

# **SERENE: Saturn Enceladus Return Explorer with Nuclear Electric Propulsion**

Steven Oleson<sup>1</sup>, Elizabeth Turnbull<sup>2</sup>, Steven McCarty<sup>3</sup>, and Zachary Zoloty<sup>4</sup>

*National Aeronautics and Space Administration, Cleveland, OH, 44135, USA*

Thomas Packard<sup>5</sup> and Allison C. Fox<sup>6</sup>

*HX5, LLC, Brook Park, OH, 44142, USA*

Geoffrey Landis<sup>7</sup>, Lee Mason<sup>8</sup>, and Benjamin Abshire<sup>9</sup>

*National Aeronautics and Space Administration, Cleveland, OH, 44135, USA*

James Fittje<sup>10</sup>

*Science Applications International Corporation, Fairfax, VA, 20190, USA*

Anthony Colozza<sup>11</sup>

*HX5, LLC, Brook Park, OH, 44142, USA*

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<sup>1</sup> Lead, Compass Team, Mission Architecture and Analysis Branch, and AIAA Senior Member.

<sup>2</sup> Lead Systems Engineer, Compass Team, Mission Architecture and Analysis Branch, and AIAA Young Professional.

<sup>3</sup> Aerospace Engineer, Mission Architecture and Analysis Branch.

<sup>4</sup> Aerospace Engineer, Mission Architecture and Analysis Branch.

<sup>5</sup> Aerospace Engineer, Mission Architecture and Analysis Branch.

<sup>6</sup> Associate Researcher, Smart Sensing and Electronics Systems Branch.

<sup>7</sup> Researcher, Photovoltaics and Electrochemical Systems Branch, AIAA Associate Fellow.

<sup>8</sup> Senior Technologist, Aerospace Power Systems Division.

<sup>9</sup> Electrical Engineer, Power Architecture and Analysis Branch.

<sup>10</sup> Aerospace Engineer, Electric Propulsion Systems Branch, and AIAA Senior Member.

<sup>11</sup> Research Engineer, Photovoltaic and Electrochemical Systems Branch.

Ryan McDonough<sup>12</sup>, Bushara Dosa<sup>13</sup>, Shadi Zogheib<sup>14</sup>, and Benjamin Wozniak<sup>15</sup>

*National Aeronautics and Space Administration, Cleveland, OH, 44135, USA*

John Gyekenyesi<sup>16</sup>

*HX5, LLC, Brook Park, OH, 44142, USA*

Cassandra Chang<sup>17</sup> and Maya Havens<sup>18</sup>

*National Aeronautics and Space Administration, Cleveland, OH, 44135, USA*

A ‘quick’ Enceladus sample return mission concept was developed based on the scientist recommendations at the recent ‘Accelerating Space Science with Nuclear Technology Workshop’. The Nuclear Electric Propulsion spacecraft assumed a follow-on 40 kWe nuclear reactor using the demonstrated 40 kWe Fission Surface Power system, expected in the early 2030s. The NEP vehicle also utilized a set of NEXT-C ion thrusters as well as planned Artemis commercial launchers. By launching the 40 kW NEP vehicle on a Starship and adding the propellants of 15 tankers, the 27t probe could be sent on a direct trajectory to Saturn (no Earth or Jupiter flybys) where NEP was used for Saturn capture, spiral down, spiral up and return to the Earth. A small lander obtained the surface Enceladus sample. Using the 40 kW NEP provided a round-trip time of only 16.5 years. A second option was more attractive from a science perspective whereby the NEP vehicle would deliver a large, 6t chemical lander to low Enceladus orbit where it would grab and return a sample to Earth (similar to the recent Orbilander design but in reverse). After deploying the lander, the NEP vehicle would stay in Saturn space performing a moon tour by orbiting four more moons and mapping the large moon of Titan. This option took slightly longer (18.5 yrs) due to the chemical return leg limitations. Both options demonstrated the agility, payload capability, and sample return goals the workshop recommended. An all-chemical option with two stages was roughly analyzed but took 21.5 yrs and required a Jupiter gravity assist.

## 1.0 Introduction:

Nuclear electric propulsion has been considered for many planetary missions since the 1980’s and the SP-100 program [1]. The use of NEP for planetary science peaked in the early 2000s with the Jupiter Icy Moons Orbiter (JIMO) project termed the Prometheus Project [2] whereby a 100-kW-class NEP vehicle would visit three moons of Jupiter – culminating in an orbit around Europa. While cancelled due to changing NASA priorities, two of the major challenges for JIMO, paying for the reactor development (> \$4B FY03-05) and launcher availability (3 launches with chemical stages), may no longer be limitations due to two recent developments: the lunar Fission Surface

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<sup>12</sup> Electronics Engineer, Communications Systems Architecture and Analytical Studies Branch.

<sup>13</sup> Electronics Engineer, Advanced High Frequency Communications Branch.

<sup>14</sup> Electronics Engineer, Avionics Branch.

<sup>15</sup> Aerospace Engineer, Mission Architecture and Analysis Branch.

<sup>16</sup> Mechanical Engineer, Structural Mechanics Branch.

<sup>17</sup> Operations Research Analyst, Program Planning and Control Office.

<sup>18</sup> Aerospace Engineer, Mission Architecture and Analysis Branch.

Power (FSP) project, and Artemis, with its reliance on large, refuellable commercial launchers (e.g. Starship). If the reactor and power conversion system is used as is (40 kWe at the time of this design), the large development costs could be avoided, making the use of NEP for space science much more affordable. The large commercial launchers promise significant improvements on mass delivered to high C<sub>3</sub>s at advertised affordable prices. These two developments, combined with avoiding the high radiation environment of Jupiter, promise more affordable and potentially higher performing planetary science missions.

With its high Isp, an NEP vehicle can perform large  $\Delta V$  missions (10-20 km/s) without refueling. With a sufficiently high-power NEP vehicles should be able to perform these missions quickly. Such ‘quick’ missions were one of the scientist recommendations at the recent ‘Accelerating Space Science with Nuclear Technology Workshop’ [3]. A more complete list follows in the next section. One of the JPL studies shared sought to estimate how long, with various technologies, it would take to return samples from all the solar system bodies. Combining this with the decadal survey’s great interest in potential life at Enceladus prompted a study into how quickly an NEP vehicle, using an FSP derived power system and large commercial launchers could gather and return a sample from Enceladus.

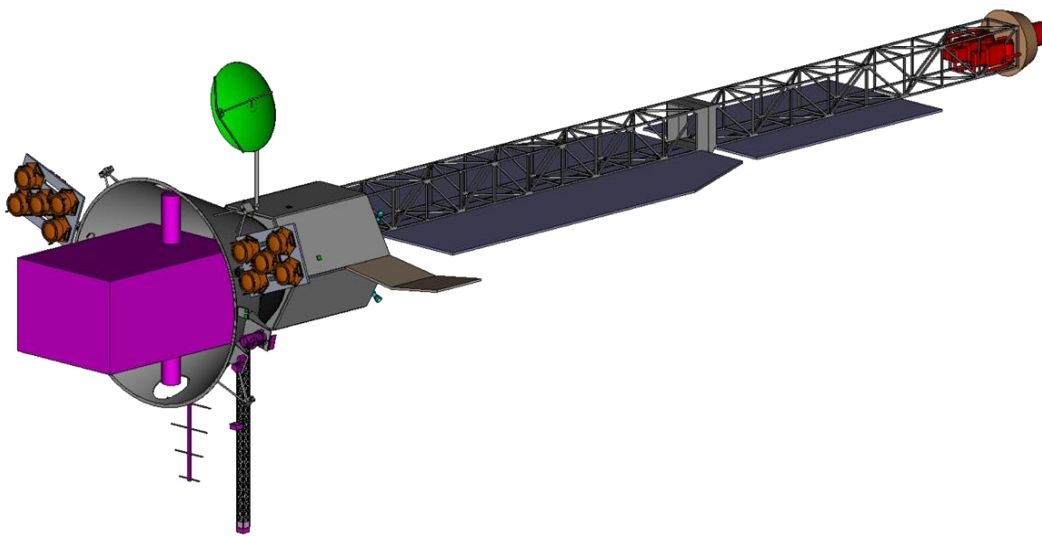


**Fig. 1 – Saturn Enceladus Return Explorer with Nuclear Electric propulsion (SERENE)**

## **2.0 Study Approach: Requirements, Assumptions, and Trades**

The results of the Accelerating Space Science with Nuclear Technology Workshop gathered which important attributes the science community was seeking from nuclear technologies. Surprisingly, increasing science instruments mass (to utilize the kW of NEP power when not thrusting) was not an objective due to the cost/kg of science instruments. Instead, delivering large landers (significant propellant loads), visiting many targets, and relaying large amounts of data were included. A summarized list of objectives for nuclear technologies are as follows:

- Use in concert with new large commercial rockets
- High-delta V destinations
- Enabling much higher capability orbiters and landers
- Significantly reduce trip times
- “Sample return from anywhere in the Solar System”
- Increase launch window flexibility by reducing or eliminating the need for gravity assists
- Increase communication rates
- Deliver ground penetrating radar systems at the icy moons/subsurface ice/liquid interface

- Enabling the delivery and landing of capable vehicles on the ocean worlds (Enceladus in particular) is a priority to support the search for life.
- Maximizing commonality between FSP and NEP subsystems (where possible), to help control costs and enable more rapid applications.

Nuclear systems should be designed for mission lifetimes of 10 - 20 years to ensure ample opportunities for science, even for the fast trip times to the outer planets. Nuclear architectures may enable missions to reach multiple destinations, though the long-term funding needs for such missions must be addressed.

A sample return from Enceladus addresses many of these objectives as will be seen. The mission concept was dubbed SERENE: Saturn Enceladus Return Explorer with Nuclear Electric propulsion and is shown in Fig. 1.

## 2.1 Requirements and Assumptions:

To simplify the study effort, the recent Enceladus ‘Orbilander’ design study was baselined to provide science objectives, instruments, and the lander to gather the Enceladus samples [4]. This allowed the Compass team to concentrate on the SERENE 40 kWe NEP mother spacecraft conceptual design.

As mentioned previously, the 40 kW FSP would be the baselined power system to reduce development costs. Changes to its shadow shield and a different radiator setup were made, along with the addition of a boom for reactor separation and to hold the radiators. A Starship launcher was assumed with the ability of tanker refueling in order to send SERENE on a high velocity departure from Earth.

A challenge to gather and return the Enceladus sample in less than 20 years was made based on an earlier study that was limited to a smaller launcher and lower power [5].

## 2.2 Trades:

Even by limiting the primary propulsion option to NEP there are a surplus of trades for an Enceladus sample return (ESR) mission. Some of the top-level ones are described here.

*NEP Propulsion:* Of greatest importance is what an NEP vehicle brings to benefit an ESR mission. Of course, it provides primary propulsion for quickly transiting interplanetary space to and from Saturn, but also it enables quickly spiraling down to Enceladus and back in a large gravity well. As will be seen, there are small moons that can be used to ‘gravity assist’ any spacecraft to move up and down in Saturn orbit. While a slower transit than using NEP, they do allow saving NEP propellant for other portions of the mission. Spacecraft normally benefit from utilizing a Jupiter gravity assist (allows for a lower  $C_3$  and thus more payload) however, for Saturn transits the Jupiter gravity assist only repeats every 20 years or so. Using NEP to remove this launch opportunity limitation was an objective of this study.

*Reactor kW of Power:* With the large amounts of power available when not thrusting, the NEP vehicle could provide two types of radar mapping (deep ice and terrain) as well as high power, small antenna data relay. Such a data relay capability removed the need for the lander to communicate directly back to Earth. Indeed, given the inclination of Enceladus, near the south pole (the preferred landing place for samples) would not see the Earth for long periods of time. With the relay support of the NEP orbiter no limitation on landing date would be needed.

*Number of Spacecraft:* Given the capability of the NEP vehicle, from a propulsive, power, and communications standpoint, the use of this vehicle for most, if not all, of the mission was sought. Given that the landing/ascent  $\Delta V$  for Enceladus, due to its small size, is quite small ( $\sim 100$  m/s each way), it might be possible to land the entire NEP vehicle with sufficient chemical propellant. However, given the normally long, dumbbell shape of an NEP vehicle and the radiation shielding, this option was left for further work. One offshoot option from landing the NEP would be to land a sufficiently large laboratory to do life investigations in situ.

As such, a small lander could be used to land, gather a sample and contextual information, ascend and re-dock with the NEP vehicle. As mentioned previously, the recent Orbilander concept provides a complete vehicle design for performing this part of the mission. In fact, given the Orbilander’s chemical capability to reach Enceladus using a large launcher from Earth, it was concluded that by taking the Orbilander (fully fueled) to Enceladus, it would have

sufficient propellant to return the sample to Earth. While this would require the NEP vehicle to deliver a ~6t wet lander to Enceladus orbit, the delivery of ‘much higher capability landers’ was one of the Accelerating Space Science with Nuclear Technologies workshop’s demonstrations.

This option for the sample to chemically return on the lander created the baseline mission option, whereby the NEP vehicle would deliver the chemical sampling lander to Enceladus orbit, and the chemical sampler would return the sample itself to Earth, freeing up the NEP vehicle for other activities in Saturn space. These activities included orbital science mapping of Enceladus, Tethys, Dione, and Rhea, with the final spiral down to map Titan from a low orbit. Such a mission would perform two New Frontiers (and Enceladus flyby and a Titan Orbiter), a Flagship (Enceladus lander), and a ‘super flagship’ (ESR) science mission all in one – enabled by NEP. While this became the baseline mission, an all-NEP ‘there and back’ sample return became an alternate option, one indeed that had the quickest sample return time (~16.5 yrs). A secondary benefit of using a chemical return leg option was to allow easy trades for an all-chemical, two stage mission for comparison to the NEP mission.

*Lander Power:* Since the baseline assumed delivering the Orbilander, powering the lander had already been solved by the use of two RTGs. This power would be necessary not just for surface operations but for the return to Earth. The backup, all-NEP-transit option allowed for a smaller lander due to a much smaller propulsion requirement. Battery power might be possible, eliminating RTGs (with their associated cost and availability limitations) from the mission. As an alternative for both lander options, a beamed power solution, using the kW’s of power from the NEP vehicle might be possible given the repeating overflights. A similar concept was recently published [6] showing the feasibility of such an approach. However, time constraints required pushing the beamed power option to further work.

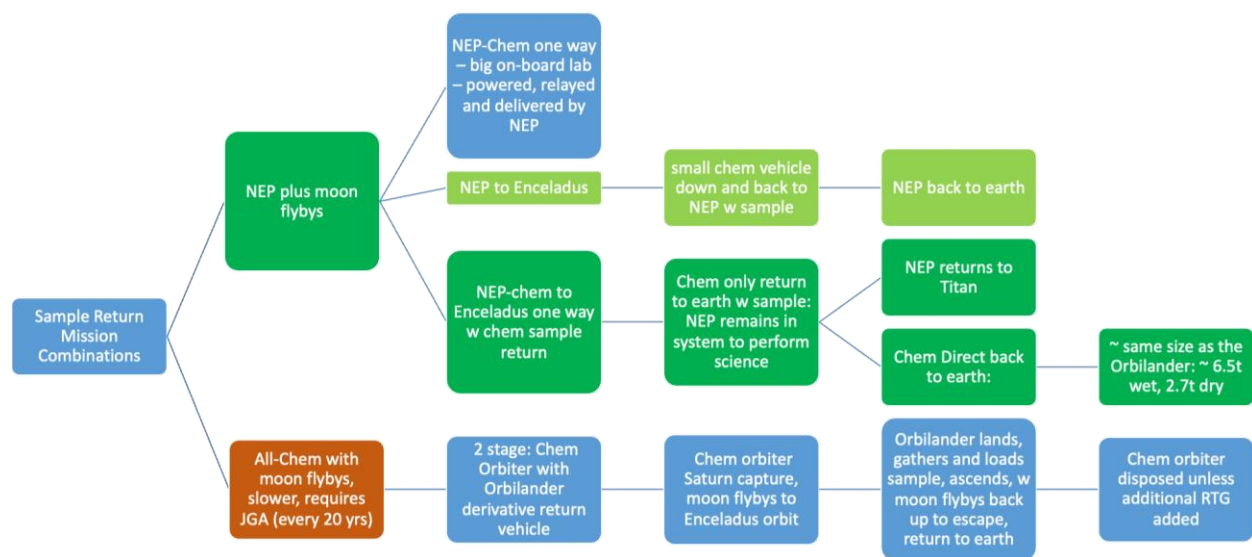


**Fig. 2 - Architecture Level Trades for Enceladus Sample Return**

*Launchers:* Based on the direction from the workshop, large commercial launchers under development were traded which allowed substantial payloads to high escape velocities, which provided at least a first leg boost to the mission. While the New Glenn launcher and SLS were traded, a Starship with multiple tanker refuelings, like those planned for Artemis, was baselined. With such large masses launched to escape, it was thought that an all-chemical sample return mission might be possible. However, the significant  $\Delta V$ s required to descend and return from Enceladus showed that, while possible with a Jupiter gravity assist, an all-chemical mission was outperformed by the NEP approach. In fact, the exclusive use of a Starship for the ESR was not possible and smaller, storable chemical stages for delivery and return would be needed.

*NEP Power and Propulsion Technologies:* While different power levels, reactor and power conversion technologies, and electric propulsion technologies were initially traded, the assumed maturity of a 40 kW FSP and the NEXT-C Ion propulsion system made them the cheapest, nearest term options.

Fig. 2 shows the top-level architecture trades of combining NEP and chemical propulsion as discussed.

### 3.0 Mission Design:

#### 3.1 Mission Design Assumptions

The mission design for roundtrip missions from Earth to Enceladus are extremely complex. To enable extensive analysis to be completed in the short time available for this study, a number of convenient simplifications were incorporated. A patched conics approach was used for the interplanetary mission design, and operations at Saturn were incorporated as fixed time and  $\Delta V$  allocations, leveraging previous Saturn moon tour analysis by Strange et. al [7]. Capture at Saturn was assumed to use a Titan Flyby to enter a 96-day orbit with periapsis at Titan radius from an arrival v-infinity of 1.5 km/s. After capture, a series of moon flybys with small low-thrust maneuvers are used to efficiently reduce the orbital energy before Enceladus orbit insertion. A similar sequence is used to raise the energy of the orbit in preparation for return to Earth. For the NEP return, a final Titan flyby is assumed to enable a departure v-infinity of 1.5 km/s. For the chemical return options, a departure v-infinity of 5.4 km/s is achieved through a series of Titan flybys and a powered Saturn flyby similar to that in Pekosh, 2022 [8]. For options where the NEP vehicle remains at Saturn, up to 9 km/s of  $\Delta V$  is reserved to enable extensive continued moon exploration.

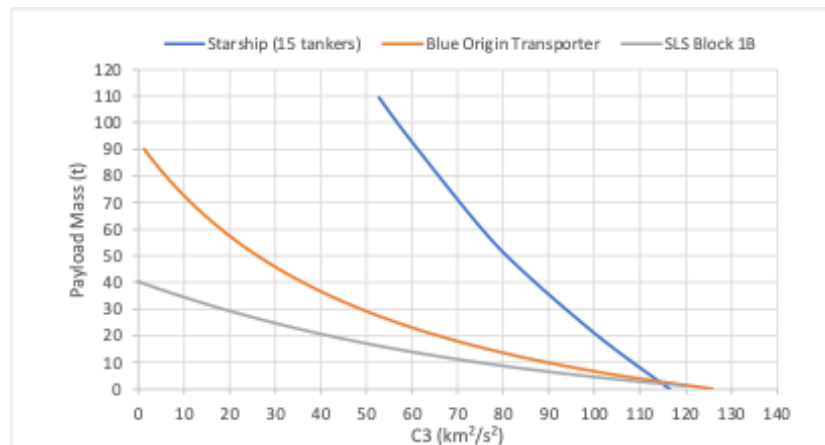


**Fig. 3 - Assumed launch vehicle performance for Starship, Blue Origin Transporter, and SLS Block 1B**

All point cases to be presented used the NEXT-C thruster performance with an Isp of 4,000 seconds. Multiple launch vehicle options were considered to enable Earth departure as shown in Fig. 3. Starship with 15-tanker refuels offers the highest level of performance, while the Blue Origin Transporter and SLS Block 1B offer lower performance alternatives. The  $\Delta V$  budgets for roundtrip NEP, and the split NEP/Chem portions of the mission are shown respectively in Table 1 and Table 2.

