KOIOS

Nuclear-Rocket Science Mission to Saturn

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December 19th, 2022
University of Michigan, NERS 579 Space Nuclear Power

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Mission CONOPS

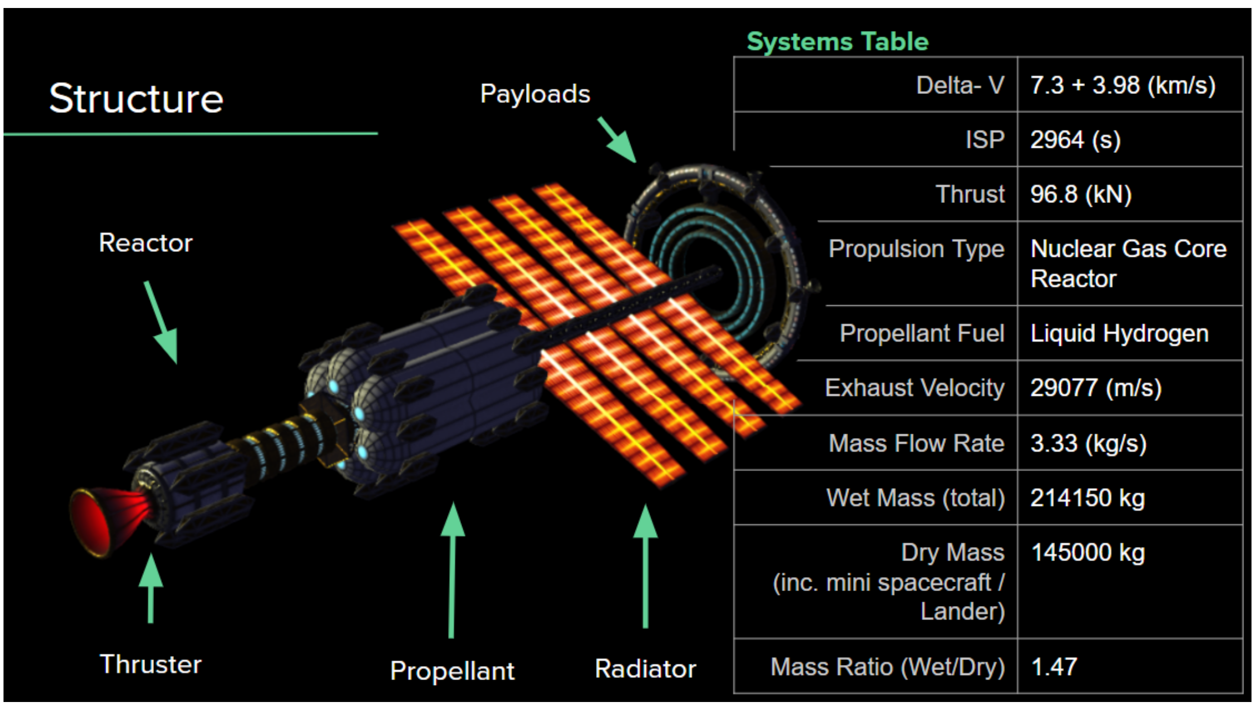
Responsible: Erika Hathaway, team

KOIOS is a joint NASA and US-SF science exploration mission to Saturn's moon Titan, aimed at establishing an outer-heliosphere science and refuel hub for deep space missions. Equipped with novel life support and radiation shielding for three astronauts, the 6-year journey and 15-year exploration will be powered by a nuclear reactor. KOIOS is aided by several 'mini-spacecrafts', capable of autonomous operations mid-journey and science quests at Titan and other orbital bodies around Saturn. One of the main features of this craft is the ability to extract ice from Enceladus and transport it back to the main hub as a fuel source.

Requirements

All preliminary top-level requirements of key subsystems are listed within the report and/or in the report appendix.

System Overview



Delta- V	7.3 + 3.98 (km/s)
ISP	2964 (s)
Thrust	96.8 (kN)
Propulsion Type	Nuclear Gas Core Reactor
Propellant Fuel	Liquid Hydrogen
Exhaust Velocity	29077 (m/s)
Mass Flow Rate	3.33 (kg/s)
Wet Mass (total)	214150 kg
Dry Mass (inc. mini spacecraft / Lander)	145000 kg
Mass Ratio (Wet/Dry)	1.47

(Rendering of Spacecraft was created using https://spaceengine.org/shipeditor/)

Orbit

Responsible: Joe Chen

Four missions have been sent to Saturn: Cassini, Voyager 1, Voyager 2, and Pioneer 11. Cassini was an orbiter, while the others were flybys. Nasa's upcoming mission, Dragonfly, plans to launch a drone to explore the surface of Saturn. Our mission is to establish an orbiter and refueling station around Titan and study the trajectory of Cassini and Dragonfly, as they are most relevant to our mission. These missions require a high level of expertise and resources, so we will take their experiences into consideration when planning our own trajectory. As we plan our trajectory, we will also perform a first principles analysis to determine the most efficient route.

To reach the Saturnian System, our spacecraft will have to cover a distance of approximately 1.2 billion kilometers through the vast expanse of space. The duration of this journey will depend on the specific trajectory and speed of the spacecraft, which could take several years to complete. For our mission to Titan, we will consider one of the most commonly used trajectories for spacecraft travel, the Hohmann transfer orbit. This orbit involves sending the spacecraft on an elliptical path. This allows the spacecraft to take advantage of the gravitational pull of the Sun and the planets to change its velocity and direction, conserving fuel and minimizing the overall energy required for the journey.

To further refine our trajectory, we will also apply the principles of patched conics, which take into account the gravitational influence of both the departing and arriving planets. This will allow us to fine-tune our trajectory and optimize the energy required for the mission. Patched conic methodology separates the trajectory to three parts:

1. *The Departure:* During this phase, we will use a hyperbolic escape maneuver to break free from the gravitational influence of the departing planet and set ourselves on a course towards Saturn
2. *The Transfer:* This phase involves traveling through space along the Hohmann transfer orbit, with the Sun at the center of mass. This trajectory allows us to take advantage of the gravitational pull of the Sun and planets to change our velocity and direction, conserving fuel and minimizing the overall energy required for the journey.
3. *The Arrival:* Upon reaching Saturn, we will use a burn maneuver to enter into orbit around the planet and begin our mission to study the unique topology of the Saturnian system.

We will also consider using simple single planet gravitational slingshots to reduce the Delta-V requirements for this mission.

After consideration, we have decided to construct our spacecraft in low Earth orbit (LEO). There are several benefits to this approach. Firstly, building the spacecraft in LEO would reduce the risk of damage to the ship during launch from the Earth's surface. Secondly, we would be able to save a significant amount of fuel, approximately 9.4 km/s, by eliminating the need to lift the spacecraft from the surface of the Earth. However, there are also some drawbacks to this approach. One of the main challenges is the cost of transporting the necessary parts and materials to LEO, which may require multiple trips and add to the overall cost of the mission. Additionally, the process of constructing the spacecraft in LEO itself may pose logistical and technical challenges. Using the ISS and robots for construction may be the best option, there are plans to commercialize the ISS [1]. Nevertheless, we believe that the technology will be advanced enough to overcome these obstacles and that cost is not a major concern for this mission.

Orbit Calculations

To perform a Hohmann transfer from Earth to Saturn, we need to calculate the heliocentric velocities of both planets (VE and VS) and the transfer velocities (VtransE and VtransS). The delta-v, or change in velocity, required for this transfer is the difference between the transfer velocity and the velocity of each planet. Our calculations show that the delta-v needed at departure is 10.3 km/s and at arrival is 5.4 km/s, for a total delta-v of 15.4 km/s for the mission. This is shown in the top three rows of Table [1]. The Hohmann transfer is the most energy-efficient method for a spacecraft to move from one planet to another.

Departing Earth

$V_E = \sqrt{\frac{\mu_{sun}}{R_E}}$	29.7 km/s
$V_{transE} = \frac{1}{R_E}\sqrt{\frac{2\mu_{sun}R_E R_S}{R_E + R_S}}$	40.0
$V_{infE} = \lvert V_{transE} - V_E \rvert$	10.3
$V_{parkE} = \sqrt{\frac{\mu_E}{R_E + r_{pE}}}$	7.4
$V_{burnE} = \sqrt{V_{infE}^2 + 2V_{parkE}^2}$	14.6
$\Delta V_{dep} = \lvert V_{burnE} - V_{parkE} \rvert$	7.3

Arriving at Saturn

$V_S = \sqrt{\frac{\mu_{sun}}{R_S}}$	9.6 km/s
$V_{transS} = \frac{1}{R_S}\sqrt{\frac{2\mu_{sun}R_E R_S}{R_E + R_S}}$	4.2
$V_{infS} = \lvert V_{trans} - V_S \rvert$	5.4
$V_{parkS} = \sqrt{\frac{\mu_S}{R_S + r_{pS}}}$	5.5
$V_{burnS} = \sqrt{V_{infS}^2 + 2V_{parkS}^2}$	9.5
$\Delta V_{arv} = \lvert V_{parkS} - V_{burnS} \rvert$	3.98

Table 1. Equations and Calculations for Hohmann Transfer

To include the sphere of influence (SOI) of the planets in our Hohmann transfer calculation, we used the patched conics method [2][3]. In the heliocentric frame, the transfer itself follows an ellipsoidal Hohmann trajectory. However, in the planetary frame, we need to perform a hyperbolic escape to push through the SOI and enter the Hohmann transfer. To do this, we need to choose a parking orbit distance away from the planet. In this case, we have chosen a distance of 1000 km from Earth. The reason for this choice will be explained later in the section. We can then calculate the velocity of the parking orbit. The burn velocity required for the hyperbolic escape is calculated in row 5 of Table 1, with the delta-v from the Hohmann transfer denoted as Vinf. By taking the difference between the burn velocity and the parking velocity, we can determine the delta-v required to depart the planet's SOI and enter the transfer orbit, as well as the delta-v required at arrival.

$$\tau = \frac{\pi}{\sqrt{\mu_{sun}}}\left(\frac{(R_E + r_{parkE}) + (R_S + r_{parkS})}{2}\right)^{\frac{3}{2}} \approx 6.04\ years$$

$$\theta = \pi - \sqrt{\frac{\mu_{Sun}}{R_S^3}}\tau = 106\ deg$$

$$t_{wait} = \frac{-2\theta - 2\pi N}{\sqrt{\mu_{sun}}\tau(\frac{1}{R_S^3} - \frac{1}{R_E^3})} = 1.65\ years$$

The time of flight for a one way trip to Saturn takes about ~6.04 years with a launch phase angle of 106 degrees between the two bodies. Since this is a permanent orbiter there is no need for a return time, but if there is a need to return. The wait time before the return trip to align is 1.65 years. This is calculated with the equations above.

