass flow rate.

Barber-Nichols was the manufacturer whose products most closely met our needs. While they did not have an off-the-shelf pump design which directly met these requirements, they provided two conceptual designs derived from existing products which will meet these performance specifications.

5.0 LANDER DESIGN

5.1 CONFIGURATION OVERVIEW

The proposed configuration for the Europa Lander is shown in Figure 5-1 and Figure 5-2. In the stowed configuration, the lander is 954 mm tall and 878 mm at the widest. Figure 5-1 shows the general dimensions of the lander. Figure 5-2 shows a detailed view of the Europa Lander, identifying key components of the system. The lander consists of a cylindrical body surrounding the hydrogen and oxygen tank. The hydrogen tank will be made of aluminum and used as a structural member in the assembly, while the oxygen tank will be made of a composite material

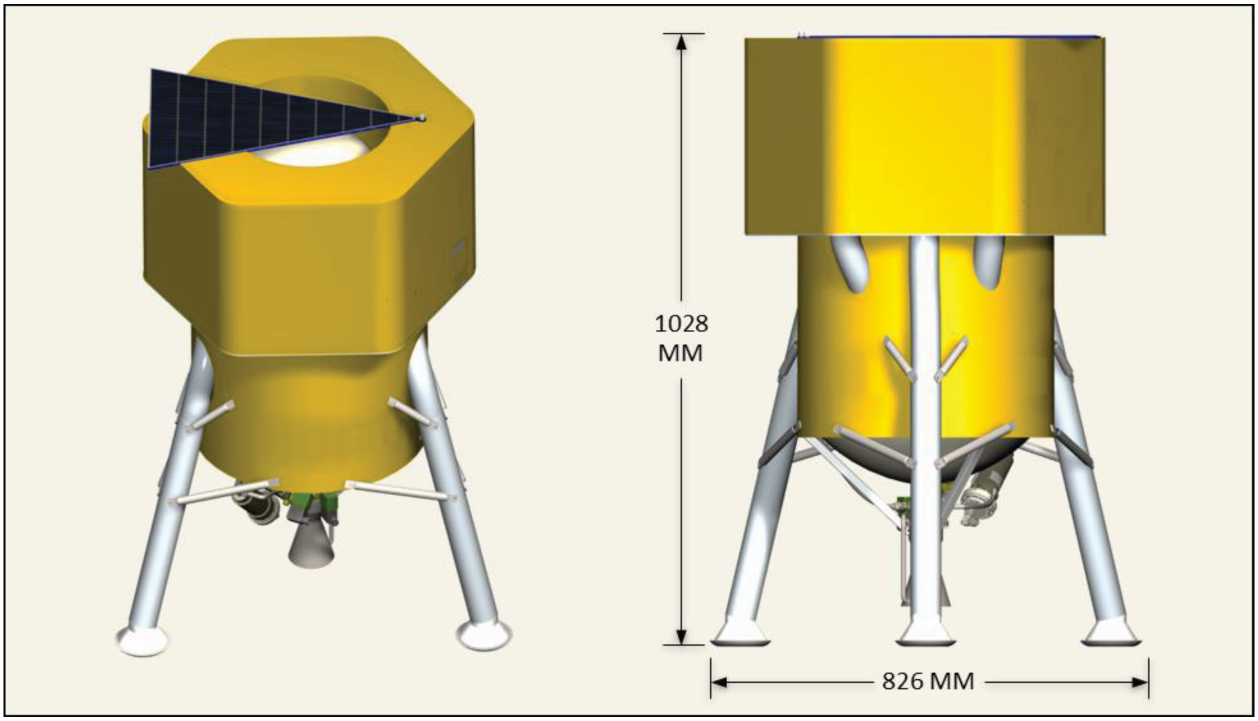

Figure 5-1: Europa Lander with Solar Array in Stowed Configuration

to save on weight. Attached to the main structure are the three landing struts with the heaters in the feet, which will sublimate the ice on the surface. Since the lander legs will likely get frozen into the icy surface of the moon during the water harvesting process, the legs will detach from the lander during takeoff. Leaving the legs behind will also reduce takeoff mass and improve launch margins. The top section of the lander, shown in yellow, contains all the avionics, batteries, communications devices, as well as the ISRU equipment to convert the harvested water into hydrogen and oxygen fuel.