**Table 1 - Delta-V budget for roundtrip NEP mission option**

| Moon Tour to Enceladus Orbit via EP (Roundtrip, 44 kW) |                                |                              |              |                                               |
|--------------------------------------------------------|--------------------------------|------------------------------|--------------|-----------------------------------------------|
| Event                                                  | Deterministic $\Delta V$ (m/s) | Statistical $\Delta V$ (m/s) | Time (days)  | Note                                          |
| Earth to Saturn                                        | 8690                           | 435                          | 1826         | Power and launch dependent                    |
| Saturn Orbit Insertion                                 | 20                             | 10                           | 96           | Capture via Titan Flyby ( $V_{inf}=1.5$ km/s) |
| Moon Tour Down                                         | 492                            | 225                          | 913          | Strange Tour                                  |
| Enceladus Orbit Insertion                              | 242                            | 12                           | 49           | 5% statistical, estimated low thrust TOF      |
| Enceladus Ops                                          | 183                            | 0                            | 365          | 1 year orbit maintenance (2/3 Orbilander)     |
| Enceladus Departure                                    | 242                            | 12                           | 49           | 5% statistical, estimated low thrust TOF      |
| Moon Tour Up                                           | 492                            | 225                          | 913          | Strange Tour in Reverse                       |
| Saturn Departure                                       | 20                             | 10                           | 96           | Titan Flyby Escape to $V_{inf}=1.5$ km/s      |
| Low Thrust Penalty                                     | 254                            | 74                           | 124          | 15% $\Delta V$ and 5% TOF                     |
| Saturn to Earth                                        | 9100                           | 455                          | 1826         | Power and launch dependent                    |
| <b>Total</b>                                           | <b>19,735</b>                  | <b>1,458</b>                 | <b>6,257</b> |                                               |

**Table 2 - Delta-V budget for roundtrip split NEP/Chemical mission option**

| Moon Tour to Enceladus Orbit via EP then Spiral to Titan (44 kW) |                                |                              |              |                                                                           |
|------------------------------------------------------------------|--------------------------------|------------------------------|--------------|---------------------------------------------------------------------------|
| Event                                                            | Deterministic $\Delta V$ (m/s) | Statistical $\Delta V$ (m/s) | Time (days)  | Note                                                                      |
| Earth to Saturn                                                  | 7789                           | 389                          | 2009         | Power and launch dependent                                                |
| Saturn Orbit Insertion                                           | 20                             | 10                           | 96           | Capture via Titan Flyby ( $V_{inf}=1.5$ km/s)                             |
| Moon Tour Down                                                   | 492                            | 225                          | 913          | Strange Tour                                                              |
| Enceladus Orbit Insertion                                        | 242                            | 12                           | 49           | 5% statistical, estimated low thrust TOF                                  |
| Enceladus Ops                                                    | 183                            | 0                            | 365          | 1 year orbit maintenance (2/3 Orbilander)                                 |
| Enceladus Departure                                              | 242                            | 12                           | 49           | 5% statistical, estimated low thrust TOF                                  |
| Low Thrust Penalty                                               | 177                            | 39                           | 74           | 15% $\Delta V$ and 5% TOF                                                 |
| Post Enceladus Mission                                           | 9000                           | 0                            | 1096         | Spiral from Enceladus to Titan (1,000 km Circular)                        |
| <b>Total</b>                                                     | <b>18,145</b>                  | <b>688</b>                   | <b>4,650</b> |                                                                           |
| Split Chemical Return from Enceladus (One Way)                   |                                |                              |              |                                                                           |
| Event                                                            | Deterministic $\Delta V$ (m/s) | Statistical $\Delta V$ (m/s) | Time (days)  | Note                                                                      |
| Landing                                                          | 260                            | 13                           | 1            | Down to Surface                                                           |
| Enceladus Surface Ops                                            | 0                              | 0                            | 180          | 6 months on surface                                                       |
| Launch                                                           | 260                            | 13                           | 1            | Up to Enceladus Orbit                                                     |
| Enceladus Departure                                              | 100                            | 5                            | 0            | Orbilander departure from halo orbit                                      |
| Moon Tour Up                                                     | 492                            | 225                          | 913          | Strange Tour Up                                                           |
| Periapsis Lower                                                  | 640                            | 32                           | 180          |                                                                           |
| Saturn Departure                                                 | 506                            | 25                           | 0            | Burn at Saturn Periapsis (Titan Sample Return Study)                      |
| Additional Titan Pump Up                                         | -200                           | 0                            | 240          | Extend the departure sequence based on Titan Sample Return Study Analysis |
| Saturn to Earth                                                  | 0                              | 35                           | 1972         | Ballistic Return to Earth                                                 |
| <b>Total</b>                                                     | <b>2,058</b>                   | <b>348</b>                   | <b>3,487</b> |                                                                           |

### 3.2 NEP One-Way Point Designs:

Four different point cases were designed for the one-way NEP mission options. In these cases, 38 kW is used for the NEP system to deliver the chemical return vehicle and stays in the Saturnian system for continued exploration. Case 1, shown in Fig. 4, uses a 15-tanker Starship launch vehicle to arrive at Saturn 5.51 years after departure. The sample is returned to Earth by the chemical return vehicle 18.25 years after departure. Case 2, shown in Fig. 5, is the same as Case 1, except the NEP vehicle inert mass is reduced by 1,400 kg to show the potential impact of advanced technology to reduce the reactor mass. The time to Saturn is reduced to 5.24 years, but the sample return time remains unchanged. Case 3, shown in Fig. 6, uses the Blue Origin Transporter as the launch vehicle to arrive at Saturn 7.29 years after departure. An outbound Earth Gravity Assist (EGA) is necessary as this launch vehicle performance is less than Case 1. The sample is returned to Earth by the chemical return vehicle 20.0 years after departure. Finally, Case 4 shown in Fig. 7, uses SLS Block 1 B with an outbound EGA to arrive at Saturn 7.5 years, and returns the sample to Earth in 20.12 years. Each figure provides additional data for each mission.

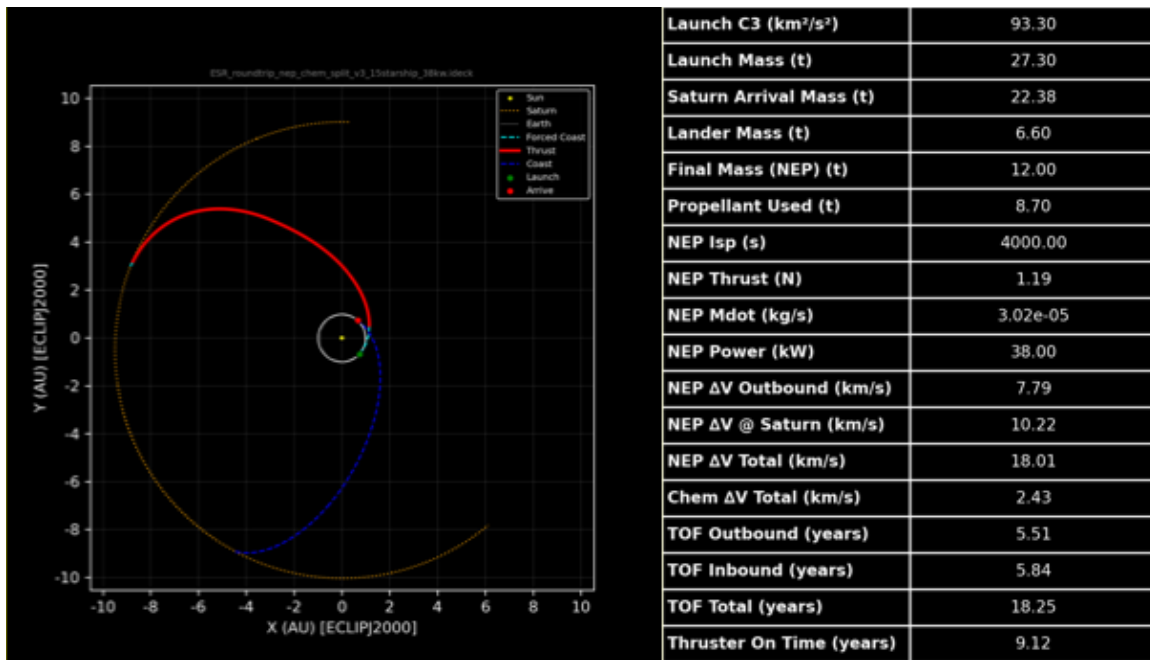


Fig. 4 - Split NEP/Chem mission option with 38 kW to NEP and 15 tanker Starship

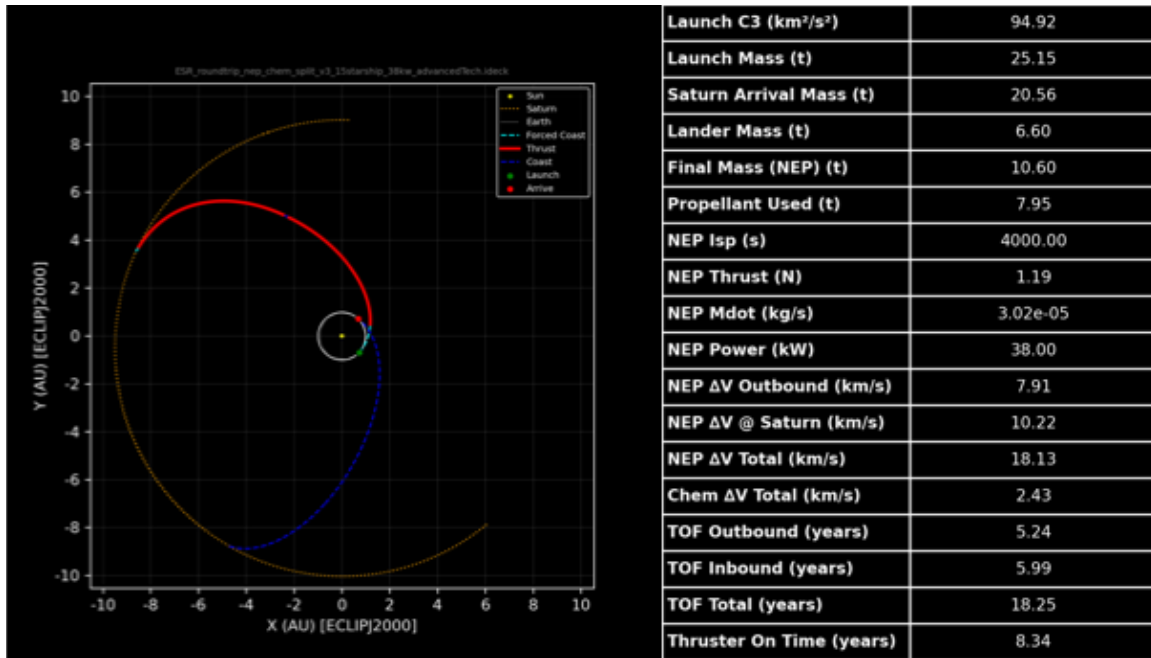


Fig. 5 - Split NEP/Chem mission option with 38 kW to NEP, 15 tanker Starship, and 1400 kg inert mass reduction

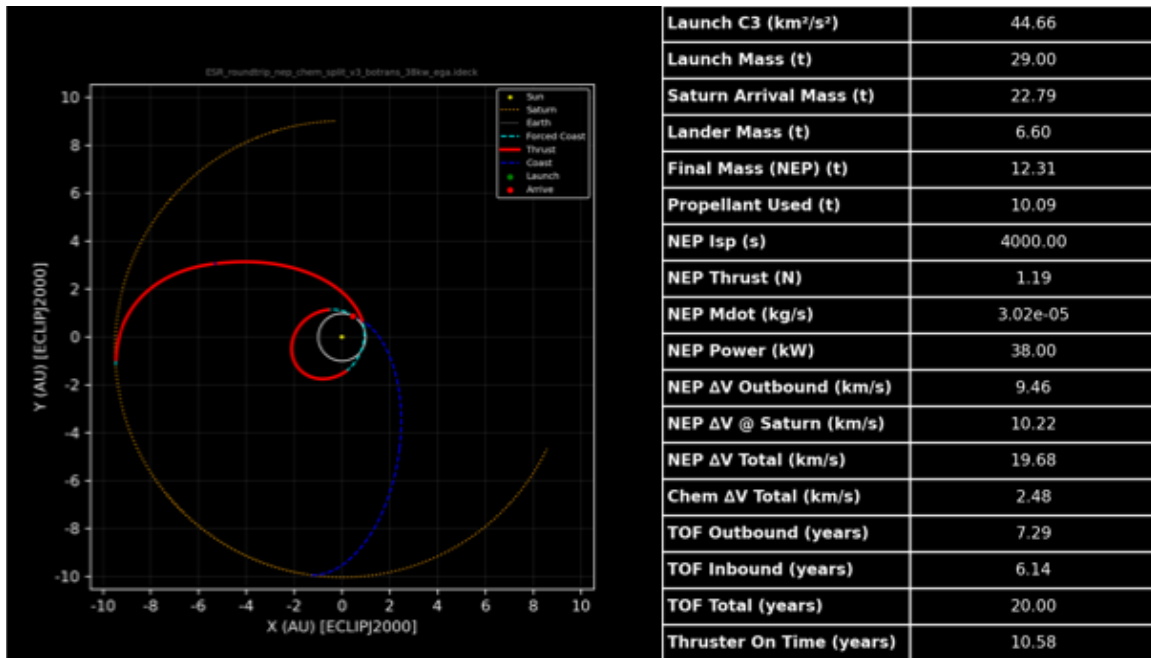


Fig. 6 - Split NEP/Chem mission option with 38 kW to NEP and Blue Origin Transporter with outbound EGA

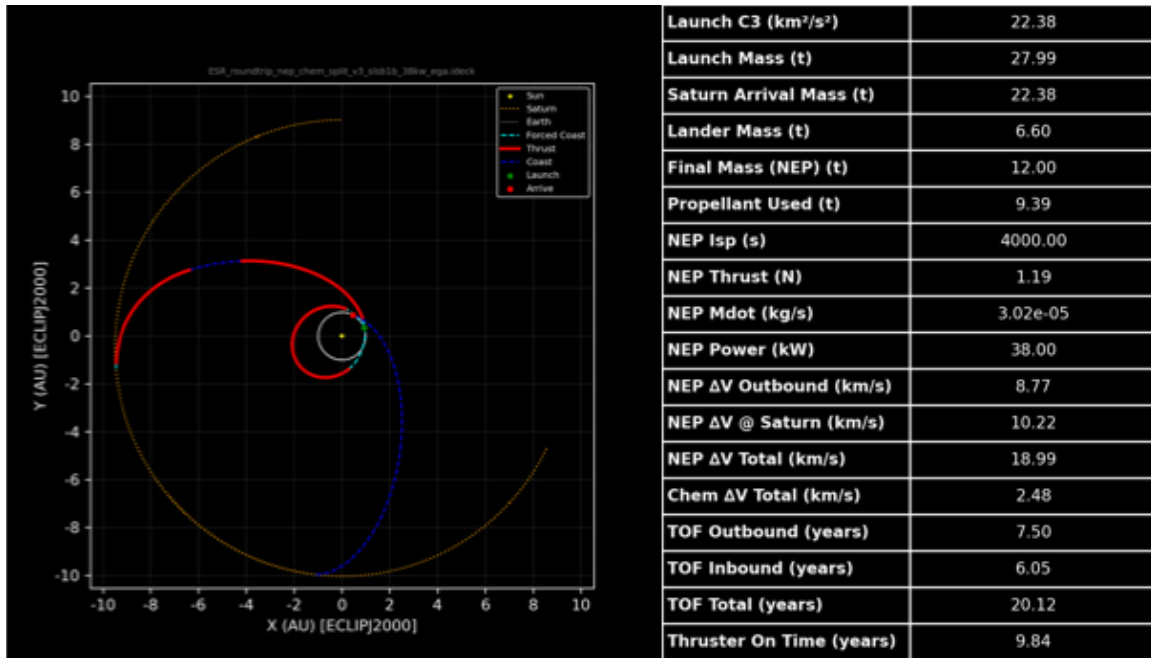


Fig. 7 - Split NEP/Chem mission option with 38 kW to NEP and SLS Block 1B with outbound EGA

### 3.3 NEP Round-Trip Point Designs:

Three different point cases were designed for the roundtrip NEP mission options. In these cases, NEP and a 15-tanker Starship are used, and the NEP vehicle also returns the sample to Earth. Case 1, shown in Fig. 8, uses 38 kW to the EP system to arrive at Saturn 5.03 years after departure. After 6.15 years at Saturn, the NEP system is used to return the entire vehicle and sample to Earth in 4.97 years, for a total roundtrip time of 16.15 years from Earth departure to sample return. Case 2, shown in Fig. 9, uses 744 kW to the EP system (along with the appropriate increase in inert mass) to reduce the total roundtrip time to 13.1 years. Case 3, shown in Fig. 10, uses the same 744 kW to EP, but forgoes the fuel-efficient moon tour to get in and out of the Saturn gravity well in favor of a faster all low-thrust propulsive option. Using EP to travel down to Enceladus and back up reduces the total roundtrip time of flight to 11.05 years at the cost of additional propellant used.