Leave	Month Day	Year	Position degree	Arrive	Month Day	Year	Position degree
2023.3580	5/10	2023	229	2029.4046	5/27	2029	49
2024.3932	5/23	2024	241	2030.4397	6/9	2030	61
2025.4283	6/5	2025	254	2031.4748	6/22	2031	74
2026.4635	6/18	2026	267	2032.5100	7/5	2032	87
2027.4986	7/1	2027	279	2033.5451	7/18	2033	99
2028.5338	7/13	2028	292	2034.5803	7/30	2034	112
2029.5689	7/26	2029	305	2035.6154	8/12	2035	125
2030.6041	8/8	2030	317	2036.6506	8/25	2036	137
2031.6392	8/21	2031	330	2037.6857	9/7	2037	150
2032.6744	9/3	2032	343	2038.7209	9/20	2038	163
2033.7095	9/16	2033	355	2039.7560	10/3	2039	175
2034.7447	9/28	2034	8	2040.7912	10/15	2040	188
2035.7798	10/11	2035	21	2041.8263	10/28	2041	201
2036.8149	10/24	2036	33	2042.8615	11/10	2042	213
2037.8501	11/6	2037	46	2043.8966	11/23	2043	226
2038.8852	11/19	2038	58	2044.9317	12/6	2044	238

Table 2.Hohmann Launch Windows

We have included a table of launch windows [4], Table 2, for the next 15 years starting in 2023. With this we can determine when to launch our spacecraft from parking orbit into the transfer. The synodic period between the planets is 377 days.

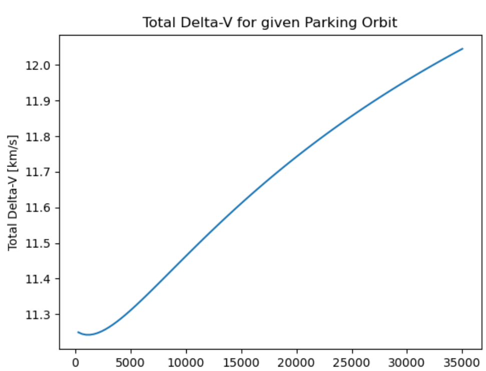


Figure 1. Total Delta-V vs Parking Orbit

To determine the optimal parking orbit for our spacecraft and minimize the amount of delta-v required to leave Earth, we conducted a sweep of parking orbit distances and calculated the total delta-v needed to perform the necessary maneuvers for our mission. Our results are shown in Figure 1.

From this analysis, we found that there was a potential minimum at around 1000 km from the Earth's surface, which would require a departure delta-v of 7.3 km/s. Based on this information, we have chosen this point as the departing point for our mission, resulting in a total delta-v of 11.25 km/s. It is worth noting that the amount of delta-v required only varied slightly with the parking orbit distance, ranging from approximately 11.3 km/s near the surface to 12 km/s at 35000 km.

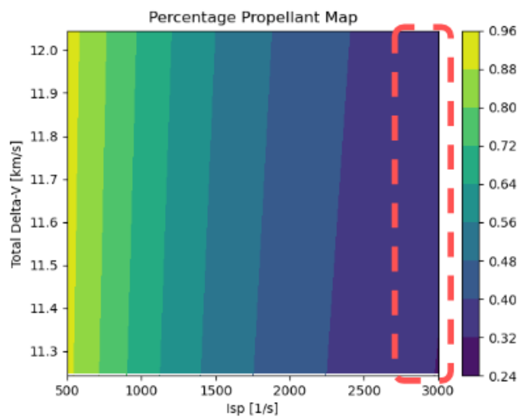


Figure 2. Percentage Propellant Map

To determine the mass ratio for our one-way trip to Saturn, we used the rocket equation and varied the specific impulse (ISP) from 500 to 3000. The resulting contour map is shown in figure 2, with the color bar representing the percentage of propellant required for the entire spacecraft. According to this map and the reactor we have selected, the mass ratio falls within the range of 32-40% propellant. This indicates that the amount of propellant required for our mission, with a delta-v of approximately 11.3 km/s, is reasonable. However, if the percentage of fuel needed were to reach around 50% for ISP values below 1600, we would consider additional methods to reduce delta-v, such as staging or gravity assist maneuvers. In the next section, we will explore how much delta-v we can gain through a single planet flyby using a gravity assist.

Gravity Assist Consideration

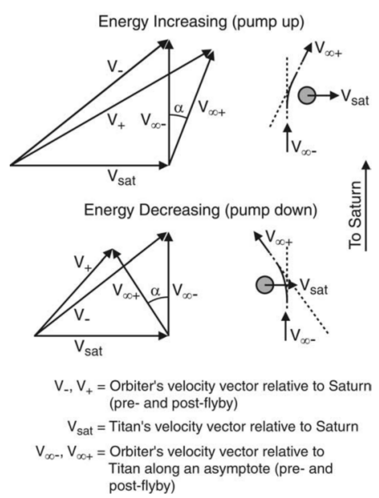


Figure 3. Gravity Assist Velocity Vectors

$$\Delta V_{ga\_max} = \sqrt{\frac{\mu_{ga}}{r_{ga}}}$$

$$V_{inf} = V^{-} - V_p$$

$$a = -\frac{\mu_p}{V_{inf}}$$

$$e = 1 - \frac{r_p}{a}$$

$$\delta = 2\sin^{-1}\left(\frac{1}{e}\right)$$

$$\overrightarrow{V^{+}} = \begin{bmatrix} cos\delta & -sin\delta \\ sin\delta & cos\delta \end{bmatrix}\left(\overrightarrow{V^{-}} - \overrightarrow{V_p}\right) + \overrightarrow{V_p}$$

V^{+} = *Exit Heliocentric Velocity from gravity assist planet*

V^{-} = *Arrving Heliocentric Velocity to gravity assist planet*

ΔV_{ga} = *Delta V gain from gravity assist planet*

δ = *optimal turning angle*

We wanted to know how much delta-v we can gain through a single planet flyby at the optimal turning angle. The gravity assist maneuver is innately a vector problem. To perform an assist one would need to consider the heliocentric frame as well as the planetary frame. The craft is sent to the assisting planet via Hohmann transfer, as it enters the SOI of the assisting planet we know the Heliocentric velocity V- and the planet velocity. The magnitude difference in the two determines the Vinf in the planetary frame. The craft or object then turns at some angle, exiting away from the planet with the same magnitude Vinf as incoming. In the planetary frame, there is no gain in velocity but when you go back to the heliocentric frame there is. The delta-v gain or loss is defined by V- - V+. The exiting velocity vector V+ is gained through the change in direction, this is shown in the figure 3 [5].

The references used for the gravity assist:[6-8] For calculations of the assist, we assume that the incoming velocity is determined by a Hohmann transfer. The equations to the left are used to determine the maximum turning angle. We first need to know the eccentricity as we arrive onto the planet. Is a function of the radius at which we are turning, the maximum radius we can achieve is to the surface of the planet, rp. From this we have a turning angle and can plug it into the equation. We also assume that the velocity vector for the planet and incoming object is purely in a single direction at arrival, parallel to each other. These calculations are shown in table 3.

Gravity Assist Planet	V^-	V_p	V_{inf}	e	δ, deg	V^+	ΔV_{ga}	ΔV_{ga_max}	ΔV_{dep} Via Hohmann	*Time of Flight* Years
Venus	37.73	35.03	2.7	1.136	123.35	$[33.55\,\hat{u}\,;2.25\,\hat{y}]$	4.75	7.3	2.47	0.4
Mars	21.48	24.13	2.65	1.556	79.98	$[23.67\,\hat{u}\,;-2.6\,\hat{y}]$	3.4	3.6	2.95	0.71
Jupiter	7.41	13.06	5.64	1.018	154.42	$[18.16\,\hat{u}\,;-2.4\,\hat{y}]$	11.02	42.6	8.79	2.73

Table 3. Gravity Assist Parameters

We consider three planets for a gravity assist that is in reasonable reach: Venus, Mars, and Jupiter. We find that Jupiter departs the most delta-V onto our flight while Mars is the least. This would make sense due the size of the planet. However we will be able use most of Mars delta-v gain since we gain 3.4 km/s while the max is 3.6kms/s. We also must take into consideration the departure delta-v to reach the gravity assist planet. To get to Jupiter would require more delta-V then to Venus and Mars. It is difficult to tell which trajectory would be best to take without further detailed calculations or proper simulation since the burn velocity into Saturn after the assist is not known. If we did know the burn into Saturn from the assisted planet then the total required delta-v for the mission would be the sum of our departure, burn to Saturn, and transfer velocity needed after the gravity assist to the destination planet. Normally we would like to minimize this transfer velocity to zero but that may not be the case for Venus and Mars since there is a small delta-v gain from assist and they are also further away from Saturn. Jupiter however may have an excess amount of gain which may not be needed. Also considering the additional time of flight needed to perform this maneuver. For Mars and Jupiter, since they are within the path to Saturn it would only change the total time of flight slightly. If a gravity assist was to be performed on Venus, one must take in account an additional 0.4+ years to the flight time. Ultimately a gravity assist is a tradeoff between delta-V and time of flight.