The solar array of the lander is a disk shaped solar array measuring 1m in diameter, with 10 triangular sections. In the stowed configuration the triangular sections stack onto of the lander. The solar array deploys, in a clockwise circle, with each section following the one before. The solar array gimbal to deploy and maneuver the array consists of an extending rod to raise the array above the lander, and two independent rotation gimbals, to allow for a ±180° pitch rotation, and a ±35° yaw rotation. This range of motion allows the solar array to track to the orbiting satellite to maintain maximum power transfer.

The engine of the lander is located below the hydrogen tank, attached to the main structure of the lander. Two electromagnetic pumps are located with the thruster, to supply the hydrogen and oxygen fuel to the system. An Attitude Control System (ACS) will be used on the lander. Two pairs of opposing cold gas thruster will be placed near the exterior of the vehicle. These thrusters will provide pitch, yaw and roll control. The cold gas used will be hydrogen siphoned off the regenerative cooling of the hydrogen tank.

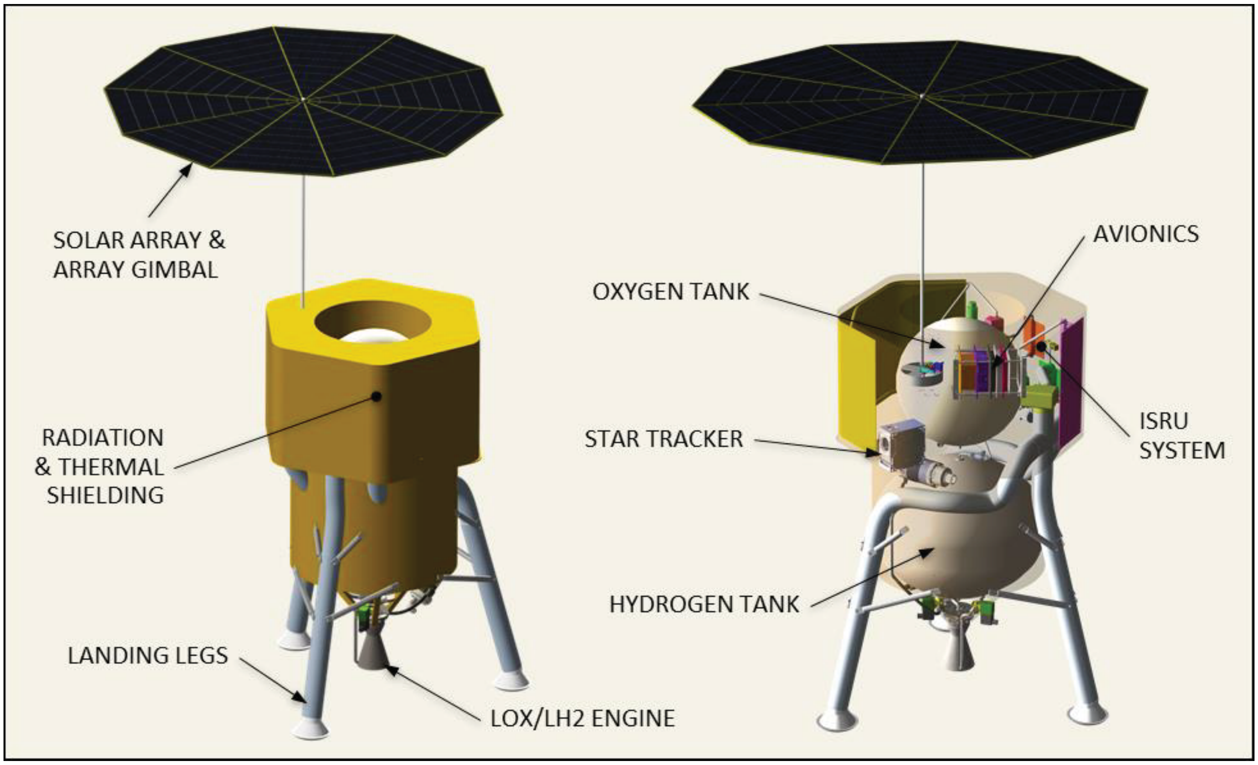

Figure 5-2: Detailed View of Europa Lander

5.2 MASS SUMMARY

The estimated masses for the Europa Lander are summarized by subsystem in Table 5-1. The current best estimate for the lander dry mass is 24.24 kg. For all categories, a Mass Growth Allowance (MGA) of 1.1 to 1.2 was used to allow for any uncertainty in the estimate based on the fidelity of the design. The estimated mass with MGA is 27.75 kg. We also add a 10% system margin to the vehicle. This margin takes into account project scope creep, unknowns and other changes, which inevitably happen on any project. Therefore, the Europa Lander has an estimated total dry mass estimate with margin of 30.52 kg.

Calculated ΔV for ascent is 1.6 km/s. In addition, we allocate .05 km/s for attitude control through the thrust vector control. Adding a 10% ΔV margin results in 1.81 km/s. We also include a 5% unused propellant margin. This results in a propellant mass of 21.95 kg, bringing the total mass to 52.5 kg.