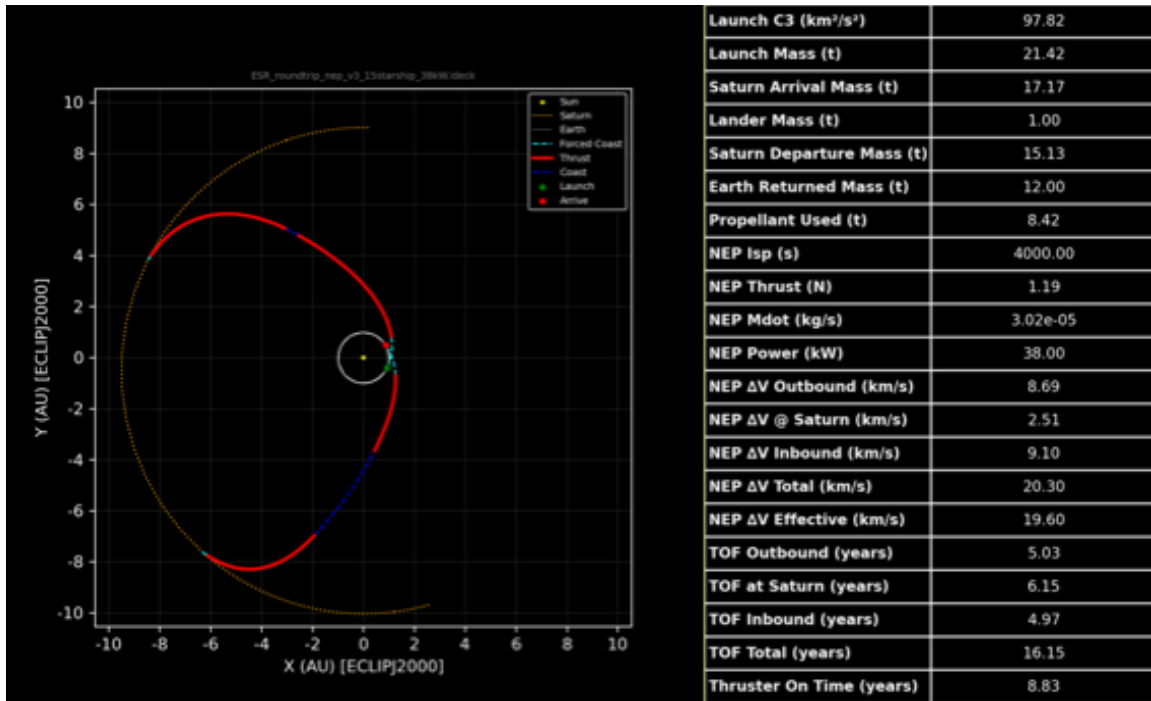


Fig. 8 - Roundtrip NEP mission option with 38 kW to NEP and 15 tanker Starship

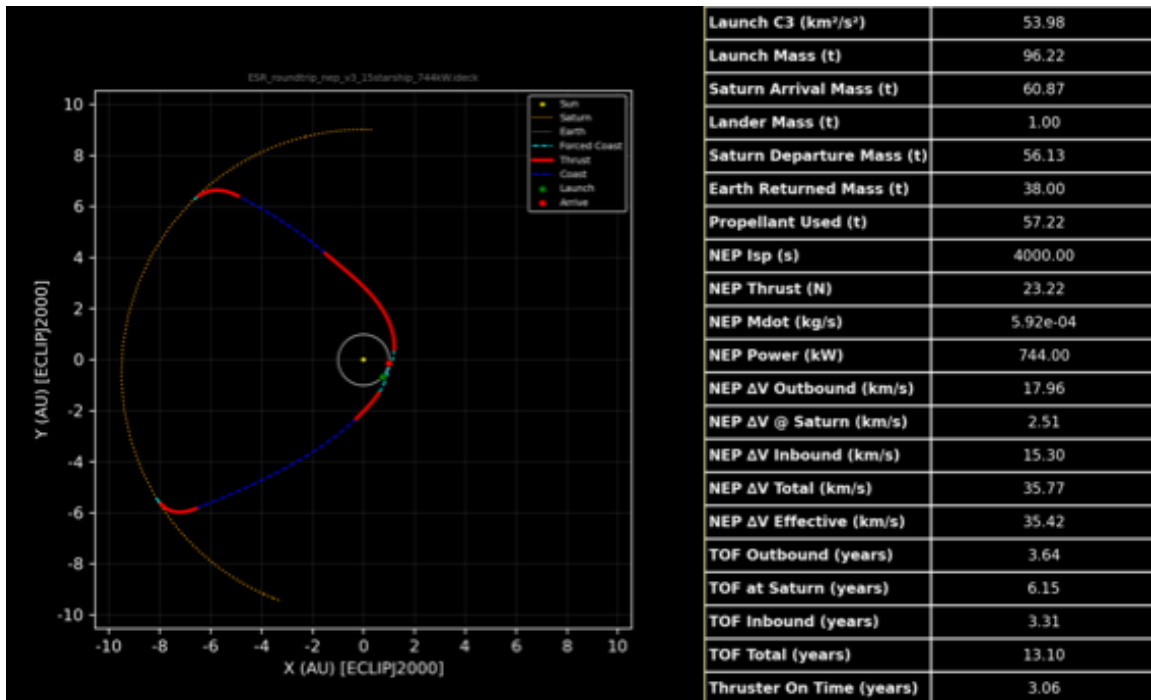


Fig. 9 - Roundtrip NEP mission option with 744 kW to NEP and 15 tanker Starship

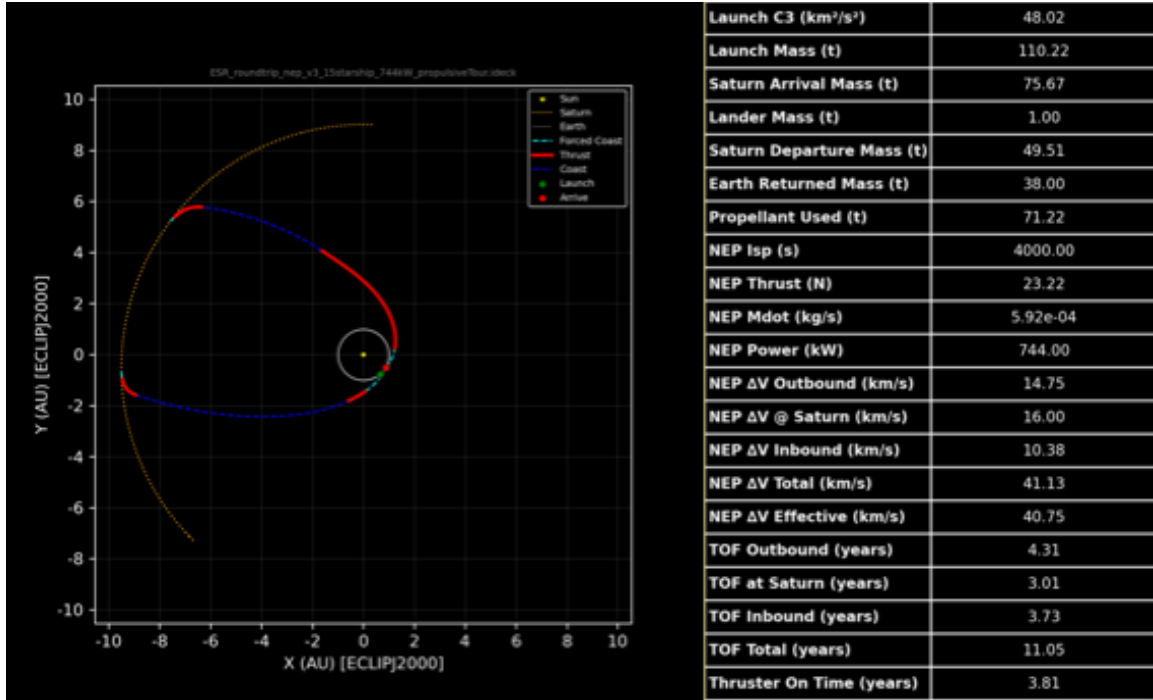


**Fig. 10 - Roundtrip NEP mission option with 744 kW to NEP, 15 tanker Starship, and all propulsive Saturn operations**

### 3.6 All-Chemical Mission Options:

The all-chemical mission architecture was developed to determine whether an Orbilander-type spacecraft could be delivered to Enceladus using only chemical propulsion stages in combination with the SpaceX Starship. Two variations of the all-chemical mission were analyzed: a direct Earth-to-Saturn vicinity trajectory, and a Jupiter gravity-assist (JGA) trajectory. The All-Chemical Mission Profile Delta-V Budget can be seen in Table 3, which includes both a one-way trip budget and a round-trip budget, both using a Jupiter gravity assist.

Both the JGA and Direct trajectories were modeled in Copernicus. The JGA mission was set up to launch on October 20<sup>th</sup>, 2038, with an image of the trajectory in Copernicus shown in Figure 3. The trajectory was found to have the JGA occur on October 11<sup>th</sup>, 2040, with capture at Saturn on October 19<sup>th</sup>, 2045. The model terminated after initiating its first Titan flyby on March 13<sup>th</sup>, 2046, which begins the moon tour down to Enceladus. This came out to have a total time of flight of 7.4 years, with Starship providing the initial trans-Saturn injection burn of 6.98 km/s to achieve a required C<sub>3</sub> of approximately 96 km<sup>2</sup>/s<sup>2</sup>. For this mission date, the model converged with a Saturn orbit injection of 230 m/s and a periaapsis raise burn of 950 m/s. It should be noted that this date was chosen in order to provide proper alignment for a JGA opportunity, which is constrained by the approximately 19.86-year synodic period between Jupiter and Saturn. The total ΔV and TOF for the JGA mission can be seen in Table 4, which is based on the aforementioned roundtrip budget and split between the delivery to Enceladus and the return trip home.

**Table 3 - All-Chemical Mission Profile Delta-V Budget**

| Split Chemical Return from Enceladus (One Way) |                                   |                                 |                |                                                                              |
|------------------------------------------------|-----------------------------------|---------------------------------|----------------|------------------------------------------------------------------------------|
| Event                                          | Deterministic $\Delta V$<br>(m/s) | Statistical $\Delta V$<br>(m/s) | Time<br>(days) | Note                                                                         |
| Landing                                        | 260                               | 13                              | 1              | Down to Surface                                                              |
| Enceladus Surface<br>Ops                       | 0                                 | 0                               | 180            | 6 months on surface                                                          |
| Launch                                         | 260                               | 13                              | 1              | Up to Enceladus Orbit                                                        |
| Enceladus<br>Departure                         | 100                               | 5                               | 0              | Orbilander departure from halo orbit                                         |
| Moon Tour Up                                   | 492                               | 225                             | 913            | Strange Tour Up                                                              |
| Periapsis Lower                                | 640                               | 32                              | 180            |                                                                              |
| Saturn Departure                               | 506                               | 25                              | 0              | Burn at Saturn Periapsis (Titan Sample Return Study)                         |
| Additional Titan<br>Pump Up                    | -200                              | 0                               | 240            | Extend the departure sequence based on Titan Sample<br>Return Study Analysis |
| Saturn to Earth                                | 0                                 | 35                              | 1972           | Ballistic Return to Earth                                                    |
| <b>Total</b>                                   | <b>2,058</b>                      | <b>348</b>                      | <b>3,487</b>   |                                                                              |

| Moon Tour to Enceladus Orbit via Chem (Roundtrip) |                                   |                                 |                |                                                                       |
|---------------------------------------------------|-----------------------------------|---------------------------------|----------------|-----------------------------------------------------------------------|
| Event                                             | Deterministic $\Delta V$<br>(m/s) | Statistical $\Delta V$<br>(m/s) | Time<br>(days) | Note                                                                  |
| Earth to Saturn                                   | 0                                 | 35                              | 2557           | Earth to Saturn (JGA) from Orbilander                                 |
| Saturn Orbit<br>Insertion                         | 240                               | 12                              | 0              | $V_{\infty} = 3.3$ km/s w/ JGA (Orbilander)                           |
| Periapsis Raise                                   | 640                               | 32                              | 180            | Raise to Titan Radius (Orbilander)                                    |
| Moon Tour Down                                    | 560                               | 176                             | 1461           | Orbilander (4 years)                                                  |
| Enceladus Orbit<br>Insertion                      | 100                               | 5                               | 0              | 5% statistical                                                        |
| Enceladus Mapping                                 | 92                                | 0                               | 180            | 1/3rd of Orbilander OM budget                                         |
| Landing                                           | 260                               | 13                              | 1              | Down to Surface                                                       |
| Enceladus Surface<br>Ops                          | 0                                 | 0                               | 180            | 6 months on surface                                                   |
| Launch                                            | 260                               | 13                              | 1              | Up to Enceladus Orbit                                                 |
| Enceladus<br>Departure                            | 100                               | 5                               | 0              | Orbilander departure from halo orbit                                  |
| Moon Tour Up                                      | 492                               | 225                             | 913            | Strange Tour Up (Faster)                                              |
| Periapsis Lower                                   | 640                               | 32                              | 180            |                                                                       |
| Saturn Departure                                  | 506                               | 25                              | 0              | Burn at Saturn Periapsis (Titan Sample Return Study)                  |
| Additional Titan<br>Pump Up                       | -200                              | 0                               | 240            | Extend the departure sequence (Titan Sample Return Study<br>Analysis) |
| Saturn to Earth                                   | 0                                 | 35                              | 1972           | Ballistic Return to Earth (From Titan Sample Return Study)            |
| <b>Total</b>                                      | <b>3,690</b>                      | <b>608</b>                      | <b>7,865</b>   |                                                                       |

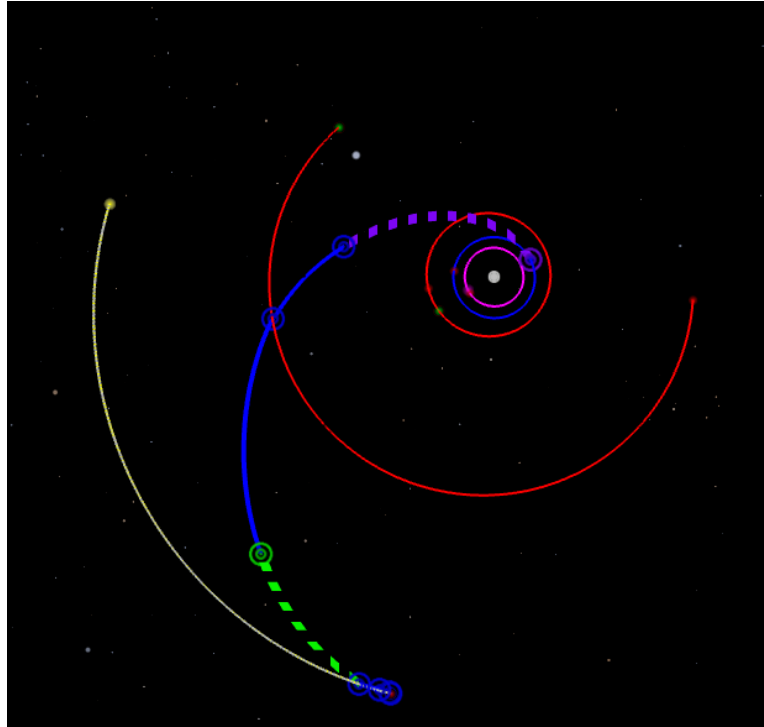


**Fig. 11 - Earth-to-Saturn vicinity with JGA Trajectory Model**

**Table 4 - Delta-V Allocation and TOF for the Delivery and Return Legs of the JGA All-Chemical Roundtrip mission**

|                   | $\Delta V$      | Time of Flight    |
|-------------------|-----------------|-------------------|
| Delivery          | <b>1892 m/s</b> | <b>12.0 years</b> |
| Orbilander Return | <b>2406 m/s</b> | <b>9.6 years</b>  |
| Total Mission     | <b>4298 m/s</b> | <b>21.5 years</b> |

Similarly, the Direct mission was set up to launch on March 9<sup>th</sup>, 2035. Unlike the JGA mission architecture, there is more flexibility here with when the mission can launch. In this case, the date was chosen to provide better lighting conditions upon arrival at Enceladus. Starship provides an 8.41 km/s trans-Saturn injection burn to achieve the required  $C_3$  of 135.45 km<sup>2</sup>/s<sup>2</sup>. On January 17<sup>th</sup>, 2041, the spacecraft performs the Saturn insertion burn, which came out to a  $\Delta V$  of 720 m/s. After SOI, the spacecraft begins its moon tour down to Enceladus following a periapsis raise  $\Delta V$  of 615 m/s. Fig. 12 shows the direct trajectory modeled in Copernicus. Table 5 provides the total allocated  $\Delta V$  and TOF for the delivery and return legs based on the Roundtrip budget. Due to the lack of a JGA, the total  $\Delta V$  for the mission is increased, although the increase is entirely allocated to the delivery leg of the mission. This allows the direct mission to be almost 3 years faster than the JGA mission, at the cost of  $\Delta V$ . Unfortunately, as is shown in the following section, this direct mission proved to be infeasible given the mission architecture constraints.

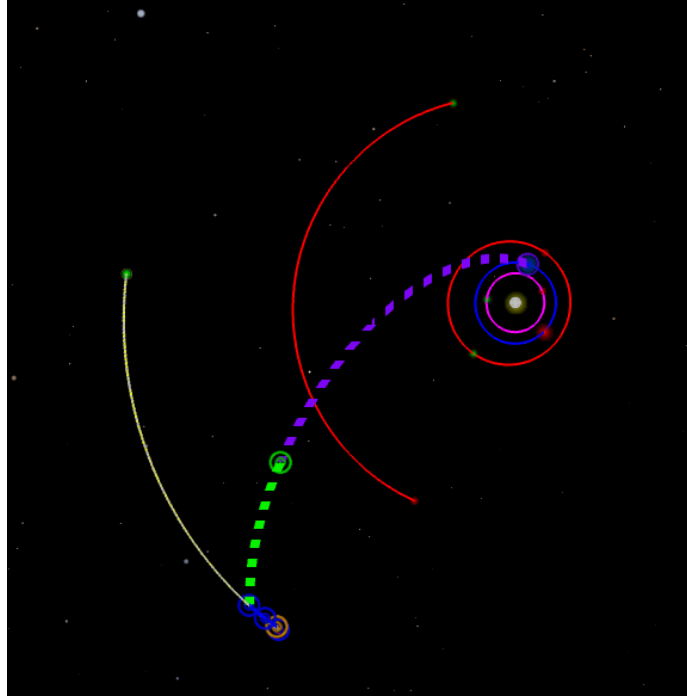


**Fig. 12 - Direct Earth-to-Saturn vicinity Trajectory Model**

**Table 5 - Delta-V Allocation and TOF for the Delivery and Return legs of the Direct All-Chemical Roundtrip mission**

|                 | <b>Deterministic <math>\Delta V</math></b> | <b>Statistical <math>\Delta V</math></b> | <b>Total <math>\Delta V</math></b> | <b>Time of Flight</b> |
|-----------------|--------------------------------------------|------------------------------------------|------------------------------------|-----------------------|
| <b>Delivery</b> | 2080 m/s                                   | 282 m/s                                  | 2362 m/s                           | 9.2 years             |
| <b>Return</b>   | 2058 m/s                                   | 348 m/s                                  | 2406 m/s                           | 9.6 years             |
| <b>Total</b>    | <b>4138 m/s</b>                            | <b>630 m/s</b>                           | <b>4768 m/s</b>                    | <b>18.8 years</b>     |

### 3.7 All-Chemical Stage Sizing

Based on the required  $\Delta V$ s,  $C_3$ s, and payload mass, a Jupiter Gravity Assist trajectory was identified as the only viable all-chemical option. Fig. 3 shows the deliverable payload masses for Starship as a function of  $C_3$  and number of fueling tankers. With the required  $C_3$  of the JGA case at  $98.2 \text{ km}^2/\text{s}^2$ , Starship can deliver up to 90 tons with 23 tanker flights or 30 tons with 15 tanker flights. This option leverages Jupiter's gravity to reduce the energy required for Saturn capture and the subsequent moon targeting and does not require complex staging configurations. Table 6 shows the properly sized delivery stage as Stage 1, with Stage 2 (Payload) being the 6610 kg Orbilander vehicle. Specific impulse for the delivery stage was set to be 305 seconds with a 0.6 maximum propellant fraction for all missions. Based on this table, Starship is capable of launching this vehicle with a  $C_3$  of  $98.2 \text{ km}^2/\text{s}^2$  and 15+ tankers (a fully fueled Starship would be capable of throwing another 60 tons of spacecraft/payload in this scenario).

**Table 6 - All-Chemical Staging for JGA Mission**

|               |                               | Stage Number |                   | Overall   |
|---------------|-------------------------------|--------------|-------------------|-----------|
|               |                               | Stage 1      | Stage 2 (Payload) |           |
| <b>Mass</b>   | <b>Dry</b>                    | 9,445 kg     | 6,610 kg          | 16,055 kg |
|               | <b>Propellant</b>             | 14,167 kg    | 0 kg              | 14,167 kg |
|               | <b>Wet</b>                    | 23,612 kg    | 6,610 kg          | 30,222 kg |
|               | <b>Ratio</b>                  | 2.5          | 1                 | 1.88      |
|               | <b>Inert Fraction (IMF)</b>   | 0.4          | 1                 | 0.53      |
|               | <b>Propellant Fraction</b>    | 0.6          | 0                 | 0.47      |
| <b>Engine</b> | <b>Specific Impulse (ISP)</b> | 305 s        | N/A               | N/A       |
|               | <b>Exhaust Velocity</b>       | 2,991 m/s    | N/A               | N/A       |
| <b>ΔV</b>     | <b>Value</b>                  | 1,892 m/s    | N/A               | 1,892 m/s |
|               | <b>Fraction</b>               | 1            | N/A               | 1         |

By contrast, the direct Earth-to-Saturn transfer required a  $C_3$  of 135.45 km<sup>2</sup>/s<sup>2</sup>. This limits Starship's payload capacity to approximately 30 tons even when fully fueled with 23 tankers. Multiple staging approaches were investigated in an attempt to make this a feasible mission option. Adding single, dual, and even triple delivery stages could not get the direct trajectory to satisfy mission constraints without major architectural modifications. The single delivery stage configuration was mass-limited by Starship's maximum payload capacity, with an overall wet mass of over 73 tons. This single delivery stage configuration can be seen in Table 7.