Past Trajectories to Saturn

Taking a look at Cassini's trajectory[9-10], this mission performed a Venus-Venus-Earth-Jupiter gravity assist which took 6.7 years to reach Saturn with a total delta-V of 2 km/sec. While with a straight Hohmann transfer, the time required is less but the delta-V required was 15.7 km/sec. The amount of fuel saved is tremendous when gravity assist is applied properly. Dragonfly's trajectory requires even less delta-v than Cassini, a grand total of 0.217 km/s [11]. The maneuvers performed were: Earth gravity assist, Venus gravity assist, two more Earth gravity assists into Titan. With this the time of flight for this trajectory will take ~ 9 years to perform. Another mission proposed was the TSSM which uses a solar electric propulsion system[12]. This mission will take 9 years as well and a total delta-v of 2.3 km/s while using the same gravity assist path as dragonfly. We can also take a look at Voyager's velocity gain from each flyby from figure [13]. It gained ~11km/s from Jupiter, ~10km/s from Saturn, ~2km/s from Uranus, and a decrease from Neptune of ~2km/s. With just a single flyby from Jupiter, Voyager 2 had enough velocity to escape the solar system.

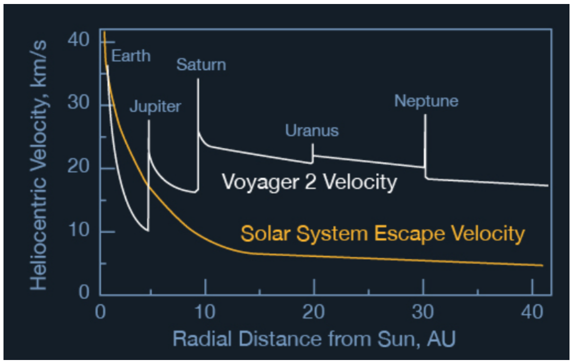


Figure 4. Heliocentric Velocity of Voyager 2

System: Propulsion

Responsible: Marco Daniel Acciarri

The nozzle design for the space hub requires as input the chamber pressure, temperature and propellant mass flow from the nuclear reactor design and the thermal analysis as shown in the following table. The nozzle exit pressure is arbitrarily set to 50000 Pa.

P_0	67.6 Mpa
T_0	33800 K
P_{ex}	50000 Pa
dm/dt	3.33 kg/s

In all the following analysis the isentropic flow assumption was made and no frictional losses were accounted for. In all the calculations hydrogen was assumed to be the propellant. The P_{ex}/P_0 ratio for this nozzle is set to 0.0007396 and the initial hydrogen density for the initial pressure and temperature is calculated using the equation of an ideal gas obtaining a value of ϱ_0 = 0.48497 kg/m^3 .

The design consists of a converging - diverging nozzle. The converging part increases the flow speed monotonically until the flow becomes choked (sonic) at the throat. Then, in order to keep increasing the flow velocity to the supersonic regime, due to the compressibility of the gas, the cross sectional area needs to increase down to the exit plane. The throat parameters (pressure, cross sectional area, density and temperature) where computed using the following isentropic gas flow equations [1]:

Throat parameters

$$P_t = \frac{P_0}{[1+(\frac{\gamma-1}{2})]^{\frac{\gamma}{(\gamma-1)}}} \qquad A_t = \frac{\dot{m}\sqrt{T_0}}{P_0\sqrt{\frac{\gamma}{R}\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma+1}{\gamma-1}}}} \qquad V_t = \frac{\dot{m}}{\rho_t\, A_t}$$

$$\rho_t = \frac{\rho_0}{(P_0/P_t)^{\frac{1}{\gamma}}} \qquad T_t = \frac{P_t}{R\,\rho_t}$$

Where γ is the ratio between the specific heat at constant pressure to the specific heat at constant volume of Hydrogen, T is the temperature, P is the pressure, V is the flow velocity, ṁ is the mass flow, A is the cross sectional area and ϱ is the gas density. The values obtained for each parameter are shown in the following table:

P_t	35.71 MPa
ϱ_t	0.30744 kg/m^3
T_t	28166 K
A_t	0.0008494 m^2
V_t	12752 m/s
D_t	0.032885 m

Exit parameters

The exit plane parameters (Mach number M_{ex}, exit temperature T_{ex} and exit sound speed Cs) where computed using the following equations [1]:

$$M_{ex} = \sqrt{\left[\left(\frac{P_{ex}}{P_0}\right)^{(\gamma-1)/\gamma} - 1\right]\frac{2}{\gamma-1}} \qquad T_{ex} = \frac{V_{ex}^2}{M_{ex}^2\,\gamma\,R} \qquad C_s = \sqrt{\gamma\,R\,T_{ex}}$$

And the specific impulse of the system using:

$$Isp = \frac{\sqrt{2\,\frac{\gamma}{\gamma-1}\,R\,T_0\,\left[1-(\frac{P_{ex}}{P_0})^{(\gamma-1)/\gamma}\right]}}{g_0}$$

where R is the specific gas constant of hydrogen and g_0 is the acceleration of gravity on the surface of the Earth. The values for the specific impulse (Isp), M_{ex}, T_{ex}, and C_s are shown in the following table. It is remarkable that thanks to the large chamber temperature and pressure, the specific impulse of this system, based on a gaseous core nuclear reactor, is almost 3000 s. This is more than 6 times larger than conventional chemical rockets based on LOx+LH_2 fuel, while providing a large thrust of $\dot{m}$*Vex = 97.5 kN.

Isp	2989 s
M_{ex}	5.85
T_{ex}	4308 K
V_{ex}	29295 m/s
c_{ex}	4987 m/s

For the diverging part of the nozzle, a standard cone half angle of 15 degrees was assumed, giving a length (from the throat) of 0.3626 m as shown in the following table:

A / A_t	47.75
A_{ex}	0.040559 m^2
D_{ex}	0.2273 m
Length	0.3626 m

The exit area of the nozzle is then 0.040559 m^2 and the nozzle area ratio (exit plane to throat) is 47.75.

Calculation of Propellant for the Mission and final mass

Responsible: Marco Daniel Acciarri

The weight of each system is indicated in the following table, including mini spacecraft and propellant tank:

System	Weight (kg)
Life support	8500
Radiation	1500
Reactor+Nozzle	90000
Mini spacecraft	9083
Electrolysis module	3000
Radiator mass	11000
Propellant Tank	9000
Structural Mass	13208
Dry mass	**145291**

The amount of propellant needed for the mission was calculated in the following way, assuming an initial dry mass (Mf) that did not include the tanks weight, the mass Mi was calculated using the rocket equation (using the delta V from the orbit section):

$$M_i = M_f \exp\left(\frac{\Delta V}{g_0 \, Isp}\right)$$

Where Isp is the specific impulse computed using the isentropic flow equations in the last section. Then the propellant was calculated by subtracting Mi-Mf. Then the propellant tank weight was estimated using the equation M_{tank}(kg) = 9.09 (Propellant Volume [m^3]) [1]. Then, the mass tank was accounted for in the calculation of the propellant needed again giving a final value of 68240 kg. The final dry mass of the spacecraft is 145291 kg. Then the rocket equation was applied again, separating the analysis between the departure burn and the arrival burn to Saturn through a Hohmann transfer. The obtained parameters are shown in the following table as well as the burn time in each case for the specified mass flow. The payload to total mass ratio is 0.68 meaning that only 32 % of the total weight is propellant.

Initial Mass (kg)	213531
Final Mass (kg)	145291
Propellant Mass (kg)	68240
Payload to Total mass Ratio	0.6804
Burn 1 Propellant Mass (kg)	47098
Burn 1 final Mass (kg)	166434
Burn 1 total time (s)	14143
Burn 2 Propellant Mass (kg)	21143
Burn 2 final Mass (kg)	145291
Burn 2 total time (s)	6349

Reference:

[1] https://spacecraft.ssl.umd.edu/academics/791S16/791S16L08.MERsx.pdf

Reactor Design

Responsible: Levi Welch

The main principle of any thermal nuclear rocket is to heat propellant via fissionable material such as Uranium, Americium, and Plutonium to name a few. Typically, the hotter the reactor is, the greater the exhaust velocity, and in return a substantial specific impulse is achievable. When the specific impulse is large enough the rocket has more delta-V to work with; the rocket can go further or carry a greater payload. Conventionally, nuclear thermal reactors (NTR) are limited heavily on the melting point of their housing materials. Solid core NTR rockets cool their fuel to achieve low control-able temperatures which limit its specific impulse to ~1,200 seconds (upper-bounds). Admittedly, NTR are considerably better than chemical rockets, averaging at 450 seconds, yet are not practical to explore the solar system in a practical, timely manner. That begs the question, why even bother with cooling the reactor in the first place? If one allows uranium to reach a liquid temperature the specific impulse increases to 2,000 seconds. The basic idea here is to spin a chamber about an axis to utilize centrifugal force applied on the uranium fuel to ”confine” it as propellant travels along the axis. Again why stop here, consider the gaseous core reactor (GCR) which vaporizes the fuel into white-hot gas at about 1,000’s of Kelvin! At this temperature the specific impulse can be up to 3,500 seconds for GCR.