Table 5-1: Mass Summary of Europa Lander

Subsystem	Current Best Estimate (kg)	Total Mass (kg)
Avionics	0.20	0.22
EPS	6.04	6.86
Telecom	0.30	0.33
GN&C	1.51	1.73
Harness	0.65	0.85
Thermal	0.38	1.02
Structure	5.53	6.34
Mechanism	0.96	1.38
Tanks	1.58	1.82
Propulsion	3.03	4.28
Payload	2.74	2.91
Sub Total	24.24	27.75
Total w/ Margin		30.52

5.3 POWER SUMMARY

As discussed in section 2.3, the NIMPH lander operates on power transmitted from

Table 5-2: Power Summary

Subsystem	Laserlight (W)	Eclipse (W)
Avionics	1.5	1.5
EPS	.4	.4
Telecom	6	0
Thermal	10	40
Mechanisms	10	1
ISRU	243	0
Total	271	43.9
Margin	17.7	6.6
Total w/ Margin	289.1	50.5
Battery Charge	188.9	

the orbiting satellite, receiving it via a deployable resonant array and regulating and distributing it to onboard systems and storage.

A 1 m diameter steerable resonant array (±180° pitch ±35° yaw; see Figure 4-2) tracks the satellite from horizon to horizon, receiving power transmitted via laser telescope. With a 500 km altitude orbiter, the satellite is in view for 39.2 minutes of each 173.1 minute orbit. The laser provides a flux of 2734 W/m2 at the array. We assume a 60% conversion efficiency. However, we have 17.8% losses from packing factor & manufacturing imperfections and 8% losses from environmental effects such as radiation & temperature. The latter actually offsets radiation as the cold temperatures improve cell efficiency. This results in 478 W of electrical power at 12 VDC.

While on the surface, very little power is drawn outside of the μISRU system. Besides the μISRU, power is required for the processor, telecommunications system, gimbal, heaters and recharging the battery. During eclipse, power is driven by heating the lander. Table 5-2 summarizes the power requirements. Power margin is currently at 6.5% during laserlight and 15% during eclipse. A late increase in ISRU power during Ph I eroded 20 W of planned margin.