**Table 7 - All-Chemical Staging for Single Delivery Stage Direct Mission**

|               |                               | Stage Number |                   | Overall   |
|---------------|-------------------------------|--------------|-------------------|-----------|
|               |                               | Stage 1      | Stage 2 (Payload) |           |
| <b>Mass</b>   | <b>Dry</b>                    | 26,776 kg    | 6,610 kg          | 33,386 kg |
|               | <b>Propellant</b>             | 40,164 kg    | 0 kg              | 40,164 kg |
|               | <b>Wet</b>                    | 66,940 kg    | 6,610 kg          | 73,550 kg |
|               | <b>Ratio</b>                  | 2.5          | 1                 | 2.20      |
|               | <b>Inert Fraction (IMF)</b>   | 0.4          | 1                 | 0.45      |
|               | <b>Propellant Fraction</b>    | 0.6          | 0                 | 0.54      |
| <b>Engine</b> | <b>Specific Impulse (ISP)</b> | 305 s        | N/A               | N/A       |
|               | <b>Exhaust Velocity</b>       | 2,991 m/s    | N/A               | N/A       |
| <b>ΔV</b>     | <b>Value</b>                  | 2,362 m/s    | N/A               | 2,362 m/s |
|               | <b>Fraction</b>               | 1            | N/A               | 1         |

The dual (Table 8) and triple (Table 9) delivery stage configurations provide more ΔV per total mass but are slightly outside of the range of Starship payload capacity. Payload delivery efficiency to Saturn via Starship decreases rapidly when shifting away from a JGA, with the added tankage and structural mass of the additional stages reducing the usable Starship payload. As such, the direct case was eliminated from further consideration.

**Table 8 - All-Chemical Staging for Dual Delivery Stage Direct Mission**

|            |                        | Stage Number |           |                   | Overall   |
|------------|------------------------|--------------|-----------|-------------------|-----------|
|            |                        | Stage 1      | Stage 2   | Stage 3 (Payload) |           |
| Mass       | Dry                    | 9,185 kg     | 3,198 kg  | 6,610 kg          | 18,993 kg |
|            | Propellant             | 13,777 kg    | 3,578 kg  | 0 kg              | 17,355 kg |
|            | Wet                    | 22,962 kg    | 6,776 kg  | 6,610 kg          | 36,348 kg |
|            | Ratio                  | 2.5          | 2.1       | 1                 | 1.91      |
|            | Inert Fraction (IMF)   | 0.4          | 0.5       | 1                 | 0.52      |
|            | Propellant Fraction    | 0.6          | 0.5       | 0                 | 0.48      |
| Engine     | Specific Impulse (ISP) | 306 s        | 306 s     | N/A               | N/A       |
|            | Exhaust Velocity       | 3,000 m/s    | 3,000 m/s | N/A               | N/A       |
| $\Delta V$ | Value                  | 1,429 m/s    | 933 m/s   | N/A               | 2,362 m/s |
|            | Fraction               | 0.61         | 0.39      | N/A               | 1         |

**Table 9 - All-Chemical Staging for Triple Delivery Stage Direct Mission**

|            |                        | Stage Number |           |           |                   | Overall   |
|------------|------------------------|--------------|-----------|-----------|-------------------|-----------|
|            |                        | Stage 1      | Stage 2   | Stage 3   | Stage 4 (Payload) |           |
| Mass       | Dry                    | 4,964 kg     | 2,874 kg  | 2,123 kg  | 6,610 kg          | 16,571 kg |
|            | Propellant             | 6,744 kg     | 3,714 kg  | 3,184 kg  | 0 kg              | 13,642 kg |
|            | Wet                    | 11,709 kg    | 6,588 kg  | 5,307 kg  | 6,610 kg          | 30,213 kg |
|            | Ratio                  | 2.4          | 2.3       | 2.5       | 1                 | 1.82      |
|            | Inert Fraction (IMF)   | 0.4          | 0.4       | 0.4       | 1                 | 0.55      |
|            | Propellant Fraction    | 0.6          | 0.6       | 0.6       | 0                 | 0.45      |
| Engine     | Specific Impulse (ISP) | 306 s        | 306 s     | 306 s     | N/A               | N/A       |
|            | Exhaust Velocity       | 3,000 m/s    | 3,000 m/s | 3,000 m/s | N/A               | N/A       |
| $\Delta V$ | Value                  | 758 m/s      | 672 m/s   | 933 m/s   | N/A               | 2,362 m/s |
|            | Fraction               | 0.32         | 0.28      | 0.39      | N/A               | 1         |

### 3.8 Low-Thrust Spiral Transfer Comparison

To determine how much time the moon tours save, a low-thrust spiral transfer from Titan directly to Enceladus was analyzed. Ranges of initial mass (6000 kg to 20000 kg), specific impulse (3000 s to 5000 s), thrust magnitude (0.5 N to 3 N), and power (20 kW to 100 kW) were examined, with a focus on the case of 40 kW at 4000 Isp. An example trajectory modeled in Copernicus can be seen in Fig. 13.

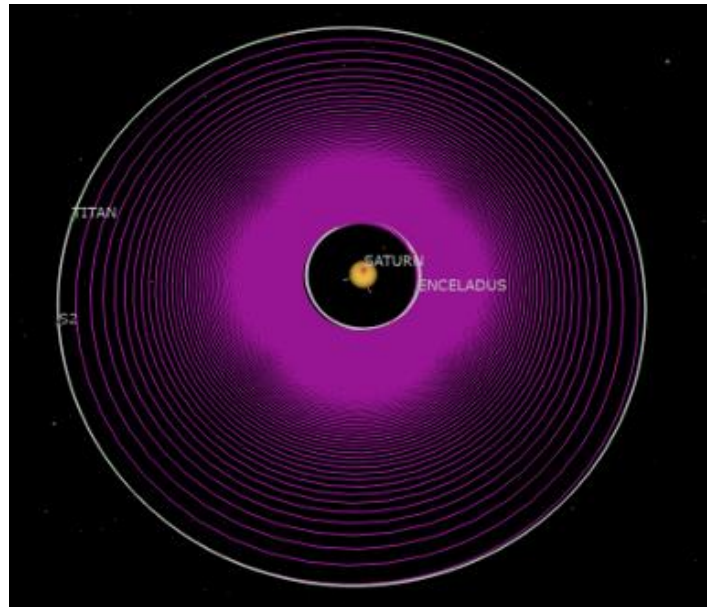


**Fig. 13 - A Low-Thrust Spiral Transfer from Titan to Enceladus**

Using Edelbaum's method for estimating  $\Delta V$  for low-thrust transfers between circular orbits, the one-way and roundtrip  $\Delta V$ s were calculated to be 7.06 km/s and 14.12 km/s respectively [9]. Additionally, by using the equation shown in Fig. 14 to calculate the dry mass for a given system power level, the initial mass of the spiral can be found by working backwards from dry mass (final mass at return to Earth) using the rocket equation. The mass of the spacecraft was based on previous work of a similar power-level NEP vehicle [10].

Combined System Dry Mass with 50% Growth:

$$\text{Mass} = (1902.1 + 81.218 \cdot P - 0.1424 \cdot P^2 + 2209 + 4.75 \cdot P) \cdot 1.5$$

P = power in kW, valid from 20 - 100 kW

Note: this does not include xenon tank mass (~10% of Xenon mass)

**Fig. 14 - Calculating Dry Mass for a given System Power Level**

The results for this analysis can be seen in Fig. 15. For the highlighted case of 4000 Isp at 40 kW, the spiral transfer had a thrust of 1.4 N, an initial mass of 19 tons, a transfer time of 5.1 years, and a 5.7 t propellant requirement. Due to the increased transfer time, high  $\Delta V$ , and existence of previously analyzed Saturnian moon tours, the low-thrust direct spiral option was eliminated from further consideration.

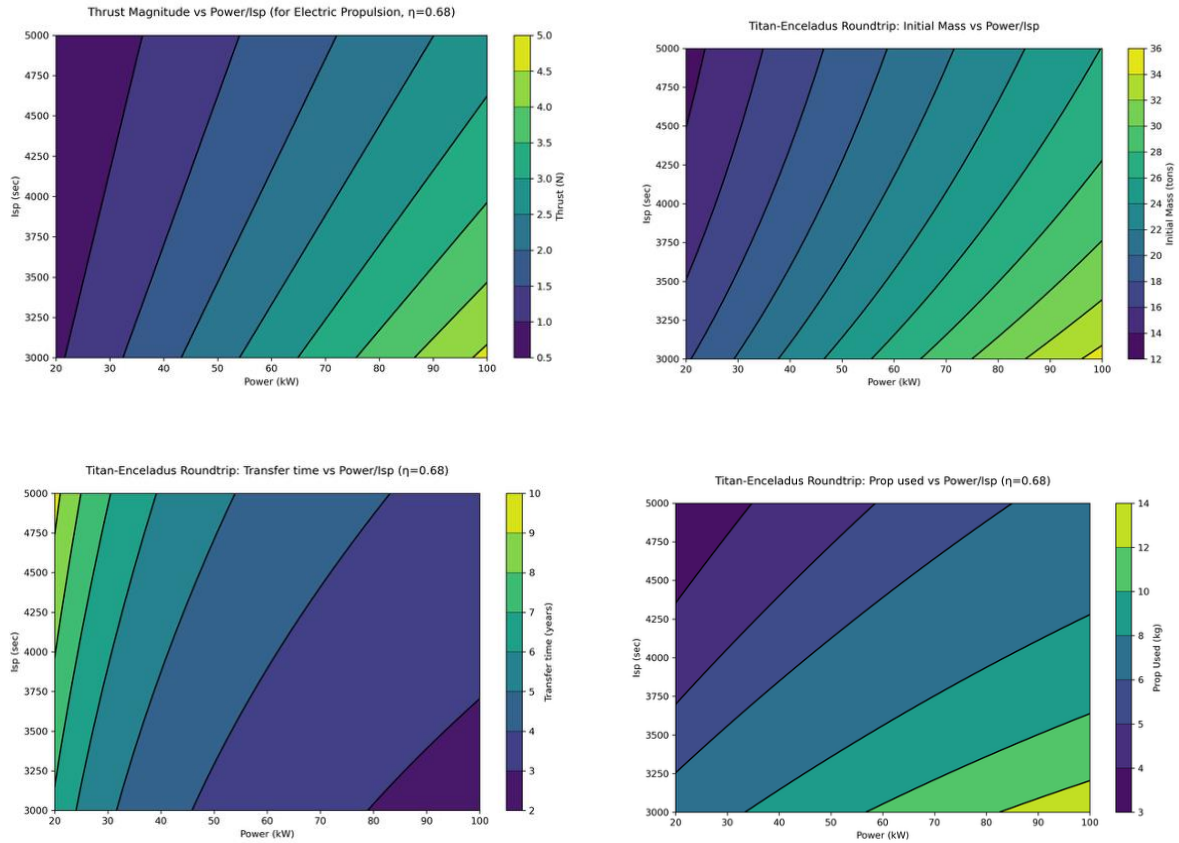


**Fig. 15 - Enceladus-Titan Low-Thrust Spiral Transfer Analysis Results**

#### 4.0 Concept of Operations (ConOps):

The baseline concept of operations is shown pictorially in Fig. 16. As described in the trades section above, this version of the 40 kW NEP vehicle carries a 6.6t Orbilander-representative sample return vehicle to Enceladus, supports the lander with mapping and relay, and then moves on to explore the rest of Saturn while the lander returns the sample to Earth. Earliest launch was chosen as 2033, but the use of NEP should allow for yearly launch opportunities, with large windows every year, since it is a direct trajectory and does not need gravity assists (especially Jupiter's which is only available roughly every 20 yrs). The mission assumes a Starship launcher containing the NEP vehicle is refueled with 15 tankers (over the course of at least three months) in 2000 km orbit. Launching nuclear materials on a Starship will need to be approved. During this time the NEP vehicle is assumed to be supplied with survival power by the Starship vehicle and the reactor is kept unpowered. The Starship next fires its engines to deliver a  $C_3$  of  $93 \text{ km}^2/\text{s}^2$  to begin its direct journey to Saturn.

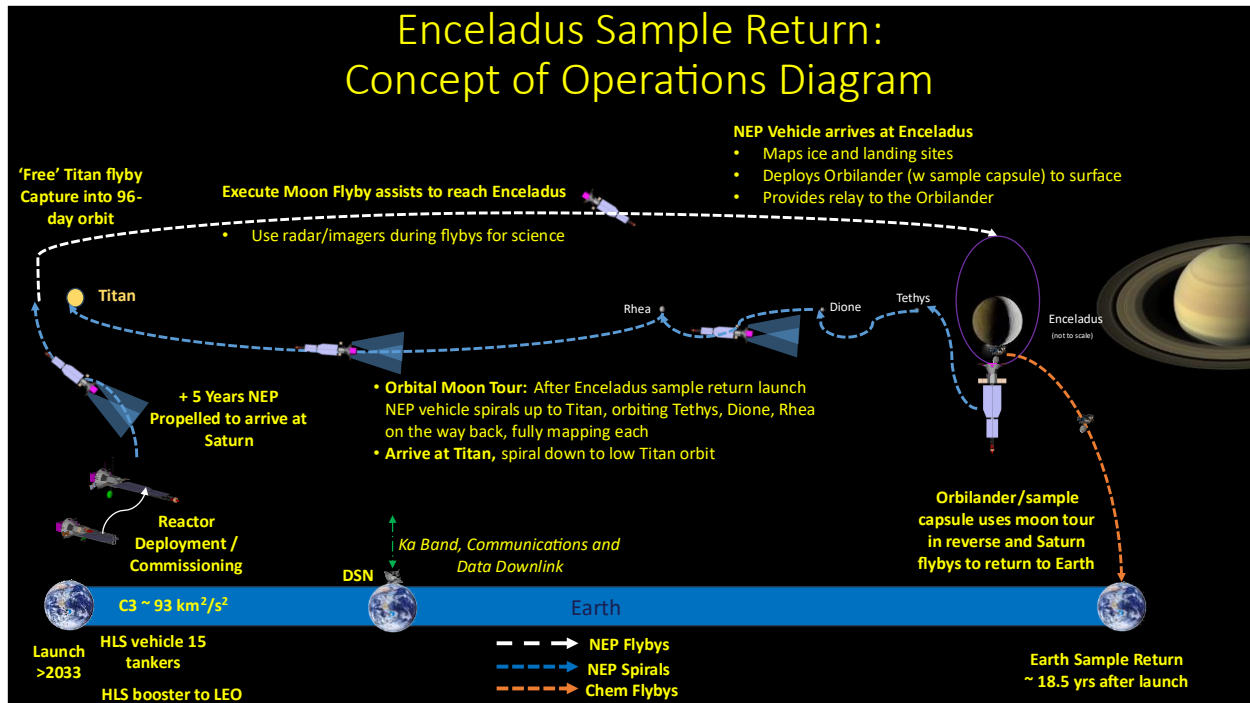


**Fig. 16 - SERENE Sample Return ConOps**

After separation from the Starship, the reactor boom is unfolded, and the Starship is discarded. An approximately 24-hour startup for the reactor is begun, supported by a commissioning body mounted solar array and batteries. The NEP vehicle operates for five years thrusting and coasting to reach a high elliptic Saturn orbit using a Titan flyby. It next lowers its orbit to Enceladus via an orbital moon tour, leveraging flybys along the way as discussed. The moon tour conserves Xenon propellant for the post sampling mission. Once it reaches Enceladus, the vehicle inserts into an elliptic Lagrange orbit around Enceladus, maps the moon, and finds potential landing sites.

After a site is chosen, the Orbilander will land, perform surface science, and gather samples which are loaded using its sampling arm into an Earth entry capsule. The Orbilander then launches and begins the moon flyby tour to depart Saturn, followed by a burn to return to an Earth flyby where it deploys the Enceladus samples. While the Orbilander is returning to Earth, the NEP vehicle spirals up to orbit and map four additional moons, the final one being Titan. Data are returned at up to 500 kbps using a Ka-band antenna to the Deep Space Network (DSN). To minimize propellant for disposal, SERENE will be placed in a parking orbit whose ground track avoids Titan's lakes. While not stable long term, the reactor will cool sufficiently once shut down that no high temperature materials will hit the surface. The rest of the spacecraft is assumed to breakup upon entry, similar to the Titan Orbiter study [11].

## 5.0 Design:

### 5.1 Configuration:

The Saturn Enceladus Return Explorer with Nuclear Electric propulsion (SERENE) vehicle is a 40 kWe-class NEP vehicle that delivers a ~6.5 t lander/sample return vehicle to Enceladus to conduct a science/sample return mission. Launch would be performed by a Starship utilizing the extended fairing that allows for payloads up to a length of 22 meters. Fig. 17 and Fig. 18 below show the stowed SERENE vehicle inside the Starship extended Payload Fairing (PLF) mounted to a Payload Adaptor (PLA) that is designed specifically for the launch of the SERENE vehicle.