A significant drawback to GCRs is its problematic exhaust composed mainly of propellant and varying degrees of radioactive fuel. It's expensive to let your deadly fuel exhaust from the chamber into the environment. Hence there are two schools of thought, close-core and open-core systems. Closed GCR, tries to have it both ways to its detriment; halving its specific impulse compared to its counterpart. Open GCR, lets it all out while trying to contain as much fuel as possible, but radioactive fissionable vapor byproducts escapes the chamber making it considerably unhealthy to be near while the rocket operates.

Solid-core		Liquid-core		Gaseous-core	
Exhaust Velocity, [m/s]	5,900- 9,800	Exhaust Velocity, [m/s]	9,800 - 25,500	Exhaust Velocity, [m/s]	25,500 -69,000
Specific Impulse, [s]	601 - 1,800	Specific Impulse, [s]	1,150 - 2,080	Specific Impulse, [s]	3,500 - 6,000
Specific Power, [kg/MW]	0.18 - 700	Specific Power, [kg/MW]	1 - 5	Specific Power, [kg/MW]	2 - 3
Thrust Power, [GW]	68.7 - 49,200	Thrust Power, [GW]	0.2 - 56	Thrust Power, [GW]	1 - 70
Thrust, [N]	1,700 - 16,700,000	Thrust, [N]	1,700 - 11,000,000	Thrust, [N]	3,500,000 - 5,000,000
Engine Mass, [kg]	200 - 58,000	Engine Mass, [kg]	1,000 - 6,830,000	Engine Mass, [kg]	30,000 - 200,000
Mass Flow, [kg/s]	3.8 - 126	Mass Flow, [kg/s]	1 - 1,000	Mass Flow, [kg/s]	30 -100
Δ V, [m/s]	7,850 - 21,000	Δ V, [m/s]	17,000	Δ V, [m/s]	17,000

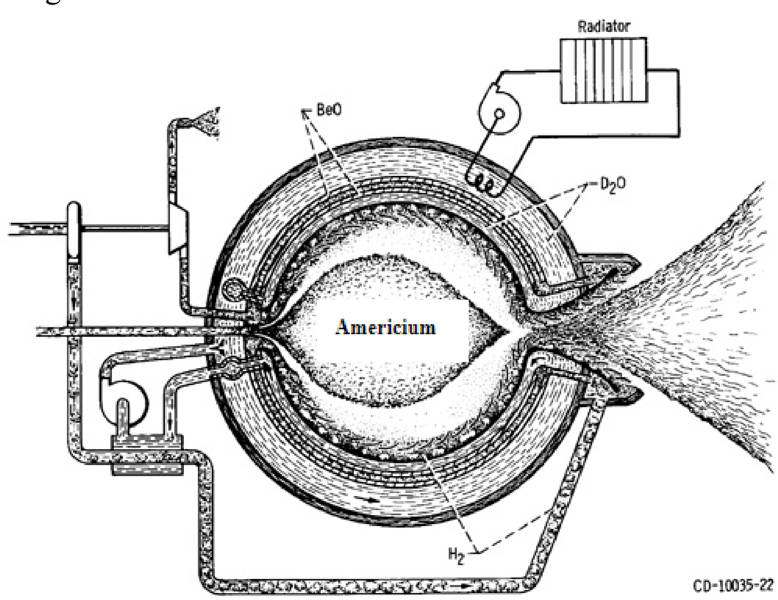
Table 4: Comparison of different NTR

Due to the nature of our mission the only feasible way to Saturn in a timely manner is to use NTRs. The reason the GCR was chosen was based on the excellent specific impulse with relatively low weights, ample power for auxiliary systems, and is not limited to a specific type of propellant. The table above shows a comprehensive list of operating parameters for solid-core, liquid-core, and gaseous-core respectively. Because our mission consists of multiple trips between various locations around Saturn our reactor has to have enough thrust to move high payload masses, has to be able to supply ample power for all other systems, and has to be able to use a diverse array of fuels while on the mission. The gaseous core reactor has a reasonable delta-v for our purposes while not being terribly massive, therefore the application of GCR is extremely attractive for extensive solar exploration.

Until a critical mass is reached for the chain reaction to initiate, gaseous fissionable fuel is injected into the reaction chamber. After that, hydrogen is permitted to enter the chamber from the walls and move to the center, where it heats and is ejected from the nozzle. The reaction is kept in a vortex designed to reduce fuel loss out the nozzle. The purpose of the fission reactions is to heat the propellant through thermal radiation. However, because hydrogen is more or less completely transparent to thermal radiation, the walls of the reaction chamber are struck by the thermal radiation. Specialized equipment is used to "seed" the propellant with material that is opaque to thermal radiation in order to correct this. The majority of reports point to tungsten dust as the primary seeding material, with tungsten dust having a particle size similar to that of smoke [1]. All but 0.5% of the thermal radiation is absorbed by the seeding, which then heats the hydrogen propellant through conduction. The 0.5 percent must be tolerated by the chamber walls.

According to some estimates, a reactor's chamber can withstand waste heat for up to 100 megawatts per square meter before it is destroyed[2]. This places the maximum specific impulse for the majority of designs at about 3,000 seconds. The specific impulse can be increased to around 7,000 seconds by including heat radiators to cool the reaction chamber walls and the moderator surrounding the reaction chamber. The disadvantage is that the necessary heat radiator gives the engine a lot of extra mass. A typical percentage is that the radiator makes up about 65% of the total mass of a gas core engine with radiator.

According to this report [3], switching to fuel made of americium-241 can reduce the chamber size by 70%. The Am-241 has the disadvantage of costing around $1,500 USD per gram, making every gram that escapes from the exhaust without being burned a financial disaster. The report calculates that the spacecraft would require about 2,000 kilograms for a 6-month trajectory.



The figure depicts the "open cycle" or "porous wall" gas core rocket. It has a primarily spherical shape and is made up of an inner porous liner, a neutron reflector/moderator region, and an outer pressure vessel. Because of its high operating temperature and compatibility with hydrogen, beryllium oxide (BeO) is chosen as the moderator material. To reach a critical mass, the open cycle GCR needs a plasma with a relatively high pressure.The gaseous fuel is sufficiently dense at these pressures for the stopping distance of a fission fragment to be equal to or less than the size of the fuel volume contained within the reactor cavity. Following ducting through the outer reactor shell, hydrogen propellant is injected through the porous wall with a flow distribution that leaves the cavity's central fuel region largely stationary and non-recirculating. However, along with the heated propellant, a negligible amount of fissionable fuel is also exhausted (between 1/4 and 1% by mass of the hydrogen flow rate).

We assume that only 7% of the reaction energy Prx makes it to the solid, temperature-limited part of the engine and that the remaining energy is transformed into jet power at an isotropic nozzle expansion efficiency of eta. Using the relationships (m-dot) between Isp, reactor power, and propellant flow rate [2]:

$$P_{rx} = (4.9 \times 10^{-5}) \frac{\dot{m}_p * \mathrm{I}_{sp}}{0.93 * \eta}$$

To calculate the pressure required to have a critical mass in the engine the following equation was used[1]:

$$\text{Pressure} = 14.6 \frac{M_c^{1.385} \times F^{0.383} \times I_{sp}^{0.383}}{D^{4.54} \times V_F^{1.51}}$$

where P is the reactor pressure in atmospheres, Mc, is the critical mass in kilograms, F is the engine thrust in newtons, Isp is the specific impulse in seconds, D is the reactor cavity diameter in meters, and VF is the fraction of the reactor cavity filled with fuel.

For calculating the chamber size a back of the envelope calculation was performed [4].

$$R_c = \sqrt{Q} * A_f,$$

where

$$A_f = \sqrt{\frac{1}{E_l} \times \frac{1}{4\pi}}$$

A_f is attenuation factor, E_l is maximum heat load (MW/m2), H is reaction chamber waste heat (megawatts), Rc is reaction chamber radius (meters).

Life Support

Responsible: Emeline Hanna

Current Life Support Technologies

The International Space Station (ISS) carries out research and testing of Environmental Control and Life Support Systems (ECLSS). Due to reliability and readiness of the ISS systems, these were initially considered for this mission. However, after comparing these to bioregenerative systems, it was clear that bioregenerative systems were a more viable life support solution. This is due to the fact that ISS ECLSS is currently heavy, requires a lot of maintenance, and requires large amounts of volume. For example, the CO_2 scrubbers on the ISS are very maintenance intensive and using the Sabatier CO_2 reduction reaction would require onboard venting or storage of CH [Ekart 1996]. Additionally, the ISS uses dehydrated, packaged food, and each astronaut uses 0.83 kilograms of food per meal each day [Allen]. The calculation below shows the weight required to provide packaged food for one astronaut for one way of the mission.

$$0.83\frac{kg}{meal * day} * 3meals * 6years * 365.25\frac{days}{year} = 5588kg$$

For only a one-way trip to Saturn, the mass required to provide food for three astronauts would be roughly 17,000 kg, which is not a feasible option. Furthermore, packaged food has a limited shelf life and the nutrients will degrade over time. Therefore, after comparing these systems to a biogenerative system, it was determined that a biogenerative system would be able to support life while requiring less space, mass, and maintenance, which were the most important measures (ranked higher than being an existing technology or cost).