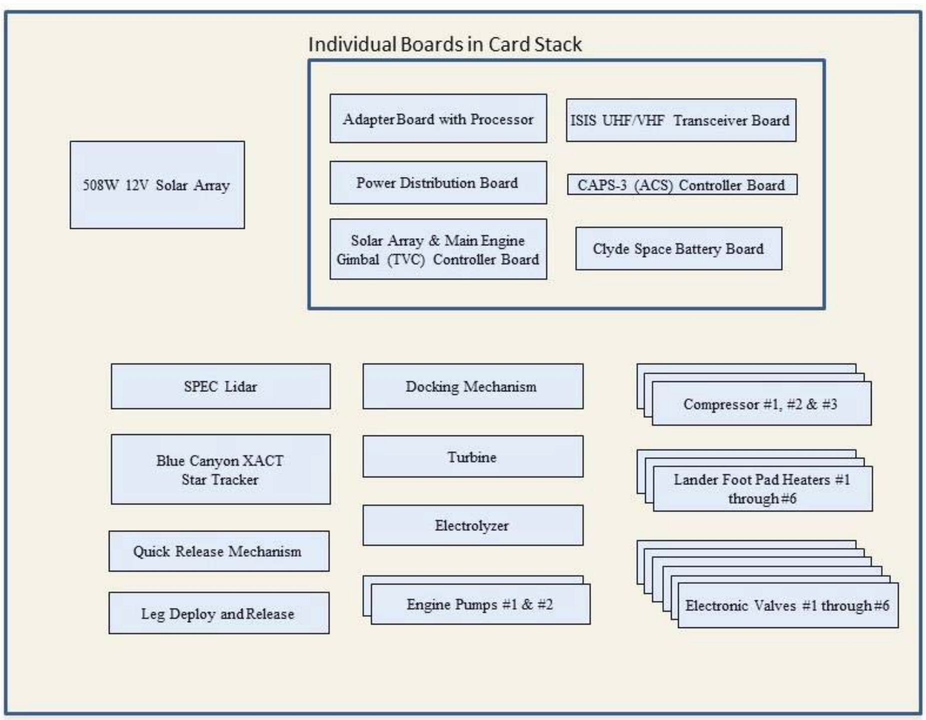

Figure 5-3: System Components

During Ph II, we will evaluate ways to increase the power margins.

Energy storage is accomplished with Li-ion batteries. We assume a 96% charge efficiency and 90% discharge efficiency. The 22.95 Ahr batteries provide a 10% energy margin to a 40% depth of discharge.

5.4 ELECTRONICS SUMMARY

The lander's core electronic boards (see Figure 5-3) are stacked together in a radiation-hardened vault for protection against the harsh radiation environment of Europa and the Jovian system.

The *Power Board* receives raw power from the resonant array and contains the regulation circuitry necessary to supply each of the components and subsystems on the spacecraft. In all, there are 26 switches (see Figure 5-4) available on the power distribution board. This includes 12V power to the avionics, guidance, pumps, electrolyzer, compressors, electronic valves, and turbine.

The *Adapter Board* consists of a rad hard processor as well as all the connectors to provide power and/or communications to the other components and units not located on the board stack.