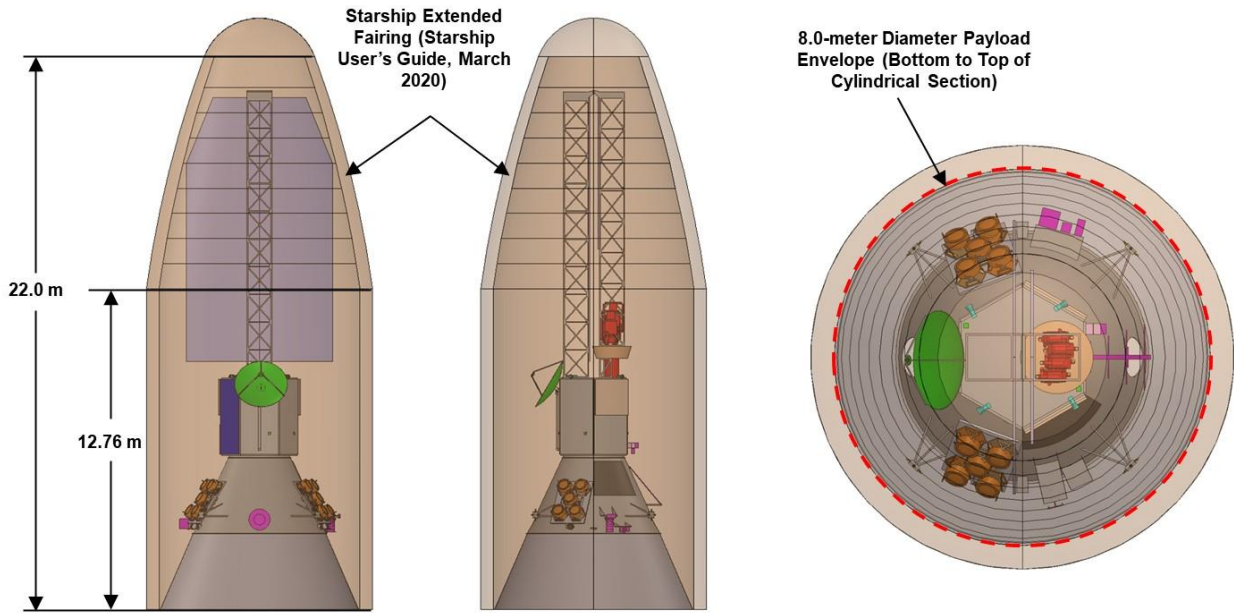


**Fig. 17 - SERENE vehicle inside the Starship extended PLF**

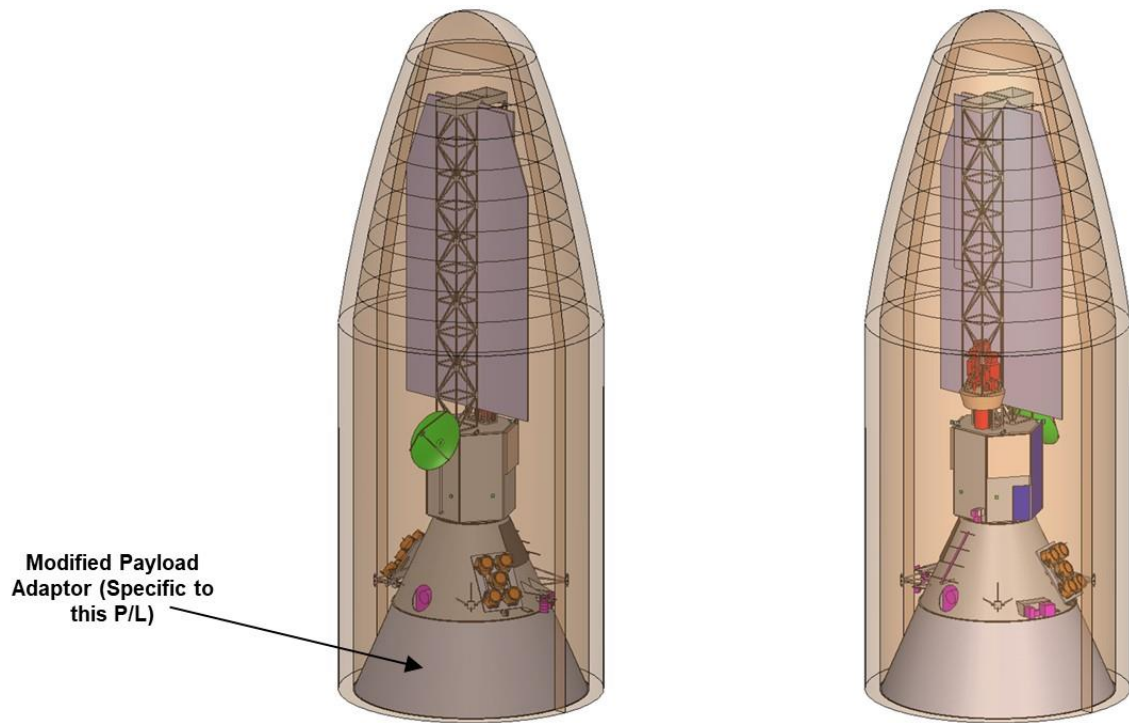


**Fig. 18 - Isometric views of the SERENE vehicle inside the Starship extended PLF**

While large, the payload specific PLA for this mission allows the lander/sample return vehicle to extend down into it thus reducing the overall height of the stack in the PLF and only adds to the launch mass, as it will not be carried through the mission. Fig. 19 below shows the series of adaptors required to carry the lander/sample return vehicle for the SERENE launch and mission. The lander/sample return vehicle is suspended from a cylindrical adaptor attached to the bus of the SERENE vehicle. This adaptor not only provides a volume for the thruster of the lander/sample return vehicle while attached, but also provides clearance between the bus and lander, allowing the

lander to fit within a conical adaptor that acts as the interface between the bus and PLA. In addition to acting as the interface between the bus and PLA, this conical adaptor also provides the surface area for mounting the parasitic load radiator, two thruster pallets, and most of the science instruments. There are two circular cutouts on opposite sides of the conical adaptor that allow the two Multi-Mission Radioisotope Thermoelectric Generators (MMRTG) to be loaded onto the lander/sample return vehicle while the vehicle is inside the PLF on the launch pad.

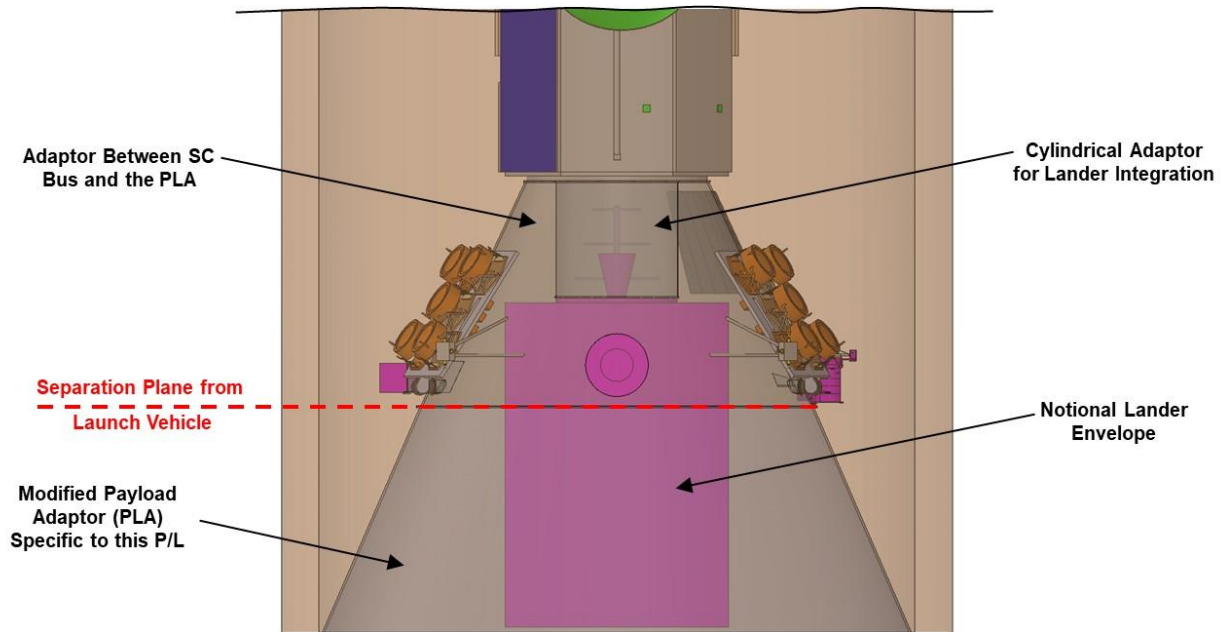


**Fig. 19 - Various adaptors for the SERENE vehicle launch and mission**

To fit within the PLF the SERENE vehicle must have several major components in a stowed position that will be deployed after separation with the PLA. These components include: the reactor power system boom; two radiator wings for the bus electronics, reactor Power Management and Distribution (PMAD) system, and the integrated Power and Processing Units (PPU) and Digital Control and Interface Units (DCIU) for the thrusters; the two Electric Propulsion (EP) thruster wings (each containing five NEXT-C thrusters); the magnetometer boom; the ice penetrating radar; a small portion of the commissioning solar array; and the 2.1-meter diameter Ka-band dish antenna/mapping radar. Fig. 20 below shows these deployable components in their stowed positions on the vehicle, while the overall stowed dimension of the vehicle is shown in Fig. 21 below.

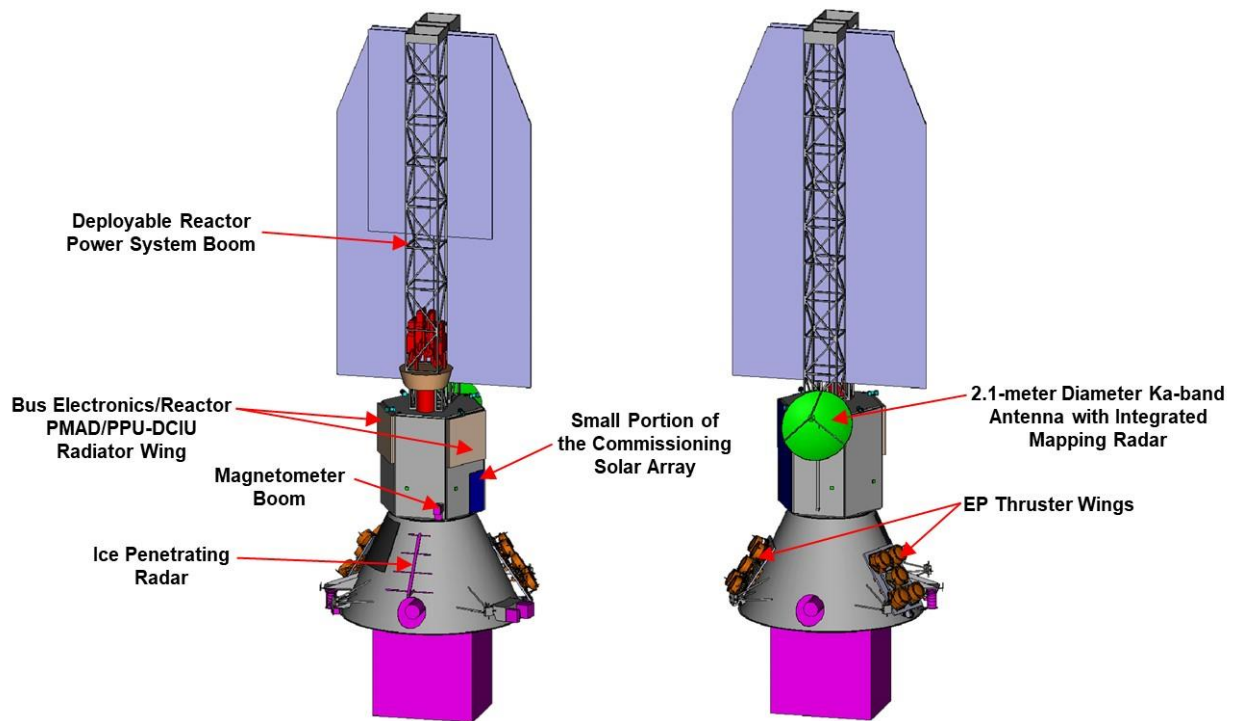


Fig. 20 - Deployable components on the SERENE vehicle

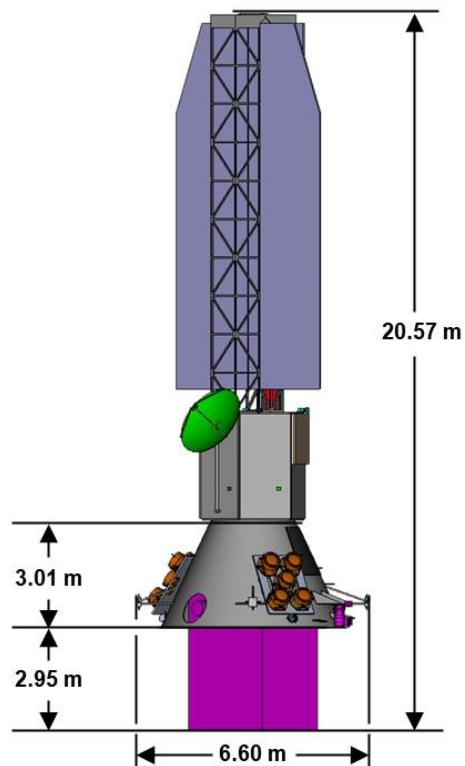


Fig. 21 - Stowed dimensions of the SERENE vehicle

The reactor power system boom is divided into two sections attached to one another by a hinge that allows the section containing the reactor, shield, Brayton units, and portion of the double-sided reactor/Brayton radiator to swing 180-degrees upon deployment. This concept allows the reactor and shield to be directly above the spacecraft bus at launch, thus lowering the Center-of-Gravity (CG) while in the PLF. When deployed, the boom provides a more than 21-meter separation distance between the reactor shield and spacecraft bus, thus reducing the reactor radiation dosage seen by the electronics located on the bus. An additional structure is located on the top of the spacecraft bus for the sole purpose of securing the reactor end of the boom when stowed for launch.

The two radiator wings used for thermal control of the bus electronics, reactor PMAD, and the integrated PPU/DCIUs are located on the spacecraft bus and are each comprised of two equal sized double-sided radiator panels connected to one another with hinges. In their stowed configuration, each of the two panels contained in a single wing are folded on top of one another and lay flat across a side of the hexagonal bus structure. Upon deployment, the panels fold out in an accordion style like the radiators on the International Space Station (ISS). Once deployed, the radiator wings extend out from opposite edges of the hexagonal bus and are parallel to the reactor/Brayton radiator panel, thus reducing the view they have of one another and maximizing their heat rejection capabilities.

When stowed, the two EP thruster wings run along the outer face of the conical structure that provides the interface between the spacecraft bus and the PLA. Each wing will be secured for launch to this conical structure. A two-axis gimbal located near the base of the cone is used to rotate the wing a little more than 90-degrees until the wing is perpendicular to the longest centerline of the conical structure. This gimbal, in addition to performing the deployment of the wing, also allows two-axis control of the full wing to position the thrusters as needed throughout the mission. Note that the two thruster wings are not 180-degrees from one another around the conical structure but are 140-degrees from one another. This helps to ensure that the thrust vector aligns more closely with the vehicle's CG as the boom containing the heavy reactor, shield, and Brayton units does not run along the centerline of the bus and conical adaptor, but is offset to ensure that when the boom is stowed it will fit within the PLF.

A deployable truss mast is used for deploying the two magnetometers, one near the middle of the mast and another at the end, away from the body of the spacecraft. This deployable mast is located at the bottom right corner of one of the side closeout panels on the hexagonal bus structure. It is shifted to the right side of this closeout panel to avoid blockage by the ice penetrating radar allowing both magnetometers an unobstructed view of Enceladus. The mast deployed straight in a direction perpendicular to the face to which it is mounted.

The ice penetrating radar is stowed up along the conical structure just above one of the circular cutouts in the cone that allows MMRTG loading while inside the PLF. For launch, this radar will be tied down to the conical structure. A spring lock mechanism at the base of the radar is utilized for deployment, rotating the radar a little more than 90-degrees to a position that is perpendicular to the longest centerline of the vehicle.

To obtain the full 5.2 m<sup>2</sup> area required for the commissioning solar array, a small portion of that area needs to be stowed for launch and deployed after separation from the launch vehicle. Most of the commissioning solar array (4.5 m<sup>2</sup>) is fixed and mounted directly to one of the side closeout panels on the hexagonal bus, taking up the entire area of the panel. The remaining 0.7 m<sup>2</sup> of the array is attached by a spring lock hinge to the fixed portion and is tied down for launch to the adjacent side closeout panel on the hexagonal bus. Upon deployment, this hinge will rotate the small section of array until the solar cells are parallel to the cells on the fixed array to avoid any cosine losses when positioning the vehicle to point the array. This small section of array was necessary to avoid increasing the size of the bus to accommodate the full 5.2 m<sup>2</sup>. Further analysis could be done to find a more optimal solution.

Lastly, the 2.1-meter diameter Ka-band dish antenna/mapping radar is located out at the end of a 2.29-meter-long boom that, when stowed, runs parallel to an open face on the hexagonal bus. This boom will be secured to the side of the bus for launch. Upon deployment, a spring lock mechanism at the base of the boom will rotate the boom 90-degrees to its deployed position. By utilizing a boom, the antenna will have fewer obstructions by other components on the vehicle, as well as the conical structure at the base of the vehicle, when performing communications or mapping of Enceladus. Fine pointing of the antenna is performed by a 2-axis gimbal located between the top of the boom and the dish antenna. In an event the antenna is blocked by another component on the vehicle, the whole

vehicle will rotate to allow for unobstructed communications. It should be noted that a small X-band medium gain antennas is mounted to the outer facing side of the feed on the dish antenna allowing for both antennas to be pointed in the same direction during communications.

Fig. 22 below shows the deployment of the components discussed above (except for the small portion of the commissioning solar array) with the fully deployed dimensions of the vehicle shown in Fig. 23 below. It should be noted that all deployed components on the SERENE vehicle remain inside the 17-degree half angle shield cone generated by the reactor shield. This cone angle was minimized to an angle that allows the double-sided reactor/Brayton radiators to stow within the PLF while avoiding the need for deployable radiator panels to achieve the required effective radiator area.

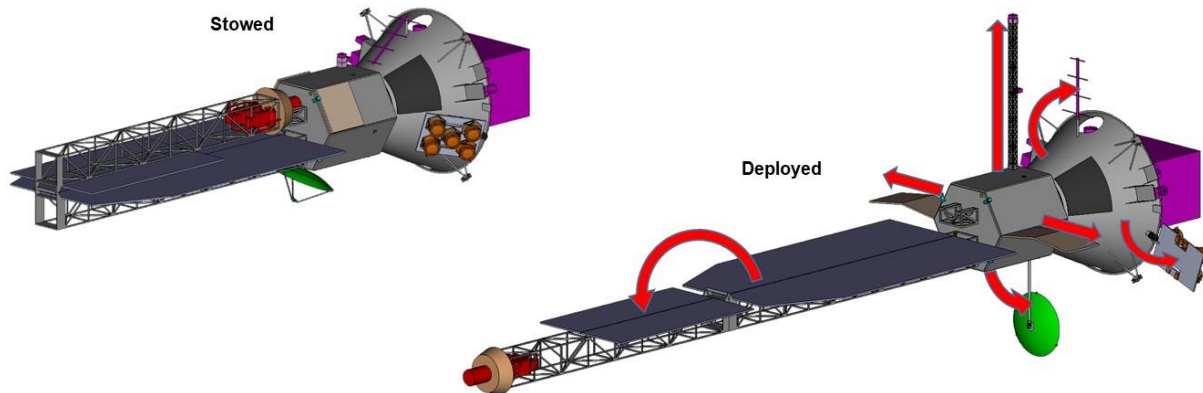


**Fig. 22 - Deployment concept for the deployable components on the SERENE vehicle**

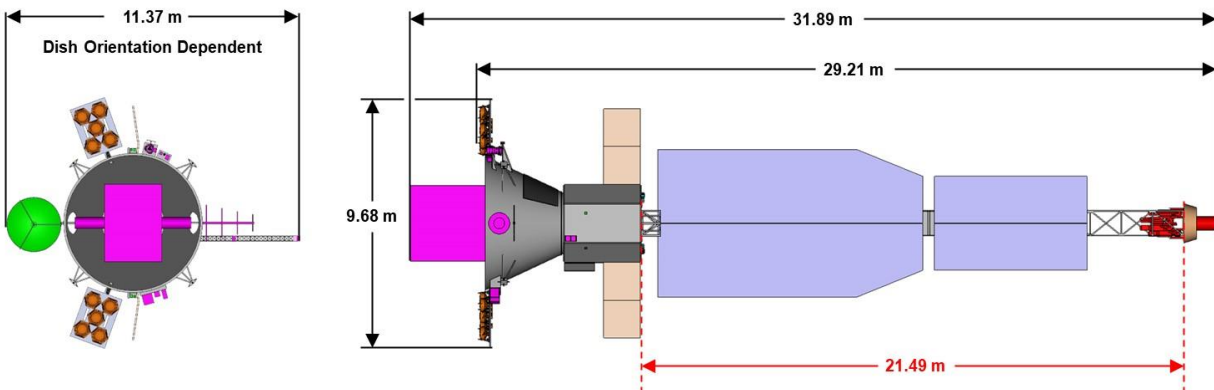


**Fig. 23 - Deployed dimensions of the SERENE vehicle**

In addition to the deployed components discussed earlier, along with the reactor, shield, and Brayton units, there are many other components located externally to the spacecraft bus. These components include: four star tracker optical heads; four Reaction Control Systems (RCS) pods containing four thrusters each; eight X-band patch antennas; two UHF patch antennas; a parasitic load radiator; and the Xenon flow controllers, NEXT-C thrusters, and single axis gimbals contained on the thruster wings. All these components, in addition to a few of the deployable components discussed earlier, can be seen in Fig. 24 below. Given the relatively large number of fixed science instruments on the SERENE vehicle, they will be discussed separately later in this section.