Bioregenerative Life Support

Bioregenerative technologies are especially valuable in providing ECLSS due to their adaptability and closed-loop capacity in dynamic environments. Photosynthetic-based systems are an attractive solution to atmosphere revitalization. Algae has a very high photosynthetic efficiency, and therefore could provide robust ECLSS for long duration deep space missions. On board a spacecraft, “algae biological functions can revitalize the atmosphere, treat waste water, and produce biomass with potential to supplement crew nutritional needs; while the culturing media can be used as a coolant within a thermal control loop and augment the radiation shielding” [Matula 2019].

The required aspects of life support can be broken down into five parts: air revitalization, food production, water recycling, waste management, and gravity creation. The first four of these requirements will be satisfied by an algae based life support system.

Past Algae Experiments

Using algae for air revitalization to provide ECLSS capabilities has been investigated since the 1960s, but it is only recently that the technology needed to grow and illuminate the algae has become efficient enough for the possibility of space travel with the use of LEDs. In the 1960s, the Soviet Union constructed closed-loop bioregenerative air revitalization systems, called BIOS-1 and BIOS-2. During 1969, three people lived in a system for a full year, during which time their oxygen and water were fully regenerated. They found that 8 m^2 of photosynthesizing algae could scrub the carbon dioxide and produce the oxygen needed for one person. The system had a volume of 315 m^3 and used 400 kW of electricity. Additionally, in the third iteration, BIOS-3, was split up into four 79 m^3 sections. The algae compartment had three *Chlorella* cultivators which produced up to 800g/day of dry algae biomass. There were two plant growth chambers hydroponically growing 17 m^2 of wheat and 3.5 m^2 of vegetables in trays, which produced 1000 L/day of O_2. The plants provided 70% of the caloric requirement. Waste wash water and some urine was added to the wheat nutrient solution. Transpired water and moisture from the plant growth chambers were condensed and purified [Ekart 1996].

In 2017, an algal photobioreactor studying part of the Micro-Ecological Life Support System Alternative (MELiSSA) project was flown on the ISS, and "resulted in comparable oxygen production rates to the ground controls using *Arthrospira*". Additionally, "the University of Stuttgart flew a membrane-based algal photobioreactor in 2019 to demonstrate integration with a life support rack on the International Space Station (ISS)" [Matula 2021]. Research on algae photobioreactors is ongoing due to their possible development as a life support system.

Life Support Design: Algae and Growing Chambers

For this mission, there will be three main chambers on the ship used for life support systems. One of these chambers will house an algae photobioreactor, and the other two will be hydroponic growing chambers. The algae chamber will hold 8 m^2 of algae surface area, which will absorb the CO_2 and produce the O_2 for one person [Ekart 1996]. To provide the same lighting conditions as used in BIOS-3, 800 W of power is needed to power LED lights, using the fact that the iGloEZ Plant Grow light requires 20 W to provide light to about 0.2 m^2 of space [Amazon].

The growing chambers will include about 51 m^2 of growing surface area (the reasoning of which is explained more below in the food section). Assuming the plants will produce at the same rate as in BIOS-3, this will create about 3.5 kg of oxygen per day.

$$51m^2 * \frac{1000L}{20.5m^2 - 1day} * \frac{0.84kgO_2}{588LO_2} = 3.5kg/day$$

Since humans need 0.84 kg of oxygen a day [Starr 2015], and the algae is already compensating for one astronaut, there will be 1.82 extra kg of O_2 each day. Since oxygen toxicity is a concern, this oxygen must be very carefully tracked. Sensors will be monitoring oxygen levels, and any extra oxygen will be stored in "crystalline materials that can bind and store oxygen in high concentrations" [University of Southern Denmark]. Heating up this crystalline material made from cobalt will cause it to release the oxygen, allowing the astronauts to have access to extra oxygen if they ever need it. Since 10 L of the material would absorb all of the oxygen in the room, the mission would only require a very small amount [University of Southern Denmark]. The astronauts could absorb the necessary amount of oxygen in this crystalline material, and release it using the vacuum of space or by heating up, and continuing this process for the duration of the mission.

The food needs of the astronauts will be fulfilled by algae and our growth chambers. A human needs around 3000 calories per day, which should include "2000 g of water, 470 g dry weight of various carbohydrates and fats, 60 to 70 g dry weight of proteins, and adequate quantities of various minerals and vitamins" [Globus]. Spirulina will be used in the algae photobioreactors due to its high nutritional value. In 7g of Spirulina, there are 20 calories, 0.5g of fat, 1.7g of carbs, and 4g of protein [Nutritionix]. Therefore, eating 122.5 g of Spirulina will provide 70g of protein and 38.5 g of fats and carbohydrates.

$$\frac{0.5g - fats + 1.7g - carbs}{7g} * 122.5g = 38.5g - fats - and - carbs$$

Each astronaut will eat 122.5 g of Spirulina each day, meaning that 431.5 g of carbohydrates and fats must come from the growing chambers each day per person, or about 1295 grams of dry biomass per day. The Biomass Production Chamber (BPC), a closed environment with an area of 20 m^2 used for performing crop growth tests, found that the best rate of production was 12.6 g/(m^2 day) for edible biomass [Advances of Space Research 1996].

An assumption is made here that by the time this mission occurs, the efficiency found in the Biomass Production Chamber will be doubled, either through improved practices or choosing higher yield crops. Therefore, a crop growing area of about 51 m^2 is needed.

$$\frac{1295g}{day} * \frac{m^2 g}{12.6 * 2} = 51.4m^2$$

The crops will include high yield vegetables such as tomatoes, soybeans, lettuce, potatoes, and nutrient dense microgreens, and peppers to provide more variety and taste to the food. The non-edible parts of the grown plants will be dried and then burned to create CO_2 using a very hot incinerator, so as not to create any smoke. Allowing for the plants to have 1.5 times the LED lighting of the algae bioreactor, this will require 7650 W.

$$51m^2 * \frac{800W}{8m^2} * 1.5 = 7650W$$

This results in the estimated total of power usage of 8.5 kW. The total power usage was given as 8.7 kW to allow for a small buffer for monitoring equipment.

The growing chambers will generate water to sustain the astronauts. Using the condensed water data from the Biomass Production Chamber, about 201 kg of water could be condensed from the growing chambers each day [Advances of Space Research 1996].

$$51m^2 * \frac{94700kg}{20m^2 - 1200days} = 201kg/day$$

This is more than enough to sustain three people, since it is estimated that people need about 30 kg of water per day [Wheeler]. Therefore, the water can keep cycling through the growth chamber and algae bioreactor systems, and the air in the growing chambers can be condensed when necessary.

Essentially all waste will be recycled. Waste wash water can be added to the nutrient solution used in the growing chambers to grow the plants hydroponically. Solid crew wastes will be heated to extract water and irradiated with ultraviolet light to kill any bacteria. This solid waste will be added with the liquid wastes to the algae tanks. The urea and nitrogen compounds in urine are a source of nutrients that allow the algae to survive [Tuantet 2019]. The algae needed for food will be collected once a day, and the solid and liquid human wastes will be added immediately afterward. This should allow for enough time for the algae to consume all of the wastes, so there is no contamination at the next algae collection. If there are any other wastes, they could be jettisoned, although this should be a very small amount of waste.

It can be estimated that each person needs at least 17 m^3 of volume for their personal space [Joosten 2007]. As a reference point, the Bigelow Expandable Activity Module (BEAM) is an inflatable habitat that NASA has been considering as a habitation concept, due to the attraction of taking up very little space at launch and weighing less than a rigid structure. It weighs 1360 kg and provides about 16 m^3 of habitable volume.

The life support system is estimated to need at a minimum six BEAMs to hold the algae photobioreactor, growing chambers, and space for the astronauts. The mass of the life support system and structure was calculated to be 8500 kg, which includes the mass of six BEAMs and 340 kg for the algae and plant growth chambers.

Past Artificial Gravity Experiments

Spending a prolonged amount of time in microgravity has many negative consequences. Effects include fluid redistribution and loss in blood volume, which leads to an increase in resting heart rate and a decrease in physical health. Additionally, severe muscle damage can occur from lack of use and bones begin to dissolve due to the large amount of calcium lost [Hall

1994]. Artificial gravity can be used to counteract the effects of microgravity. There have been some studies on artificial gravity showing that creating artificial gravity using centripetal force counteracts microgravity effects. For example, in 1977, the Soviet satellite Cosmos 936 centrifuged rats. As compared to non-centrifuged control animals, the centrifuged rats had longer lifespans, preserved bone structure and minerals, and a reduced destruction of red blood cells due to the centrifugation. Additionally, on the space shuttle flight STS-90, centrifuged crew members had preserved otolith-ocular reflexes and normal blood pressure, heart rate, and vasoconstrictive variations, as opposed to the control crew members who were not centrifuged [Globus 2017].

Life Support Design: Gravity

For this mission, a rotating flywheel will provide artificial gravity for the astronauts. The equation for centripetal acceleration can be found in the following equation, where a is acceleration (meters per second) and r is the rotation radius (meters). A conversion was done between the rotation rate in meters per second, w_{rad}, to the rotation rate in rotations per minute, w_{rpm}.

$$a = \omega_{rad}^2 r = (\frac{2\pi}{60}\omega_{rpm})^2 r = 0.011\omega_{rpm}^2 r$$

A rotation rate of 6 rpm is acceptable for humans, given they have had some training [Globus 2017]. To determine the radius needed to provide 1 g of artificial gravity using a rotation rate of 4 rpm, the above equation was solved for the radius, and the numbers were plugged in, as calculated below.

$$r = \frac{a}{0.011\omega_{rpm}} = \frac{9.81m/s}{0.011 * (6rpm)^2} = 24.8m$$

Therefore, the rotating flywheel will have a radius of 24.8 m to provide 1 g of artificial gravity, which will prevent the many negative effects of spending time in microgravity. Although this is a large radius, the flywheel can be constructed so that there are two opposing structures that spin, which would lower the amount of volume needed.