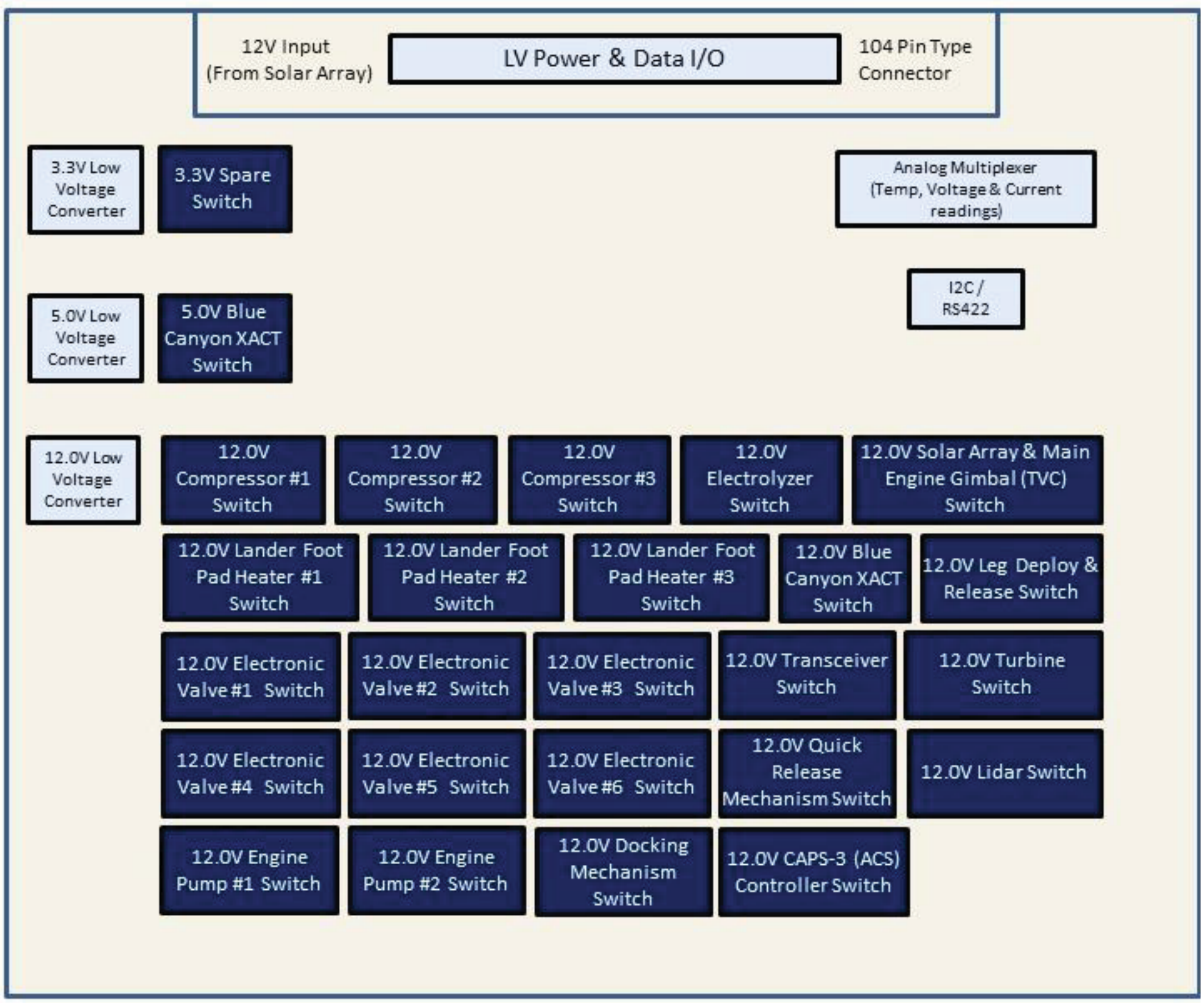

Figure 5-4: Power Distribution Board Layout

The processor controls and communicates with the stack through an onboard bus that carries both power and communications. The board links to the Battery Board via I2C, while RS422 interfaces relay commands, communications, and spacecraft health data.

The *Battery Board* controls power storage and provides backup power to vital units on the spacecraft while overhead satellite is out of range to power the lander.

The *Array and TVC Control Board* provides control to the gimbal unit connected to the resonant array to maximize power from the overhead satellite, and operates the main engine gimbal for pitch and yaw control.

The *Transceiver Board* maintains communications with the overhead satellite, the latter relaying commands and telemetry with mission control during free-flying and Europa surface operations.

A *CAPS-3 Controller Board* controls micro-thrusters which provide roll control to the spacecraft.

A Blue Canyon XACT unit combines multiple GN&C components (star tracker, inertial measurement, reaction control wheels, sun sensor) into a single package. For autonomous site selection and landing operations, we include a SPEC lidar for range and obstacle detection.