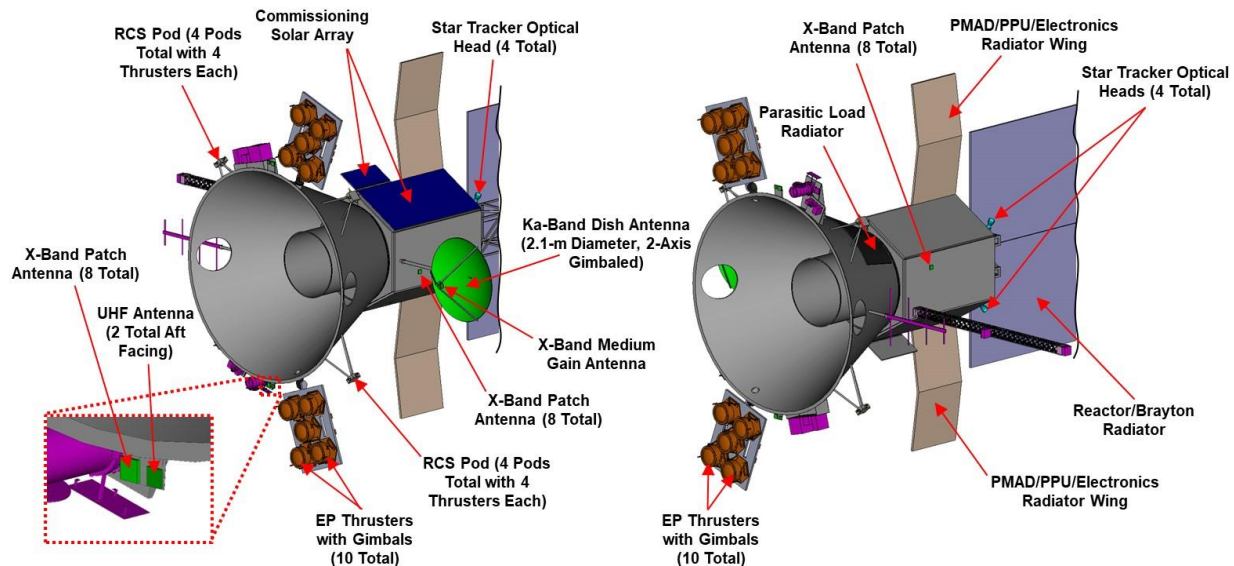


**Fig. 24 - External components contained on the SERENE vehicle**

The four star tracker optical heads are all mounted on the outer edges of the end panel of the hexagonal bus to which the reactor system boom is mounted. They are oriented in such a manner that the centerlines of the field-of-view for each extend outward in a direction parallel to the panel to which they are mounted and are also perpendicular to the tracker next to it around the perimeter of the panel (field-of-views at 0, 90, 180, and 270 degrees around the bus). There are no blockages to the field-of-view for any of the four star trackers by any other components on the spacecraft.

The four RCS pods are located on booms 90-degrees from one another near the base of the conical structure that partially encapsulates the lander/sample return vehicle. Use of the booms provides a better moment arm for the thrusters to change the attitude of the vehicle as there are no RCS thrusters located out at the reactor end of the vehicle to assist in maneuverability. Each pod contains four individual thrusters with one thruster pointed towards the reactor end, another pointed opposite of that, and two opposing thrusters in the radial direction. With this configuration of the 16 RCS thrusters, the RCS system will have full control over the vehicle's attitude.

Four of the eight X-band Patch antennas are located on four separate side panels on the hexagonal bus structure. The two side panels that do not contain an X-band patch antenna are the panels containing the commissioning solar array and the panel directly opposite that. The end closeout panel to which the reactor boom is mounted contains two of the eight X-band patch antennas with each antenna located on opposite sides of the reactor/Brayton radiator panels to allow one to have a view for communications if the other is blocked by the radiator panels. Finally, the remaining two X-band patch antennas are mounted to dedicated antenna structures located on opposite sides from one another near the base of the conical structure. Both antennas are pointed in the aft direction of the vehicle (direction opposite to that of the reactor system). The use of two antennas in this direction allows at least one to have a view of the target for communications when the other is blocked by the lander/sample return vehicle while it is still attached to the SERENE vehicle. When combined, given the location and orientation of all eight X-band patch antennas, a near full spherical coverage is achieved ensuring at least one or more antennas have a view of the target even in the event the vehicle is tumbling.

Located next to the two aft facing X-band patch antennas, on the dedicated structures added on opposite sides from one another at the base of the conical structure, are the two UHF patch antennas. Both UHF patch antennas are pointed in the aft direction of the vehicle as this is the end of the vehicle that will be pointed towards Enceladus and these two antennas are used for the uplink from the lander/sample return vehicle. The use of two antennas not only provides redundancy but allows at least one of the antennas to have a view for the uplink in the event the other is partially blocked by the conical structure if the vehicle is not pointed exactly in the nadir direction.

The 2.5 m<sup>2</sup> parasitic load radiator is mounted directly to the outer surface of the conical structure on the base of the SERENE vehicle. It is located near the top of the conical structure and at a position around the circumference to minimize the heat being rejected from heating other components on the vehicle.

As discussed briefly earlier in this section, the two thruster platforms each contain five NEXT-C thrusters, each mounted on their own dual axis gimbal, as well as five xenon flow control modules, one for each thruster. The thrusters and their gimbals are mounted to one side of the thruster platform structure in a 2-1-2 configuration as can be seen in Fig. 25 below. This thruster layout helps to minimize the length and width of the thruster platform given the odd number of thrusters on each. Mounted to the opposite side of the thruster platform structure are the five xenon flow control modules, with two on one edge of the panel and the remaining three on the opposite edge. Two cutouts are in the panel to allow the required lines to run between the xenon flow control modules and their relative thruster on the opposite side. It should be noted that while the thruster plumes have a 5-degree half-angle and the gimbals have a maximum gimbal angle of +/- 19 degrees, given the location of the two thruster platforms, there is no plume impingement on any other components on the vehicle including the notional lander/sample return vehicle when still attached. This can be seen in Fig. 26 below.

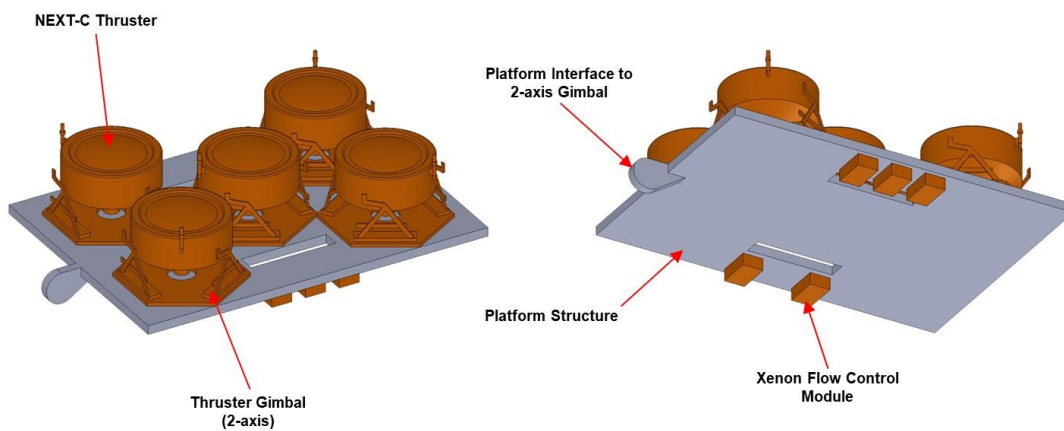


**Fig. 25 - Components located on the thruster platforms**

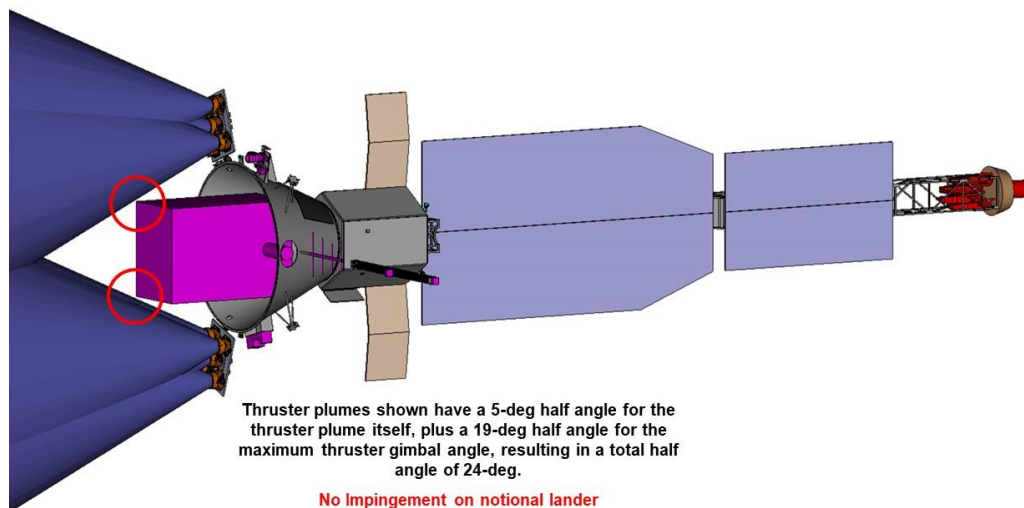


**Fig. 26 - NEXT-C thruster plumes plus gimbal angle**

In addition to the deployable science instruments discussed earlier (ice penetrating radar, magnetometers, and the mapping radar shared with the 2.1-m diameter dish) there are several other fixed science instruments located along the base of the conical structure. These include: a Narrow Angle Camera (NAC); and electrochemical sensor array; a Wide-Angle Camera (WAC); a thermal emission spectrometer; and Ultraviolet (UV) spectrograph; an Infrared (IR) imager; and a high resolution mass spectrometer. All these science instruments can be seen in Fig. 27 below.

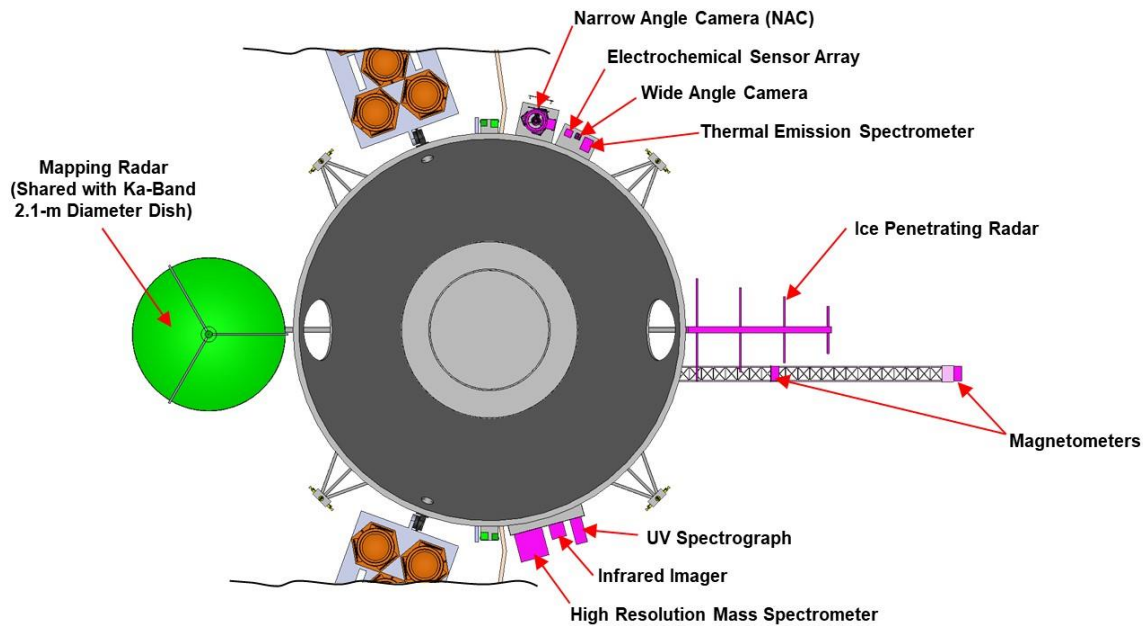


**Fig. 27 - Science instruments on the SERENE vehicle**

Like the two aft facing X-band patch antennas and the two aft facing UHF patch antennas, additional structure has been added at the base of the conical structure for mounting all the fixed science instruments as they too need to be pointed in the aft direction of the vehicle which will be the nadir direction while orbiting Enceladus during science operations. The NAC (based on the Long-Range Reconnaissance Imager (LORRI)) is mounted to its own dedicated structure a bit further up the conical structure due to its length being longer than the other science instruments. An additional structure located right next to the NAC structure, though closer to the base of the cone, is used to mount the electrochemical sensor array, WAC, and the thermal emission spectrometer. An additional structure, located about 140-deg from the WAC and NAC structures at the base of the conical structure contains the UV spectrograph, IR imager, and the high-resolution mass spectrometer. The location of all seven of these science instruments was selected to not only allow them all to point in the nadir direction while in orbit around Enceladus, but to minimize impingement from the RCS thruster plumes, avoid the EP thruster platforms, and avoid blocking the 2.1-m diameter dish antenna and the ice penetrating radar.

Those components located inside the spacecraft bus include: the Xenon tank and integrated PPU/DCIU units of the EP System; the power electronics boxes, battery, and reactor PMAD boxes of the Electrical Power System (EPS); the Ka-band, S-band, and UHF electronics of the Communications System; the reaction wheels, Inertial Measurement Units (IMUs), and star tracker electronics of the Attitude Determination and Control (AD&C) System; the Command and Data Handling (C&DH) enclosure and the driver enclosure of the C&DH System; the magnetometer electronics box for the Science System; and the hydrazine tanks of the RCS System. All these components contained inside the hexagonal bus structure can be seen in Fig. 28 and Fig. 29 below.

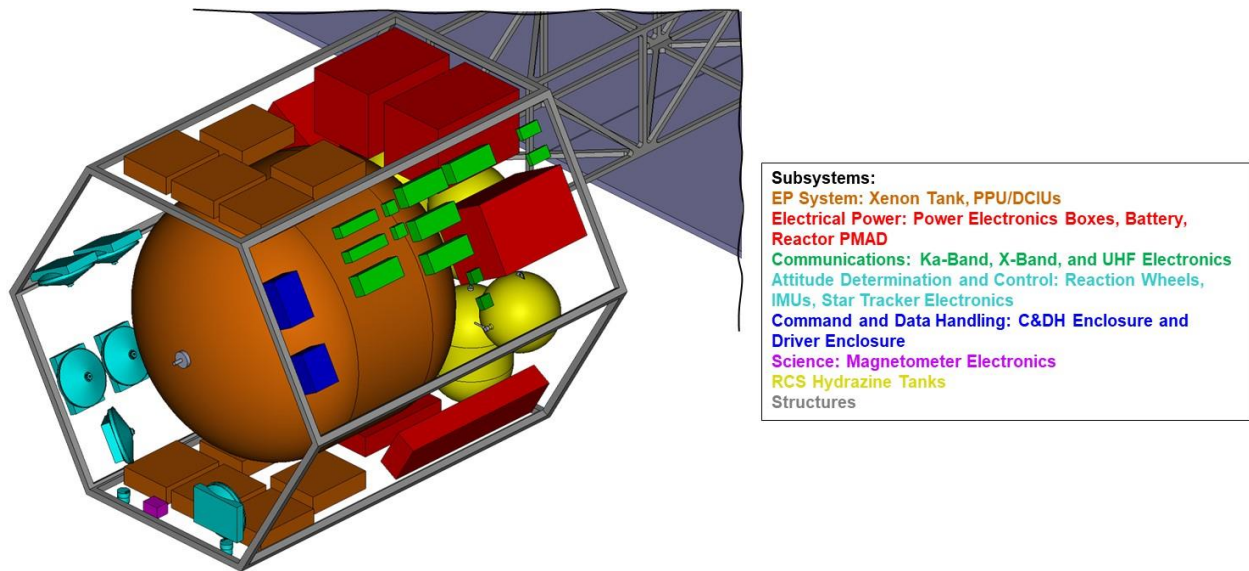


**Fig. 28 - Internal bus components on the SERENE vehicle**

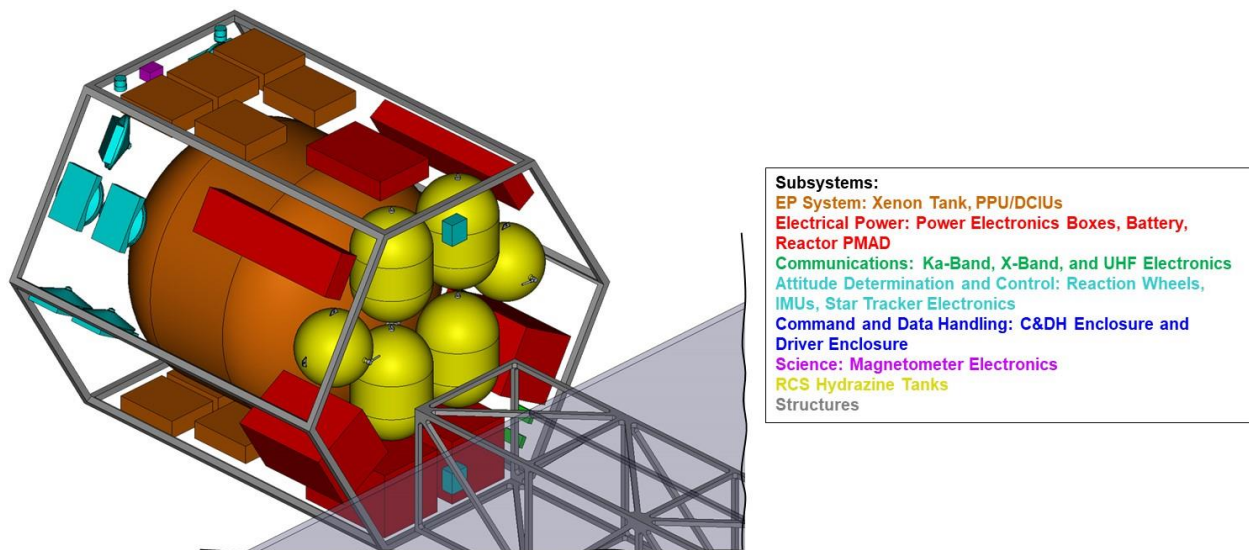


**Fig. 29 - Additional view of the internal bus components on the SERENE vehicle**

Located in the middle and near the bottom of the bus structure is the custom Xenon tank. This location ensures that the CG for the tank is located along the centerline of the bus structure and as low on the vehicle as possible when stowed for launch. Structures for mounting the tank to the bus structure were not designed during this study as the tank is a custom tank that can be designed to include polar boss mounts, mounting tabs around the circumference of the tank, or any other mounting structure that would work best with the layout of the internal bus components as the design is further developed. The ten PPU/DCIU boxes are mounted to the inside of two of the side closeout panel with five boxes on each of the panels. These two panels are opposite one another on the hexagonal bus structure with the PPU/DCIU boxes located near the bottom of the panels to put them closer to the thruster platforms.

The two power electronics boxes, lithium-ion battery, and the four reactor system PMAD boxes are all mounted directly to the inside of the side closeout panels on the hexagonal bus. All six boxes are located above the cylindrical section of the xenon tank to not only put them slightly closer to the reactor system, but also preventing the larger reactor PMAD boxes from contacting the tank which would drive the bus structure to a larger diameter. Two of the four reactor PMAD boxes are mounted side by side on the same closeout panel, with each of the other two boxes

mounted to the two adjacent panels. The two side panels adjacent to these each have a power electronics box mounted to them. The last side panel, opposite the panel with the two reactor PMAD boxes, contains the lithium-ion battery. Placement of all these boxes can be further examined to help reduce harness lengths as well as balance out their impact on the CG location for the overall vehicle.

All the Ka-band, S-band, and UHF electronics are mounted to the inside of the side closeout panel adjacent to the Ka-band dish antenna panel. By keeping all the communications electronics on the same panel, assembly and testing of the system will be easier. Placement of the communications electronics can be further optimized to help minimize the length of harnessing and cabling for the system (such as moving the UHF electronics to the bottom panel on the bus, closer to the antennas), though the X-band harnessing will still have to run from the electronics to the eight patch antennas located throughout the bus.