Radiation Shielding

Responsible: Erika Hathaway

Radiation Environment

The space radiation environment on a journey to Saturn consists of three regions: near-Earth, interplanetary space, and near-Saturn. In these spaces, trapped charged particles, solar particles, and galactic cosmic radiation pose a threat to human life and electronics. In addition, the on-board nuclear reactor creates a source of radiation that must be shielded against. Throughout the mission, the crew on-board should be exposed to radiation no higher than the recommended for a career astronaut: 600 mSv [NASA Standard 3001 Volume 1]. The yearly dose limit for a terrestrial radiation worker is 50 mSv. The non-shielded radiation levels for each region are explained below:

- **Near-Earth**

  The average dose as measured by R3DR-2 instrument onboard the ISS during 2014-2016 [Dachev et al., 2017] is:

  $$\left(71.6\,\frac{\mu Gy}{d}\right)_{GCR} + \left(567\,\frac{\mu Gy}{d}\right)_{IRB} + \left(278\,\frac{\mu Gy}{d}\right)_{ORB} + \left(9\,\frac{\mu Gy}{d}\right)_{SEP} = 925.6\,\frac{\mu Gy}{d}$$

  where sources are galactic cosmic radiation, trapped particles in the inner radiation (IRB) and outer radiation belts (ORB), and solar energetic particles (SEPs), in units of microgray per day. The calculated average equivalent dose rate in microsievert per day (Q=3.5, 1.3, 1, 1.3 respectively) is:

  $$\left(250.6\,\frac{\mu Sv}{d}\right)_{GCR} + \left(737.1\,\frac{\mu Sv}{d}\right)_{IRB} + \left(278\,\frac{\mu Sv}{d}\right)_{ORB} + \left(11.7\,\frac{\mu Sv}{d}\right)_{SEP} = 1.2774\,\frac{mSv}{d}$$

  For 90 days (approximately 3 months), this amounts to 114.966 mSv, or ~20% of the career limit for an astronaut.

- **Interplanetary**

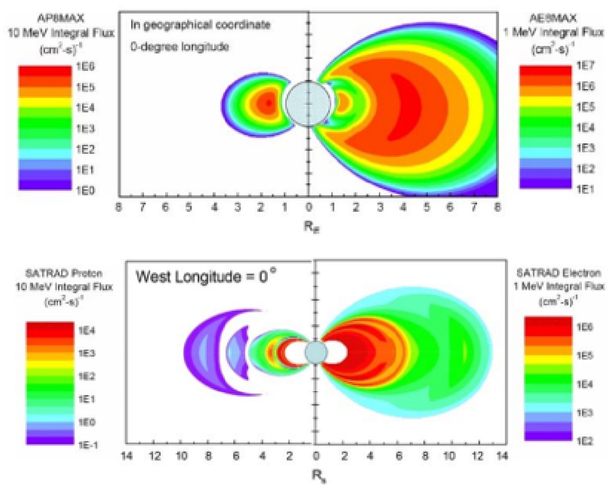
  During travel between planets, the radiation environment consists of SEPs and galactic cosmic radiation (GCRs). As the spacecraft moves away from the Sun, the amount of SEPs decrease and GCRs increase; this balance is adjusted depending on if the mission occurs during solar activity maximum or minimum. The expected effective dose from GCRs and SEPs seen at 1 AU is calculated by Dobynde [2021] to be about 62.5 cSv/year at solar minimum and 42.5 cSv/year from the January 2005 space storm: this totals to be 105 cSv/year (2.87 mSv/day).

  $$\left(62.5\,\frac{cSv}{yr}\right)_{GCR} + \left(42.5\,\frac{cSv}{yr}\right)_{SEP} = 105\,\frac{cSv}{yr} = 2.87\,\frac{mSv}{d}$$

Saturn is at 9.6 AU; within this distance, Voyager 1 and 2 have found that the GCR flux varies within the same order of magnitude. Taking into account that SEPs would be spread out over an area nearly 9 times larger (If SEPs were released radially over a quarter of a circle, area scales by ~diameter/4) than at Earth, the effect of GCR and SEP balance from Earth distance can be considered to cancel out.

- **Near-Saturn**

Saturn's magnetosphere extends between 16 and 27 R_S (1 Saturn radii, R_S is about 9.5 R_E), within which the space environment is dominated by the planet rather than the sun as for interplanetary space. Saturn's moons orbit both within and outside this boundary: Enceladus at 4 R_S, Titan at 20.3 R_S, Hyperion at 24.6 R_S, Iapetus at 59.1 R_S, and Rhea at 214.9 R_S. Although Saturn's magnetic field is nearly 8000 times stronger than Earth [Schombert 2019] implying more trapped charged particles, the radiation environment is significantly less severe than Earth's, as shown in the figure to the left. Because of Saturn's far distance from the sun, the neutral particles produced by Enceladus are ionized slowly, and often absorbed by Saturn's various moons and rings, reducing the amount and energy of charged particles in the environment. Comparing the AP8MAX model of Earth's radiation environment with the SATRAD model of Saturn's radiation environment suggests that shielding meeting astronaut needs in the near-Earth environment will satisfy shielding needs in the near-Saturn environment.



- **Nuclear Reactor**

The exact potential radiation emitted by the reactor is unknown due to the generalization of the reactor design. Most radiation will be shielded to be contained within the reactor with minor unavoidable leakage. To estimate the threat, the equivalent dose was calculated, assuming only damage from neutrons and gammas:

$$H = (QD)_n + (QD)_\gamma$$

Each dose was weighted by a radiation factor, 20 for neutrons and 1 for gamma. The dose was calculated as:

$$D = (count\ [\frac{\#}{m^3}])(E\ [J])(Human\ area/mass)(day\ [s])$$

Several assumptions are made, based on NERS 579 HW #1: the reactor is assumed to leak 1%, neutrons and gammas have energy of 2 MeV and 1MeV respectively, and neutrons have a count one order of magnitude larger than gammas.

The reactor has a neutron flux of 10^{16} counts per meter cubed, and an astronaut is generalized to have an area/mass ratio of 0.015 (0.75 m^2 for 50 kg). With no shielding and the astronauts directly adjacent to the reactor, the equivalent dose is deadly:

$$H = (20 \cdot D)_n + (1 \cdot D)_\gamma = 103.809 \frac{MSv}{d}$$

$$D_n = (10^{16})((2 \cdot 10^6)(1.602 \cdot 10^{-19}))(\frac{0.75}{50})(24 \cdot 60 \cdot 60)$$

$$D_\gamma = (10^{17})((1 \cdot 10^6)(1.602 \cdot 10^{-19}))(\frac{0.75}{50})(24 \cdot 60 \cdot 60)$$

In shielding against outside sources of radiation, being able to shield against radiation in the interplanetary region (the largest radiation source) should satisfy needs to shield in the near-Earth and near-Saturn environments. For a 15 year mission, the total dose is nearly 27 times the career limit of an astronaut. In addition, shielding against the reactor is also necessary.

$$15\, years \cdot 105 \frac{cSv}{yr}_{Interplanetary\ (larger\ than\ Near-Earth,\ Near-Saturn)} = 15.75\, Sv = 26.25 \text{ x } Limit$$

Radiation Requirements

The requirements derived from the radiation environment and applicable to this subsection are detailed before. The subsystem has main consequences for power and mass of the spacecraft, and positively affects life support.

Requirement	**Reasoning**
The shielding shall reduce the background space radiation to acceptable limits.	
The allowable dosage shall be 525 mSv for the mission.	This is 87.5% of the exposure allowed for career astronauts and includes a safety margin.
Shielding shall be designed to be low-mass.	General good mission planning reduces SWaP.
The shielding shall reduce the reactor radiation to acceptable limits.	
The allowable dosage shall be 75 mSv for the mission.	This is 12.5% of the exposure allowed for career astronauts and includes a safety margin.
Passive shielding shall shield 87.5 %	Passive shielding will protect crew while power

of the exposure in the near-Earth environment from background radiation.	systems are not at nominal operation.
Additional shielding between the reactor and crew quarters shall reduce radiation to below 12.5% of career limits during the entire mission.	Passive shielding will protect crew during the mission from the power source.

Shielding Design

The shielding aboard the KOIOS spacecraft includes both traditional material shielding (passive) and dynamic shielding (active). Passive shielding has been designed to protect crew against radiation leakage by the reactor, and additionally against exposure in the near-Earth environment. Active shielding requires power, and will be incorporated after the mission has begun, and is designed to protect crew against exposure in the interplanetary space environment.

Passive Shielding

Passive shielding is designed to shield crew against LEO space radiation and radiation from ny potential reactor leakage. To account for a mere 12.5% (75 mSv) of astronaut career limit exposure, the dosage allowed from the reactor must be as follows, meaning that most of the radiation leaked must be shielded:

$$20\left(1.3\cdot10^{3}\frac{n}{m^{2}}\cdot 2\ MeV\ in\ J\cdot\frac{0.75\ m^{2}}{50\ kg}\cdot 1\ day\ in\ s\right)_{n}+\left(1.3\cdot10^{4}\frac{\gamma}{m^{2}}\cdot 1\ MeV\ in\ J\cdot\frac{0.75\ m^{2}}{50\ kg}\cdot 1\ day\ in\ s\right)_{\gamma}$$

$$\approx 13.49\frac{\mu Sv}{d}\approx 73.89\frac{mSv}{15\ years}$$

To achieve this level of shielding, in addition to what is already designed for current systems like the ISS for LEO, more material must be concentrated in between the reactor and crew quarters. This can be achieved with 2.6 m thick graphite shielding against neutrons and 5.5 m thick shielding against gammas. This calculation was achieved with a graphite build-up factor $B = 2.2$ and $\Sigma_R = 0.1\ cm^{-1}$ [Qadr, 2019], and the thickness required for neutrons was approximately doubled for shielding against gammas (NERS 579 HW #1).

$$x_n = \frac{-1}{\Sigma_R} ln\left(\frac{I}{I_o B}\right)$$

5.5 m^3 graphite shielding weighs about 12463 kg. It is important to note that this number may change; the structure of the KOIOS spacecraft is largely undecided, and the free space between the reactor and the crew quarters will be a major factor in reducing radiation levels without the need for additional mass.