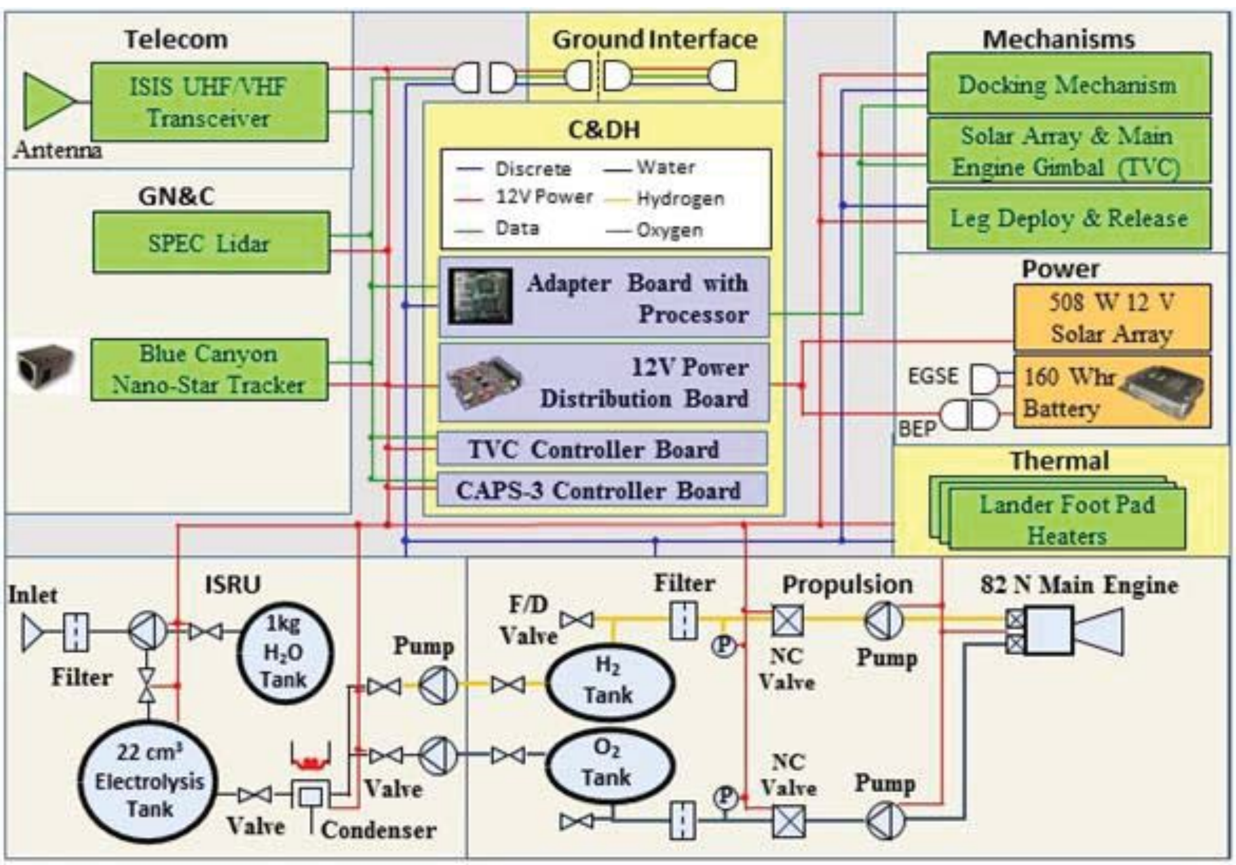

Figure 5-5: System Block Diagram

A block diagram for the lander is shown in Figure 5-5.

6.0 MISSION COST ESTIMATE

The bulk of the mission cost is driven by the launch vehicle and the SEP Module. The Atlas V cost is estimated using ULA's online tool. According to the tool, an Atlas V 551 with the full spectrum service option is $173M. We've rounded this value up to $180M.

Cost of the SEP Module is based on ExoTerra's detailed cost analysis performed as part of our Asteroid Redirect Mission Study for NASA. The cost estimate included quotes from vendors for hardware items, and assembly by Ball Aerospace. The recurring cost for the module was under $170M.

Mission operations costs have been estimated assuming a staff of 6 personnel monitoring the spacecraft 8 hrs per day for 15 years. At a labor rate of $150/hr, this results in $39M in operations labor costs. Using the DSN pricing tool, we estimate DSN costs at $100k per year, resulting in $1.5M over the lifetime of the mission. We have rounded the total cost up to $41M.