Two of the six reaction wheels of the AD&C System are mounted directly to the inside of one of the side closeout panels near the base of the bus with their axes of rotation parallel to one another but perpendicular to the longest centerline of the vehicle (running from the base of the bus to the reactor). Two wheels are required in this orientation as this is one of the largest moments of inertia axes direction. Similarly, two other wheels are located nearby with their axes of rotation parallel to one another, perpendicular to the first two wheels discussed, and perpendicular to the longest centerline of the vehicle. Again, two wheels are required in this direction as this is another of the largest moments of inertia axes direction. A fifth wheel is mounted to the bottom closeout panel of the bus with its axis of rotation parallel to the longest centerline of the vehicle and perpendicular to the previous four wheel's axes of rotation. A sixth wheel is oriented so that its axis of rotation is 45-degrees from the previous five wheels' axes of rotation, thus providing partial control in all three axes. While three of the wheels are mounted directly to closeout panels, the other three will need additional structure for mounting (not designed here) as the hexagonal shape of the bus does not allow direct mounting of all six wheels to the panels themselves to obtain the orientations necessary for full vehicle control. The two IMUs are both mounted to the inside of the side closeout panel that contains the battery down near the bottom of the bus. Both star tracker electronics boxes are mounted to the inside of the end closeout panel that contains the reactor boom as the four optical heads are located on the outside of this same panel. They are located near opposite edges of the panel as the optical heads are spread relatively equally around the perimeter of the panel and a single electronics box controls two optical heads, thus reducing the harness lengths between the optical heads and their relative electronics box.

The C&DH enclosure and the driver enclosure are mounted directly to the inside of the side closeout panel that contains the communications electronics. Both boxes are located near the base of this side panel. Further analysis can be done to determine the best location to minimize the harness runs from these boxes to the other components.

The magnetometer electronics box is located between the two IMUs. It is mounted directly to the inside of this panel near the base of the bus, putting it close to the magnetometer boom mounted to the outside face of this same panel.

Lastly, the six hydrazine tanks for the RCS system consist of four cylindrical tanks and two spherical tanks. All six tanks are located above the xenon tanks and are assumed to be mounted to the hexagonal closeout panel on the top of the bus, though the structure to mount the tanks was not designed during this study. The use of the spherical tanks allows the vehicle to not only carry the required hydrazine load but allows all the tanks to fit within the bus without having to increase the bus size to accommodate them.

All component locations within the bus should be further analyzed to minimize the required harness lengths and ensure the vehicle's CG is in the best location for launch, thrusting phases, and maneuvering the vehicle with the RCS thrusters.

Transparent views of the bus end of the SERENE vehicle showing all the components located inside and outside the hexagonal bus and conical adaptor structure can be seen in Fig. 30 and Fig. 31 below.

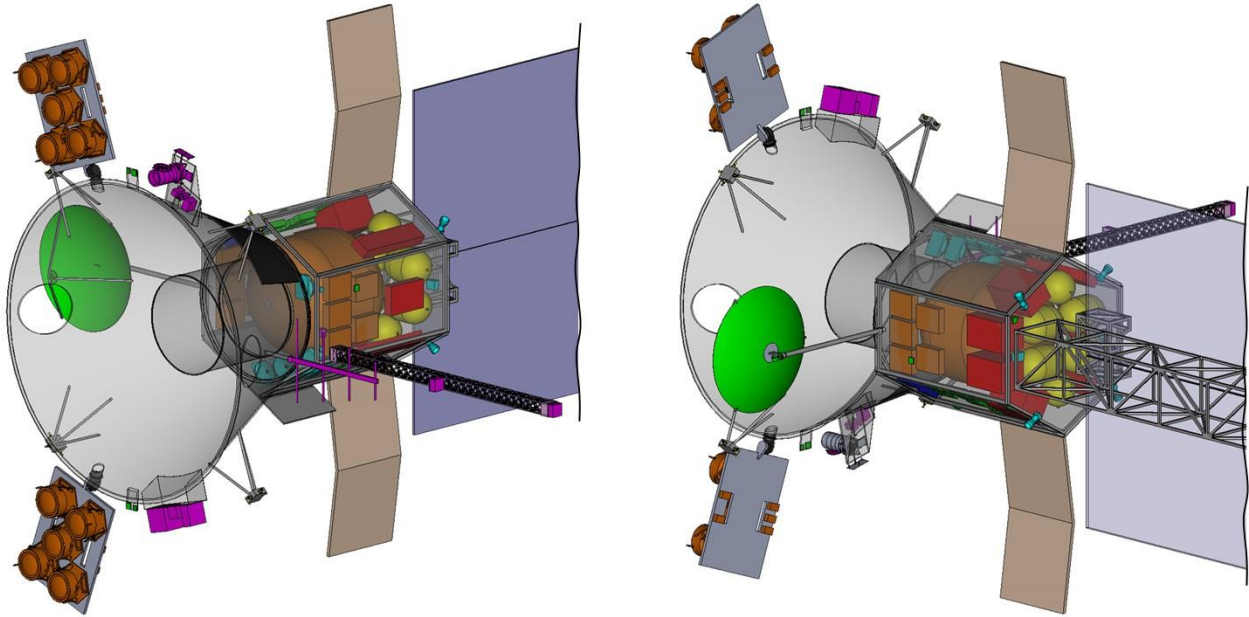


**Fig. 30 - Transparent views of the bus end of the SERENE vehicle**

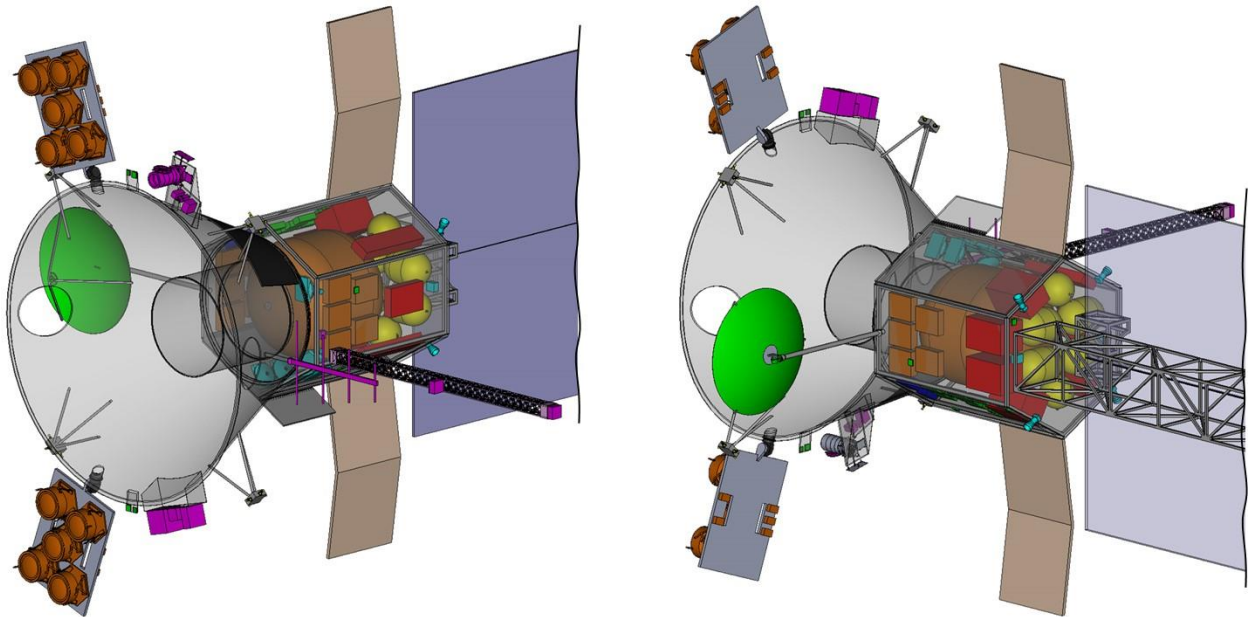


**Fig. 31 - Additional transparent views of the bus end of the SERENE vehicle**

## **5.2 Systems:**

The SERENE (including both the Orbilander and sample return capsule) has a dry mass of 13.8 t, including an aggregate 19% mass growth allowance (MGA) and 18% system-level margin, per AIAA S-120A-2015(2019) [12]. The inert mass of the system is 14.5 t and the wet mass at launch is 27.4 t. A breakdown of these masses is shown in Fig. 32.

| MEL Summary: Case 1_Enceladus Sample Return CD-2025-225 | NEP Vehicle     | Orbilander      | Sample Capsule  | TOTAL                |
|---------------------------------------------------------|-----------------|-----------------|-----------------|----------------------|
| Main Subsystems                                         | Basic Mass (kg) | Basic Mass (kg) | Basic Mass (kg) | Total Basic Mass(kg) |
| Science                                                 | 171.8           | 1909.0          | 49.4            | 2130.2               |
| Attitude Determination and Control                      | 82.8            | 0.0             | 0.0             | 82.8                 |
| Command & Data Handling                                 | 84.2            | 0.0             | 0.0             | 84.2                 |
| Communications and Tracking                             | 48.9            | 10.1            | 0.0             | 59.0                 |
| Electrical Power Subsystem                              | 4170.5          | 0.0             | 0.0             | 4170.5               |
| Thermal Control (Non-Propellant)                        | 1064.3          | 0.0             | 0.0             | 1064.3               |
| Propulsion (Chemical Hardware)                          | 115.7           | 0.0             | 0.0             | 115.7                |
| Propellant (Chemical)                                   | 475.1           | 3920.0          | 0.0             | 4395.1               |
| Propulsion (EP Hardware)                                | 1059.2          | 0.0             | 0.0             | 1059.2               |
| Propellant (EP)                                         | 9124.4          | 0.0             | 0.0             | 9124.4               |
| Structures and Mechanisms                               | 1400.0          | 0.0             | 0.0             | 1400.0               |
| <b>Element Total</b>                                    | <b>17796.9</b>  | <b>5839.1</b>   | <b>49.4</b>     | <b>23685.4</b>       |
| Element Dry Mass (no prop,consum)                       | 8197.4          | 1919.1          | 49.4            | 10165.9              |
| Element Propellant                                      | 9599.5          | 3920.0          | 0.0             | 13519.5              |
| Element Mass Growth Allowance (Aggregate)               | 1587.0          | 294.4           | 8.4             | 1889.8               |
| MGA Percentage                                          | 19%             | 15%             | 17%             | 19%                  |
| Predicted Mass (Basic + MGA)                            | 9784.4          | 2213.5          | 57.8            | 12055.7              |
| System Level Mass Margin                                | 1229.6          | 543.1           | 7.4             | 1780.1               |
| System Level Growth Percentage                          | 15%             | 28%             | 15%             | 18%                  |
| Element Dry Mass (Basic+MGA+Margin)                     | 11014.0         | 2756.6          | 65.2            | 13835.8              |
| Element Inert Mass (Basic+MGA+Margin)                   | 11673.2         | 2771.6          | 65.2            | 14509.9              |
| Total Wet Mass (Allowable Mass)                         | <b>20613.5</b>  | <b>6676.6</b>   | <b>65.2</b>     | <b>27355.3</b>       |

**Fig. 32 - System Masses for the SERENE Concept**

The power margins and power growth allowances (PGA) used are consistent with the AIAA-S-122 Electrical Power Systems for Unmanned Spacecraft guidelines [13]. The PGA is an applied factor onto all required loads, and the power margin is a calculated value that should be used to design the EPS in case of unexpected loads. Spacecraft loads were given a PGA of 30% with a power margin of 20%, and EP was given a PGA of 5% and power margin of 5%.

The SERENE concept used a modified Orbilander, as previously discussed. The primary modifications were the removal of orbital science instruments, since this functionality would be taken over by the NEP vehicle, removal of the high-gain antenna, as the lander would no longer need to communicate direct-to-Earth, and the addition of a sample return capsule and UHF communications system. Additionally, a representative bio barrier mass was added, as discussed further in the science section. Fig. 33 details the Orbilander assumptions and modifications, with the exception of the modified communications system and propellant. Communications changes are shown in Fig. 54 in the communications subsystem subsection and Orbilander propellant is shown in Fig. 34.

| Description                                | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |            |              |               |
|                                            |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Orbilander</b>                          |     |           | <b>5839</b>   | <b>5%</b>  | <b>294</b>   | <b>6133</b>   |
| <b>Modified Orbilander</b>                 |     |           | <b>1909.0</b> | <b>15%</b> | <b>293.5</b> | <b>2202.5</b> |
| <b>Orbilander Baseline</b>                 |     |           | <b>1753.0</b> | <b>15%</b> | <b>263.0</b> | <b>2016.0</b> |
| Instruments                                | 1   | 142.0     | 142.0         | 15%        | 21.3         | 163.3         |
| Mechanical                                 | 1   | 630.0     | 630.0         | 15%        | 94.5         | 724.5         |
| Landing Structure                          | 1   | 150.0     | 150.0         | 15%        | 22.5         | 172.5         |
| Avionics                                   | 1   | 15.0      | 15.0          | 15%        | 2.3          | 17.3          |
| Propulsion, including residuals            | 1   | 357.0     | 357.0         | 15%        | 53.6         | 410.6         |
| Power                                      | 1   | 174.0     | 174.0         | 15%        | 26.1         | 200.1         |
| Guidance, Navigation, and Control          | 1   | 66.0      | 66.0          | 15%        | 9.9          | 75.9          |
| Thermal                                    | 1   | 110.0     | 110.0         | 15%        | 16.5         | 126.5         |
| RF, Telecom                                | 1   | 109.0     | 109.0         | 15%        | 16.4         | 125.4         |
| <b>Orbilander Baseline</b>                 |     |           | <b>124.0</b>  | <b>15%</b> | <b>18.6</b>  | <b>142.6</b>  |
| Harness                                    | 1   | 124.0     | 124.0         | 15%        | 18.6         | 142.6         |
| <b>Orbilander Modifications</b>            |     |           | <b>32.0</b>   | <b>37%</b> | <b>12.0</b>  | <b>44.0</b>   |
| Estimated removed hardware                 | 1   | -49.4     | -49.4         | 17%        | -8.4         | -57.8         |
| ELSAH Bio Barrier                          | 1   | 81.4      | 81.4          | 25%        | 20.4         | 101.8         |

Fig. 33 - Orbilander MEL Modifications

| Description                                | QTY | Unit Mass | Basic Mass    | Growth    | Growth     | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|-----------|------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |           |            |               |
|                                            |     | (kg)      | (kg)          | (% )      | (kg)       | (kg)          |
| <b>Propellant (Chemical)</b>               |     |           | <b>3920.0</b> | <b>0%</b> | <b>0.0</b> | <b>3920.0</b> |
| <b>Main Engine Propellant</b>              |     |           | <b>3920.0</b> | <b>0%</b> | <b>0.0</b> | <b>3920.0</b> |
| <i>Fuel</i>                                |     |           | 2416.0        | 0%        | 0.0        | 2416.0        |
| Fuel Usable                                | 1   | 2416.0    | 2416.0        | 0%        | 0.0        | 2416.0        |
| <i>Oxidizer</i>                            |     |           | 1489.0        | 0%        | 0.0        | 1489.0        |
| Oxidizer Usable                            | 1   | 1489.0    | 1489.0        | 0%        | 0.0        | 1489.0        |
| <i>Main Engine Pressurant</i>              | 1   |           | 15.0          | 0%        | 0.0        | 15.0          |
| Helium                                     | 1   | 15.0      | 15.0          | 0%        | 0.0        | 15.0          |

Fig. 34 - Orbilander Propellant MEL

The sample return capsule used as representative in this study was designed previously for a Titan Sample Return mission concept [14]. Its mass is shown in Fig. 35.

| Description                                | QTY | Unit Mass | Basic Mass  | Growth     | Growth     | Total Mass  |
|--------------------------------------------|-----|-----------|-------------|------------|------------|-------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |             |            |            |             |
|                                            |     | (kg)      | (kg)        | (%)        | (kg)       | (kg)        |
| <b>Sample Capsule</b>                      |     |           | <b>49.4</b> | <b>17%</b> | <b>8.4</b> | <b>57.8</b> |
| <b>Science</b>                             |     |           | <b>49.4</b> | <b>17%</b> | <b>8.4</b> | <b>57.8</b> |
| <b>Science package group one</b>           |     |           | <b>49.4</b> | <b>17%</b> | <b>8.4</b> | <b>57.8</b> |
| Sample Capsule                             | 1   | 49.4      | 49.4        | 17%        | 8.4        | 57.8        |

Fig. 35 - Sample Return Capsule MEL, based on previously design Titan Sample Return Mission

### 5.3 Science

The icy moon ocean worlds that orbit Saturn have emerged as compelling targets for the search for life within our Solar System given the availability of liquid water, energy sources, and, for some moons, the presence of organic materials. As life detection becomes a central goal for planetary science missions, science payloads and mission architectures need to meet the high standard of evidence required for definitive life detection [15]. Multiple, independent lines of evidence coupled with a thorough characterization of the environment will likely require a diverse suite of scientific instruments deployed through a combination of orbital, *in situ*, and return sample science missions. While successive, progressive missions are practicable on inner planets such as Mars, the long transit time to the outer planets limits the feasibility of this approach for Saturn's ocean worlds. By leveraging NEP, SERENE can combine an *in situ* and return sample science mission to Enceladus, a high value astrobiological target, with an orbital tour of Rhea, Dione, Tethys, and Titan, an important target for understanding prebiotic chemistry.

Enceladus contains the necessary ingredients for life, including liquid water, a suite of bioessential elements, and energy sources. Further, direct sampling of water vapor plumes by the Cassini spacecraft revealed the presence of organic molecules, biologically available nitrogen, carbon dioxide, sodium, potassium, hydrogen, and importantly phosphate, indicating its possible habitability [16] [17] [18]. Given its high astrobiological potential, it was selected as the target for the *in situ* and sample return science phase of SERENE.

The *in situ* science objectives for Enceladus are outlined in Table 10, and were adapted from the Enceladus Orbilander mission concept [19]. Science objectives for this phase of the mission will provide a preliminary analysis of potential biosignatures (Goals 1-3) and a thorough investigation of the habitability and preservation potential of the environment (Goals 4 -5). This will provide invaluable contextual data to prioritize future science investigation on the mass-limited returned sample and help interpret possible signals of life.

Table 10 - High Level Science Traceability Matrix for In Situ Science at Enceladus

| Science Goal                                                         | Measurement                                                              | Reference Instrument                                                          |
|----------------------------------------------------------------------|--------------------------------------------------------------------------|-------------------------------------------------------------------------------|
| 1. Characterize the organic inventory of plume and surface materials | Abundance and distribution of organic species including isomers          | High Resolution Mass Spectrometer (HRMS) – Heritage: MASPEX on Europa Clipper |
|                                                                      |                                                                          | Separation Mass Spectrometer (SMS) – Heritage: SAM on Curiosity               |
|                                                                      | Relative abundances of biologically relevant molecules and their isomers | Micro Capillary Electrophoresis Laser Induced Fluorescence – In development   |
|                                                                      |                                                                          | SMS                                                                           |
| 2. Search for evidence of cells in plume material                    | Morphology of particles                                                  | Microscope – Heritage MECA on Phoenix                                         |
| 3. Search for evidence of genetic material                           | Presence of polyelectrolytes                                             | Nanopore – In development, i.e., ELIE instrument concept [20]                 |
| 4. Determine the physical and chemical environment of the ocean      | Concentrations of redox ions, inorganic species and chemical oxidants    | HRMS                                                                          |
|                                                                      |                                                                          | Electrochemical sensor array (ESA) – Heritage: WCL on Phoenix                 |

|                                           |                                                                       |                                            |
|-------------------------------------------|-----------------------------------------------------------------------|--------------------------------------------|
| 5. Characterize active geologic processes | Body wave arrival times, tide-induced displacement, free oscillations | Seismometer – Heritage: SEIS-SP on InSight |
|                                           | Noble gas abundance                                                   | HRMS                                       |

The next phase of SERENE, after the sample vehicle begins its return to Earth, involves an extended moon tour with a one-year orbit of Tethys, Dione, and Rhea with a final two-year orbit of Titan. Science objectives for this phase of the mission are given in Table 11. and were adapted from the Enceladus Orbilander Report and the Titan Orbiter and Probe report [19] [21].