Active Shielding

To mitigate the premium cost of weight, active space radiation shielding has become a recently active field of development. Inspired by Earth's ability to sustain life from space radiation with a dynamic magnetosphere, active shielding revolves around applications of various electric and magnetic fields that interact with the environment around them. A magnetic field of similar structure and scaling to Earth's is difficult to produce but recent research has suggested a possible method of artificially expanding a magnetic field: known as Mini-Magnetospheric Plasma Propulsion (M2P2) [Winglee et al.], the concept provides a solution to both shielding and propulsion. Because the KOIOS system already has a satisfactory propulsion method and system, the energy generated in the M2P2 system will be returned to the system to maintain the generation of the EM field and plasma needed to expand the magnetosphere to interact with the solar wind.

The concept is not complex: a small magnetic field is generated by a super-magnet. Plasma is injected into the small magnetic field, which begins to expand following the frozen-in flux condition, until it reaches a balance with the solar radiation outside the spacecraft, forming a giant 'bubble' safe-zone for the spacecraft and crew. The M2P2 system is able to deflect particles following the equation below, and is thus able to shield against GCRs, some of the highest energy particles at ~1 GeV.

$$v_{\perp} \leq \frac{qB_0 R_m}{2M}$$

A preliminary sizing suggested by Winglee is the 32kW powering of a several decimeter diameter supermagnet, which will over a short period of time, extend the field limit to nearly 100 km. This low SWaP system removes the need for extreme massive shielding on all sides of the crew quarters.

System: Thermal Power and Electrical Subsystems

Responsible: Evan Litch

Previous Power & Thermal Systems

Powering spacecraft for long term deep-space missions has been achieved via several Radioisotope Thermoelectric Generators (RTGs), such as for the Voyager 1 & 2 missions.[1] However, past deep-space missions were unmanned and thus had lower power requirements. When developing a manned deep-space mission, factors like life support, advanced propulsion, etc.must be considered. These factors not only impact the power requirement of the craft, but also add to the thermal load. Looking at the power and thermal systems the ISS employs, they are able to use solar and radiator grids for the power and thermal regulations for the station. The solar system will not be similar to this mission since it utilizes a nuclear reactor to provide thrust, and the residual heat will power the entire craft. However, the thermal system should not be too different as the Active Thermal Control System (ATCS) on the ISS.[2] For our mission to deploy a space station around Saturn, the craft will need to have similar systems for this “indefinite” mission.

Mission Power & Thermal Requirements

The thermal and power systems will need to be able to dissipate ~5 MW of heat from the reactor, as well as ~5 kW, which was included in the calculations but is rather insignificant to the reactor cooling requirement. The 5 MW of reactor heat is an approximation from the heat not absorbed by the hydrogen propellant from the gas core reactor. This is assumed via the moderator cooling as well as the NaK cooling loop that will reduce the heat and bring the temperature from 3000-4000 K to a 1200 K NH_3 inlet. A basic process flow diagram was created to highlight the power and thermal subsystems in Figure A.

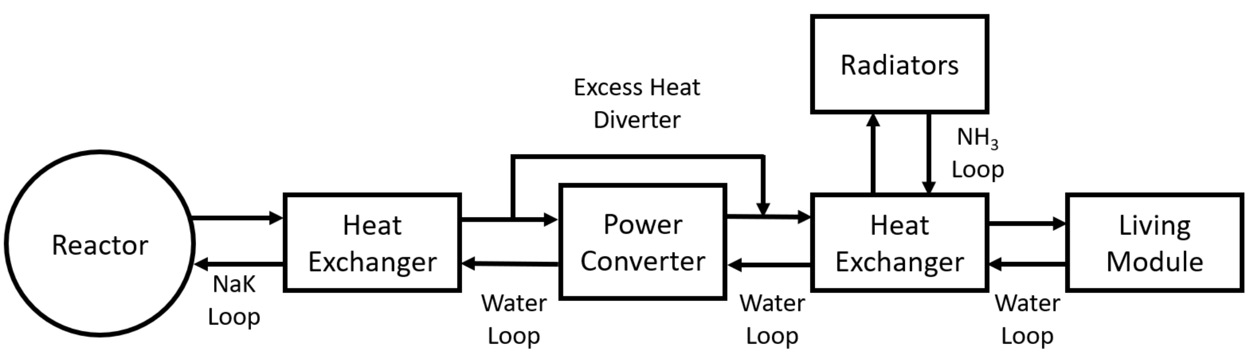


Figure A: Process flow diagram of the power & thermal control subsystems between the reactor and living module.

The power converter consists of a thermionic generator with a diverter pipe to not oversaturate the power conversion system with heat to reduce degradation of the thermionics. This system needs to produce ~65 kW of power, 32 kW for the active radiation shielding and 33 kW for all other systems requiring power. These other systems are the electronics to operate the entire craft, lift support, communications, etc. and their power requirements were estimated from the ISS power requirements scaled by volume.[3],[4]

From the power generator the rest of the heat is transferred to the NH_3 radiator loop where most of the heat is removed. If any heat is needed for the living module, the similar ATCS will reduce the flow through the radiators to bring that heat to the module. The 5 kW of heat from the radiative energy the human body gives off along with the average waste heat electronics produce is then collected and sent back to the radiator loop to be rejected. [5] The thermal control system for spacecraft is built to handle the highest residual heat from the reactor while still being able to provide power post burn.

Power & Thermal Design

For some background on how all values were calculated, the radiator sizing was found using the following equations:

$$\eta = 1 - \frac{T_{Cold}}{T_{Hot}}$$

$$W_{rad} = \left(1 - \frac{T_{Cold}}{T_{Hot}}\right)P = \eta P$$

$$P_{rad} = \varepsilon\sigma T_{Cold}^{4} A$$

Where η is the radiator efficiency, P is the heat entering the radiator array, W_{rad} is the heat actually dissipated by the radiator, and assuming a radiator emissivity of 0.85.

These equations were used to calculate the area of the radiator array and from the specific weight of the aluminum vapor fin-tubes.[6] This was the lightest radiator design for the particular study covered by Haller, comparing central fin-tube, double fin-tube and vapor fin-tube.

Power conversion was estimated from the general efficiency for each different converter. Then the power densities were found for each, excluding the gas dynamic (Brayton cycle), to determine the space each takes up with the weight finally being calculated from the specific weight of each.[7]-[10] Those values can be found under Table 5, showing thermionic not only provides the highest efficiency, but also provides the lightest option.

Table 5: Power converter comparison[6]-[10]

Type	Efficiency	Weight (kg)	Radiator Area (m^2)	Radiator Weight (kg)
Thermionic	60%	878	457.9	9683
Thermovoltaic	40%	6500	457.9	9683
Thermoelectric	15%	2373	457.9	9683
Gas dynamic	30%	3900	461.0	9706

An interesting result was that the thermoelectric was the second lightest, even with the lowest efficiency due to its low specific weight. While each converter affects its own weight, the weight of the radiator array was minimally impacted by converter type. Since most converters are static, the power they produce is assumed to equal the heat removed from the system. However, the heat removed from all systems is two orders of magnitude smaller than the rest of the residual heat so the heat the radiators are removing is dependent on the excess reactor heat. Thus why the thermionic generator was chosen and our power converter.

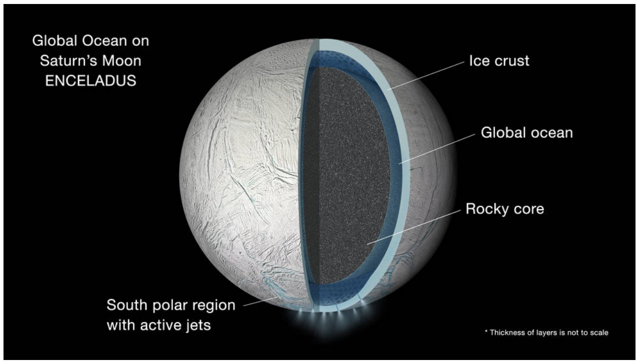
Mini Spacecraft

Responsible: Marco Daniel Acciarri

Once in Saturn's orbit, the space hub will deploy a "mini spacecraft" based on actual designs of Moon Landers from private contractors. The mini spacecraft will have the infrastructure needed to travel to the Enceladus Moon, land, extract ice from the surface and travel back to the space hub where the ice will be melted using wasted heat from the reactor (or additional heaters) and then converted to O_2 and H_2 through electrolysis. Then, the O_2 and H_2 will be liquefied and stored in tanks. The O_2 can be used as an emergency reserve for life support systems as well as the oxidizer for the chemical rocket aboard the mini spacecraft. The LH_2 can be used as propellant for the space hub as well as fuel for the mini spacecraft. Having a refueling system will allow it to extend the overall capacity and infrastructure of the mission, providing extra flexibility for logistics within Saturn's orbit and possibly, to other planets.