Lander costs are based a proposal for a similarly sized (45 kg) microsatellite ExoTerra proposed to the NASA Edison program to perform proximity operations. The equipment used in the microsatellite was used as the basis for the bus used on our lander, and the proximity operations mission was comparable in complexity to landing. Total cost to develop the microsatellite was <$15M.

In addition to the basic lander costs, we anticipate that 3 key items will need to be qualified: the ISRU, LOx/LH2 Engine, and power beaming. We anticipate that the ISRU can be developed and qualified for $2.5M. This roughly an SBIR Ph II award with Ph III support. We anticipate the engine will be more complicated and doubled the ISRU estimate to $5M to design and qualify. We have estimated the power beaming will require a similar $5M.

Due to the coarseness of the estimate, we've included a 20% ROM factor of $83M. This brings the total mission cost to $502M.

7.0 PHASE II RECOMMENDATION

Analysis of the NIMPH concept during Phase I has supplied promising results. All aspects of the design have shown feasibility with margin. This includes the trajectory analysis, power delivery system, propellant production, system mass, radiation exposure, and thruster performance.

The study has revealed an affordable means of sample return from Europa that can be used for sample return missions from the Moon, Mars, and other Jovian moons. In addition, ExoTerra has received interest in the ISRU system in support of Asteroid Mining companies. This provides potential commercial uses in addition to scientific sample return, expanding the technologies applicability.

While the system shows promise, many items still need to be resolved before it can be implemented. Key risks to the system include:

1. Effects of thermal capacitance in lines and their impact on the overall efficiency of the μISRU and μLOx/LH_2 engine. This may require a precooling system, or we may need to trade the impact of operating with gaseous H_2 v. liquid H_2.
2. Development of micro-pumps, compressors and turbines to the required scale and demonstration of performance.

3. Demonstration of the electrolyzer performance at scale.
4. Demonstration of power beaming system, including efficiencies and collimation accuracy.
5. Demonstration of power beaming pointing system.
6. Completion of the SEP Module.
7. Completion of a LOx/Methane engine development.
8. Radiation shielding

ExoTerra recommends proceeding to Phase II with a focus on developing and testing a functional prototype of the µISRU system and micro-LOx/LH2 engine. This mitigates several of the risks by demonstrating the ability to maintain stringent landed mass requirements, production efficiencies, and thruster performance.

8.0 BIBLIOGRAPHY

1. Kane, Van. "A Lander Mission for NASA's Europa Mission." The Planetary Society. 1/5/16. http://www.planetary.org/blogs/guest-blogs/van-kane/20160105-nasa-europa-lander.html
2. Berger, Eric. "How much will SLS and Orion Cost to Fly? Finally some answers." 8/19/16. ARS Technica. https://arstechnica.com/science/2016/08/how-much-will-sls-and-orion-cost-to-fly-finally-some-answers/
3. Rocket Builder. ULA. https://www.rocketbuilder.com/
4. Crump, Paul, et al. "SHEDs Funding Enables Power Conversion Efficiency up to 85% at High Powers from 975 nm Broad area Diode Lasers." http://www.nlight.net/nlight-files/file/technical_papers/SPRCS05_stanford.pdf
5. Andreev, Viacheslav, et al. "High Current Density GaAs and GaSb Photovoltaic Cells for Laser Power Beaming." Ioffe Physico-Technical Institute. St. Petersburg, Russia.
6. Laser Motive. "Laser Power Beaming Fact Sheet" pg 4
7. Orbital ATK. "UltraFlex Solar Array Systems Fact Sheet"
8. Podzolko, MV, Getselev, IV. "Radiation Conditions of a Mission to Jupiter's Moon Ganymede." Skobeltsyn Institute of Nuclear Physics. Lomonosov Moscow St University
9. Truscott, Pete, et al. "Integrated Radiation Mitigation and Shielding Design." January 18, 2010.
10. Andreas, Edgar. "New estimates for the sublimation rate for ice on the Moon" US Army Cold Regions Research and Engineering Laboratory. April 6, 2006. www.sciencedirect.com