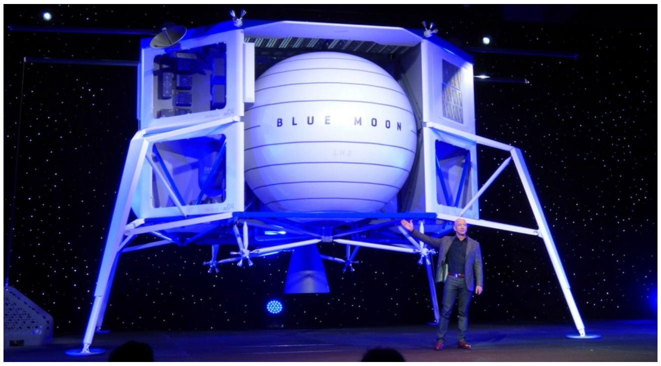
Enceladus has a 30 to 40 km layer of ice on its surface. The minimum temperature of the ice is around -200 C and the surface gravity is approximately 0.113 m/s2 (0.0113 g) [1]. The escape velocity from the moon is 0.239 km/s [1], making it attractive for extracting ice from the surface, since a relatively small amount of propellant would be needed for this operation.

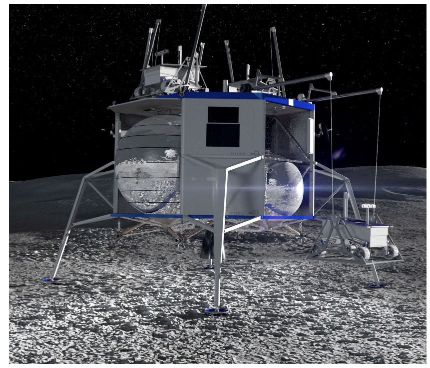
Sectional View of Enceladus Surface [2]



The "mini" spacecraft is based on private contractor designs for Moon missions where the payload is modified, in particular the Blue Moon concept from Blue Origin as shown in the following figures:

Photos and Artist representation of the Blue Moon Lander Concept [3]





The original specifications of the rocket are indicated in the following table [4].

Dry mass	10850 kg
Propellant mass	5550 kg
Payload mass	4500 kg
BE-3U engine	710 kN vacuum thrust
Propellant	LOx + LH2

In order to perform ice mining operations on Enceladus surface, the mini spacecraft needs two main modifications from the original "Moon Lander" design:

1. A subsystem for cutting the ice.
2. A subsystem for grabbing the ice and locating it in the payload bay of the craft.

Since the surface gravity of Enceladus is only ~0.01g, it makes it difficult to extract the ice from the surface using a mechanical approach like a normal excavation technique. A possible solution to this is a laser-based cutting system. Using a laser, the power used to cut the ice can be used more efficiently while minimizing the total weight of the subsystem. Here I propose an approach to fusion both subsystems using a CO_2 laser attached to a robotic arm, while the same robotic arm and a secondary one can grab the ice pieces and locate them in the payload bay. Having a laser at the end of a robotic arm with several degrees of freedom provides the flexibility of choosing the most efficient cutting angle and selecting the size of the ice blocks that will be cutted and located in the payload bay optimizing the volume available.

<u>Laser</u>

A CO_2 laser will be used at 10.6 μm, since that wavelength optimizes the energy absorption of ice and it would maximize the energy efficiency of the subsystem [5].

Sakurai et al. [5] showed that it is possible to use a CO_2 laser for ice drilling by conducting a laser-ice drilling experiment on earth.

As described in [5] the energy transfer in this system consists in four physical processes:

1. First, energy is used to raise the initial ice temperature to the melting point.
2. More energy is used to melt the ice in the cutting region (latent heat).
3. Part of the energy is lost laterally due to warming of the melt water formed between the ice and the laser.
4. Another fraction of the energy is lost due to thermal conduction outside of the borehole.

Here, a few differences arise from the considerably different environment where the ice drilling will be executed. First, the initial ice temperature in Enceladus is -200 C while in [5] was -20 C, meaning that more energy would be used for heating the ice to the melting point. However, once the ice is melted, the water will automatically vaporize due to the lack of atmosphere (and pressure) in Enceladus' surface. For this reason, no energy would be lost with this mechanism in comparison with [5]. The melted water accumulated in the hole caused a reduction of the drilling speed observed in [5], here that reduction would not occur since no water would be accumulated leading to the possibility of a larger drilling speed. For the mentioned reasons, the drilling speed of 0.8 mm/s [5] would be considerably increased. A commercial AL50 CO_2 laser with an intensity of ~ 50 W/cm2 , average 10 W of power consumption and 11 kg could be used [6].

<u>Robotic Arms</u>

The mini spacecraft would include two robotic arms that will carry the CO_2 laser and conduct the drill operation. After each block of ice is cutted, the robotic arms will grab the ice and locate it in the payload bay. Using a robotic arm with several degrees of freedom gives extra flexibility to cut different sizes and shapes of ice and increase the usability of the payload bay volume. Since the surface gravity of Enceladus is a small fraction of Earth's gravity, a single block of ice would weigh ~1.3% of its weight on Earth, reducing the requirement for torque and power of the robotic arms. The robotic arms can be based on robotic arms used for on-orbit robotics applications such as the STAARK robotic arm from REDWIRE private contractor as shown in the following figure [7]. Without including batteries, the manipulator, HDRMs and Robot Control Unit would have a weight of 32 kg each, totalling ~64 kg overall for this system.

<u>Weight, Propellant and Mass ratio</u>
Including the laser, robotic arms and Control Units the overall weight added to the mini spacecraft is ~ 75 kg. However, extra batteries will be required in order to power the laser, depending on the total drilling time and total mass of ice to extract. For this reason, an overall weight of 1000 kg is assigned to this system including also Li-ion batteries and all the electronic requirements as well as extra volume in the payload bay to locate all the ice (approx 10 cubic meters as it will be described next).
A mass analysis is shown next, describing the feasibility of using the mini spacecraft for ice mining operations and the potential amount of ice that could be extracted from Enceladus surface per unit mass of propellant used for both landing and departing from the Moon to the space hub for further electrolysis and fuel storage.
The design dry mass of the Blue Moon lander is 10850 kg, from which 4500 kg are the payload. Here, our "payload" would consist of the systems described before, needed for the ice mining operation (~1000 kg) and most of the payload bay of the lander would be empty in order to store the ice to be extracted. Therefore, the dry mass for the modified lander consists of 10850-4500+1000 kg = 7350 kg.
For the propellant mass analysis, enough fuel should be available to depart Enceladus with an extra weight of ice. We suppose that a mass of 10000 kg of ice should be extracted per mission.
In order to make an estimation for the amount of fuel needed I considered a total Δv of approximately 2 times the escape velocity of Enceladus (0.239*2 km/s). This is considering that the space hub should approach a "near" Enceladus orbit from where the mini spacecraft will be deployed.

Using the rocket equation, an Isp of 450 s, an initial dry mass Mi before landing on Enceladus of 7350 kg and a Δv that corresponds to the escape velocity of Enceladus as an approximation, the propellant needed to land on Enceladus is 409 kg giving a wet mass of 7759 kg. However, more propellant is needed to escape Enceladus' surface with an extra of 10000 kg of ice per mission and that propellant needs to be carried before the landing. This propellant is estimated using, again, the rocket equation and the same delta v with an initial mass of 7350 kg + 10000 kg(ice)=17350 kg, giving 915 kg of propellant. Then, the initial wet mass is updated including both the propellant needed for landing and departing Enceladus until a convergence is reached. The following table shows the first iterations until this algorithm converges to the final propellant mass needed, where Mi1 is the initial mass before landing, Mf1 is the mass after landing and before extracting the ice, $M_{propellant-landing}$ is the mass of propellant used to land, $M_{departure}$ is the mass before departing from Enceladus with 10000 kg of ice, Mf2 is the mass after departing (when the mini spacecraft arrives to the space hub) and $M_{propellant-departure}$ is the mass of propellant used to depart from Enceladus and reach the space hub.

Mi_1 (kg)	Mf_1 (kg)	$M_{propellant-landing}$ (kg)	$M_{departure}$ (kg)	Mf_2 (kg)	$M_{propellant-departure}$ (kg)
7759	7350	409	17350	16435	915
8726	8265	460	18265	17302	964
8777	8314	463	18314	17347	966
8779	8316	463	18316	17350	966
8779	**8316**	**463**	**18316**	**17350**	**966**

The propellant needed to land and then depart from Enceladus with the extracted ice makes a total of 1429 kg. This gives as a ratio mass extracted to propellant used of 10000/1429 = 7 per refueling mission showing that it is possible, at least conceptually, to extract more ice from Enceladus than the mass of (LOx + LH_2) needed to do the entire drilling operation and logistics.

The 10 tons of ice can then be liquified using a heater or waste heat from the reactor and converted to LOx and LH2 using an electrolysis module on board of the space hub. To make a simple approximation, the Elektron electrolysis module used at the ISS is taken as an example [8]. Each unit of the Elektron module weighs 150 kg, consumes 860 W of power and can produce ~ 1900 l (2.7 kg) of O_2 and H_2 per day. The difference is that in the ISS, the H_2 is vented out to space while here we would store it in tanks. For our mission, 20 Elektron units would be included totalling 2000 kg of weight with a power consumption of 7.2 kW if all the units are used at the same time. The output mass of this system would be 54 kg per day, needing 186 days to separate 10 tons of water into LOx and LH_2.

This can then be repeated as many times as needed to refuel the space hub and extend a mission or conduct several scientific missions to different Saturn's moons.

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