**Table 11 - Science Traceability Matrix for Orbital Science at Enceladus, Rhea, Dione, Tethys, and Titan**

| Science Goal                                                                                              | Measurement                                                     | Selected Instrument                                                                         |
|-----------------------------------------------------------------------------------------------------------|-----------------------------------------------------------------|---------------------------------------------------------------------------------------------|
| Characterize the structure and dynamics of the crust and subsurface ocean and search for active processes | Presence of liquid reservoirs                                   | Radar sounder (RS) – REASON VHF component on Europa Clipper<br>Magnetometer – MESSENGER MAG |
|                                                                                                           | Ice crust thickness                                             | RS, Magnetometer                                                                            |
|                                                                                                           | Ocean conductivity                                              | Magnetometer                                                                                |
|                                                                                                           | Surface thermal anomalies                                       | Thermal Emission Spectrometer (TES) – Heritage: MERTIS on BepiColumbo                       |
|                                                                                                           | Times-series imaging of surface                                 | Near Angle Camera (NAC) – Heritage: L’LORI on Lucy                                          |
|                                                                                                           |                                                                 | Wide Angle Camera (WAC) – MARCI on MRO                                                      |
|                                                                                                           |                                                                 | Infrared Imager (IR) – From Titan Orbiter and Probe Report                                  |
| Characterize the chemical and physical features of the surface                                            | Map surface features                                            | Radar Instrument (RI) – Heritage: RADAR on Cassini                                          |
|                                                                                                           | Characterize surface composition                                | Ultraviolet Spectrograph (UVS) – Heritage: UVS on Europa Clipper<br>IR, TES                 |
|                                                                                                           | Characterize plume composition and structure (where applicable) | HRMS, ESA, TES                                                                              |
|                                                                                                           | Image surface                                                   | NAC, WAC                                                                                    |
| Characterize atmosphere (where applicable)                                                                | Atmospheric structure and composition                           | UVS, IR                                                                                     |
| Identify landing site – Enceladus only                                                                    | Ice thickness and hazard analysis                               | NAC, WAC, TES                                                                               |

In addition to science objectives, a preliminary planetary protection plan was considered. SERENE is classified as a Class V restricted mission, requiring the highest level of planetary protection considerations [22]. Specific planetary protection requirements are established on a mission-by-mission basis with input from the planetary protection office, however for the purpose of this work we included several likely planetary protection considerations including a biobarrier to limit contamination of Enceladus by Earth life [23] and a landing site that minimizes the possibility of interacting with the subsurface ocean [19]. The mission design also includes active heating (250 C for at least 6 hours) of the sample return capsule to limit backwards planetary protection [24] [25]. However, future work will require a more detailed backwards planetary protection plan that will likely be influenced by the data returned from the *in situ* portion of the mission.

The MEL for the science subsystem can be seen in Fig. 36.

| Description                                | QTY | Unit Mass | Basic Mass   | Growth     | Growth      | Total Mass   |
|--------------------------------------------|-----|-----------|--------------|------------|-------------|--------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |              |            |             |              |
|                                            |     | (kg)      | (kg)         | (% )       | (kg)        | (kg)         |
| <b>Payload</b>                             |     |           | <b>184.6</b> | <b>28%</b> | <b>51.5</b> | <b>236.0</b> |
| <b>Orbital Science</b>                     |     |           | <b>175.3</b> | <b>28%</b> | <b>48.6</b> | <b>223.9</b> |
| Ice Penetrating Radar                      | 1   | 12.0      | 12.0         | 30%        | 3.6         | 15.6         |
| Mapping Radar                              | 1   | 41.4      | 41.4         | 30%        | 12.4        | 53.9         |
| High Res Mass Spec                         | 1   | 20.0      | 20.0         | 30%        | 6.0         | 26.0         |
| Infrared Imager                            | 1   | 57.0      | 57.0         | 30%        | 17.1        | 74.1         |
| UV Spectrograph                            | 1   | 20.0      | 20.0         | 10%        | 2.0         | 22.0         |
| Magnetometer sensor                        | 2   | 0.2       | 0.4          | 30%        | 0.1         | 0.5          |
| Thermal Emission Spectrometer              | 1   | 4.0       | 4.0          | 30%        | 1.2         | 5.2          |
| NAC                                        | 1   | 20.0      | 20.0         | 30%        | 6.0         | 26.0         |
| WAC                                        | 1   | 0.5       | 0.5          | 30%        | 0.2         | 0.7          |
| <b>Orbital Science Part 2</b>              |     |           | <b>9.2</b>   | <b>31%</b> | <b>2.9</b>  | <b>12.1</b>  |
| Electrochemical Sensor Array               | 1   | 4.0       | 4.0          | 30%        | 1.2         | 5.2          |
| Magnetometer Electronics                   | 1   | 0.8       | 0.8          | 30%        | 0.3         | 1.1          |
| Magnetometer Boom                          | 1   | 4.0       | 4.0          | 30%        | 1.2         | 5.2          |
| Magnetometer Cabling                       | 1   | 0.4       | 0.4          | 50%        | 0.2         | 0.6          |

**Fig. 36 - Science Subsystem MEL**

#### 5.4 Nuclear Power System

The NEP power system consists of the nuclear reactor, shield, power conversion, heat rejection, and PMAD. The SERENE mission dictated a nominal output power from the NEP power system of 44 kWe at 120 VDC to accommodate ion thruster and spacecraft bus operations including PGA and margin. The design approach assumed a derivative version of the NASA FSP Government Reference Design (GRD) to minimize the need for a new reactor development. Despite the major differences between the FSP and NEP missions, the basic reactor power system is fundamentally the same.

The resulting SERENE NEP power system uses a hydride moderated, HALEU reactor with Uranium-Nitride (UN) fuel, sodium heat pipe cooling, and Brayton power conversion. Brayton was selected as the primary power conversion choice over Stirling based on the goal for system extensibility to higher NEP power levels. Heat rejection is provided by composite radiators with embedded water heat pipes and a pumped-water heat transport loop. The baseline design assumes four independent power strings at 25% power each as shown in Fig. 37, similar to the FSP GRD. A power string consists of a single Brayton unit, dedicated water heat rejection loop, and dedicated PMAD power channel that delivers power to the spacecraft bus. Failure of a single power string would result in no less than 75% power output from the NEP power system. Several minor deviations relative to the FSP GRD system were implemented. The radiation shield design was revised for the NEP vehicle geometry and payload dose limits. The PMAD approach was modified to transmit the 240 Vrms, 2 kHz alternator output power 25 m down the boom and convert it to 120 Vdc before terminating at the spacecraft power distribution unit. The Brayton units are designed to operate at a reactor outlet temperature of 1100K and a compressor inlet temperature of 390K, producing a gross output power of 12.8 kWe per unit.

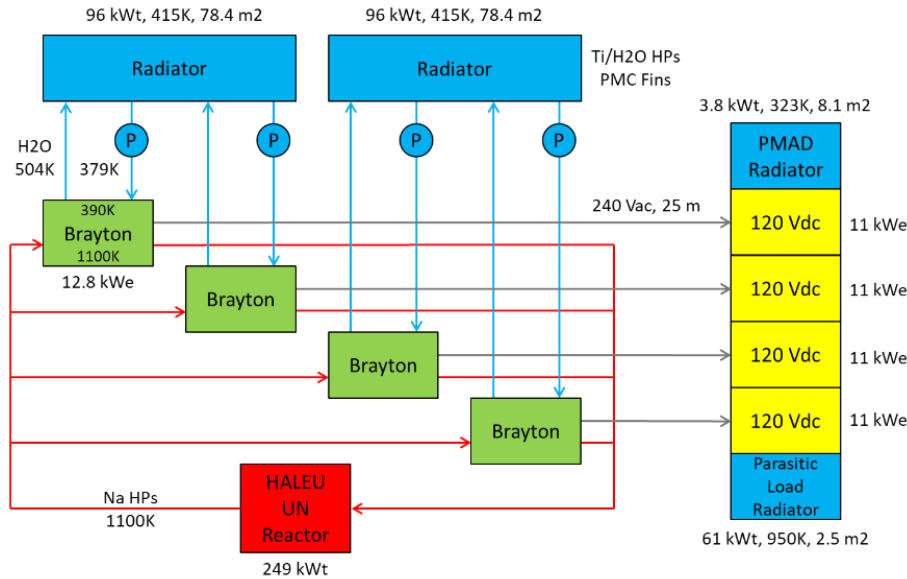


**Fig. 37 - NEP Power System Architecture**

The baseline 44 kWe HALEU heat pipe Brayton power system weighs about 4700 kg (107 kg/kWe) without margin, and the NEP power system mass summary is presented in Fig. 38. The system is split in the master equipment list (MEL) between the EPS and Thermal Control System (TCS) as shown in Fig. 39 and Fig. 40. The baseline system requires a reactor output thermal power of 249 kWt and a total main radiator area of about 160 m<sup>2</sup>.

Heat rejection for the nuclear power system is accomplished with two separate radiator assemblies, each capable of rejecting 96 kWt at an average temperature of about 415K. Each assembly includes a 22 m<sup>2</sup> forward panel and a 58 m<sup>2</sup> aft panel, separated by the boom hinge joint. The radiator panels utilize embedded Ti/H<sub>2</sub>O heat pipes in a polymer matrix composite (PMC) fin structure with two-sided heat rejection to an assumed 4K thermal sink. Each radiator assembly includes two independent, pumped H<sub>2</sub>O fluid loops that transfer the waste heat from two Brayton converters to the radiator heat pipes. In the event of a Brayton converter or radiator fluid loop failure, the remaining Brayton unit connected to that radiator assembly can produce greater than 25% power since the radiator is sized for two converters. The large number of heat pipes in the radiator panels assures that individual heat pipe failures (e.g., due to micrometeoroid impact) would have negligible impact on system performance.

Several internal trades were performed to evaluate the nuclear power system mass sensitivity. The system mass changes for varying output power resulted in a system mass of about 2250 kg at 10 kWe (225 kg/kWe) and 7600 kg at 100 kWe (76 kg/kWe). Substituting an HEU reactor for the HALEU version resulted in a total mass reduction of about 600 kg at 44 kWe. Substituting Stirling power conversion resulted in a total mass reduction of about 400 kg at 44 kWe. An advanced technology version of the 44 kWe HALEU Brayton system was also evaluated. Increasing the reactor outlet temperature from 1100K to 1300K and the compressor inlet temperature from 390K to 420K resulted in a total mass reduction of about 600 kg, which is the same mass savings estimated for the HEU reactor case. The advanced technology version would require significant upgrades to the reactor moderator, primary heat transport, Brayton conversion, and heat rejection designs that would add risk and uncertainty to the system development. For example, the higher compressor inlet temperature would require a change in the radiator heat transport loop fluid from water to NaK.

| CBE Mass Summary (kg) |  |    | Unit | Total |
|-----------------------|--|----|------|-------|
| Brayton Power System  |  |    |      | 4694  |
| Heat Pipe Reactor     |  |    |      | 1264  |
| Core                  |  | 1X | 237  | 237   |
| Reflector             |  | 1X | 614  | 614   |
| PHTS                  |  | 1X | 134  | 134   |
| I&C                   |  | 1X | 167  | 167   |
| Structure             |  | 1X | 112  | 112   |
| Safety Rod            |  | 1X | 0    | 0     |
| Shield                |  |    |      | 817   |
| LIH/W                 |  | 1X | 681  | 681   |
| Structure             |  | 1X | 136  | 136   |
| PCS                   |  |    |      | 1240  |
| HSHX                  |  | 4X | 0    | 0     |
| TAC                   |  | 4X | 72   | 289   |
| Recup                 |  | 4X | 165  | 659   |
| Cooler                |  | 4X | 40   | 161   |
| Structure             |  | 4X | 33   | 132   |
| HRS                   |  |    |      | 706   |
| Fluid Loop            |  | 4X | 35   | 141   |
| Radiator              |  | 2X | 283  | 565   |
| Deployment            |  | 2X | 0    | 0     |
| PMAD                  |  |    |      | 666   |
| Electronics           |  | 4X | 121  | 484   |
| Cabling               |  | 4X | 8    | 31    |
| Thermal               |  | 4X | 34   | 134   |
| Aux Pwr               |  | 1X | 17   | 17    |

Fig. 38 - NEP Reactor-related power system mass summary

| Description                                | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |            |              |               |
|                                            |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Electrical Power Subsystem</b>          |     |           | <b>4153.5</b> | <b>21%</b> | <b>861.9</b> | <b>5015.4</b> |
| <b>Power Generation</b>                    |     |           | <b>3861.0</b> | <b>20%</b> | <b>772.2</b> | <b>4633.2</b> |
| Solar Array                                | 1   | 24.0      | 24.0          | 20%        | 4.8          | 28.8          |
| Reactor                                    | 1   | 1264.0    | 1264.0        | 20%        | 252.8        | 1516.8        |
| Shield                                     | 1   | 817.0     | 817.0         | 20%        | 163.4        | 980.4         |
| PCS                                        | 4   | 310.0     | 1240.0        | 20%        | 248.0        | 1488.0        |
| PMAD Electronics                           | 4   | 121.0     | 484.0         | 20%        | 96.8         | 580.8         |
| PMAD Cabling                               | 4   | 8.0       | 32.0          | 20%        | 6.4          | 38.4          |
| <b>Power Management &amp; Distribution</b> |     |           | <b>195.0</b>  | <b>36%</b> | <b>70.2</b>  | <b>265.2</b>  |
| Power Electronics Box                      | 2   | 78.0      | 156.0         | 20%        | 31.2         | 187.2         |
| Hamess                                     | 1   | 39.0      | 39.0          | 100%       | 39.0         | 78.0          |
| <b>Energy Storage</b>                      |     |           | <b>97.5</b>   | <b>20%</b> | <b>19.5</b>  | <b>117.0</b>  |
| Lithium-Ion Battery                        | 1   | 97.5      | 97.5          | 20%        | 19.5         | 117.0         |

Fig. 39 - Nuclear Portion of EPS MEL

| Description                                | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |            |              |               |
|                                            |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Thermal Control (Non-Propellant)</b>    |     |           | <b>1064.3</b> | <b>18%</b> | <b>191.6</b> | <b>1255.8</b> |
| <b>Active Thermal Control</b>              |     |           | <b>3.9</b>    | <b>18%</b> | <b>0.7</b>   | <b>4.6</b>    |
| Heaters                                    | 21  | 0.1       | 1.1           | 18%        | 0.2          | 1.2           |
| Thermocouples                              | 21  | 0.0       | 0.2           | 18%        | 0.0          | 0.2           |
| Data Acquisition                           | 2   | 0.3       | 0.5           | 18%        | 0.1          | 0.6           |
| Switches                                   | 21  | 0.1       | 2.1           | 18%        | 0.4          | 2.5           |
| <b>Passive Thermal Control</b>             |     |           | <b>1028.8</b> | <b>18%</b> | <b>185.2</b> | <b>1214.0</b> |
| MLI Insulation                             | 1   | 79.0      | 79.0          | 18%        | 14.2         | 93.2          |
| Electronics Cold Plates                    | 10  | 0.1       | 1.4           | 18%        | 0.2          | 1.6           |
| Electronics Heat pipes                     | 20  | 0.9       | 18.7          | 18%        | 3.4          | 22.1          |
| PPU cold Plates                            | 10  | 1.4       | 14.4          | 18%        | 2.6          | 17.0          |
| PPU Heat Pipes                             | 10  | 1.2       | 11.9          | 18%        | 2.1          | 14.0          |
| thermal paint                              | 1   | 0.5       | 0.5           | 18%        | 0.1          | 0.6           |
| PIR Shunt Radiator                         | 1   | 61.0      | 61.0          | 18%        | 11.0         | 72.0          |
| Reactor PMAD Radiator                      | 4   | 34.0      | 136.0         | 18%        | 24.5         | 160.5         |
| Reactor Heat Transfer Sys                  | 4   | 35.0      | 140.0         | 18%        | 25.2         | 165.2         |
| Reactor Radiator                           | 2   | 283.0     | 566.0         | 18%        | 101.9        | 667.9         |
| <b>Semi-Passive Thermal Control</b>        |     |           | <b>31.6</b>   | <b>18%</b> | <b>5.7</b>   | <b>37.3</b>   |
| Electronics & PPU Radiator                 | 1   | 31.6      | 31.6          | 18%        | 5.7          | 37.3          |

**Fig. 40 - Nuclear-related portion of Thermal Control MEL**

### 5.5 Electrical Power System

The EPS for the SERENE is comprised of a solar array for power generation during launch and commissioning, a Li-ion energy storage system to provide start-up power for the nuclear reactor, and a power management and distribution system (PMAD) capable of delivering power to/from a 120VDC bus to the various vehicle loads, including routing power from the nuclear subsystem to the ten electric propulsion (EP) thrusters. During commissioning, the solar array must be capable of providing 1.4 kW<sub>e</sub> to the SERENE power bus for the various systems on-board as shown in the EPS PEL (Table 12) plus be able to recharge the energy storage within 72 hours in the event of an aborted reactor start-up. To achieve this, a 5.2 m<sup>2</sup>, body-mounted panel containing 660 Spectrolab XTE photovoltaic cells was designed. The array consists of 11 standard power modules (SPMs) arranged in parallel, each with 60 cells in series to provide a nominal 120 VDC bus voltage with a power output of ~1.5 kW [26]. Energy storage is provided via 19.3kWh Li-ion battery comprised of an array of 34s-45p LG 18650 MJ1 cells [27]. This battery can provide 1.9kW<sub>e</sub> to the reactor for 7.5 hours and still maintain the 80% maximum depth of discharge (DoD) requirement plus contain two spare strings for internal redundancy.

The SERENE PMAD system was designed using the Advanced Modular Power System (AMPS) model and components. The AMPS box contains all power “cards” necessary to distribute, monitor, and isolate bus loads [28]. For redundancy and fault tolerance, a second AMPS box is included in the PMAD as a “hot spare”, meaning it is always connected and, thus, adds parasitic losses. 44kW 120VDC power is accepted from the nuclear subsystem, which is then isolated and distributed to the EP thrusters (during thrusting), spacecraft loads, and reactor electronics. Because these are dedicated lines, the PMAD system must be sized to handle the 42kW plus efficiency losses to the thrusters (Power Mode 2) and be capable of distributing 3kW<sub>e</sub> to the various sub-system electronics during Communications (Power Mode 3) independently. The PMAD system is also designed to regulate power from the solar array and to/from the battery to maintain bus voltage. For solar array regulation, a 28VDC Terma, commercial, off-the-shelf sequential shunt unit (SSU) was scaled to match the 120VDC bus requirement and was assumed to take up 1 card space in the AMPS PMAD box [29]. Likewise, a battery charge/discharge unit (BCDU) is incorporated into the PMAD, using a bi-directional converter module (BCDM) to isolate and regulate power to/from the bus and

battery. Overall, from reactor to loads, the system efficiency was calculated to be ~90% at peak power after all EPS overhead was taken into consideration. The EPS architecture is shown below in Fig. 41. All components of the EPS are high maturity (TRL 9), except for the AMPS PMAD, which is at TRL5. Total EPS mass and breakdown is shown in Fig. 42.

**Table 12 - SERENE Powered Equipment List (PEL)**

| Description                                | Power Mode 1             | Power Mode 2       | Power Mode 3                                    |
|--------------------------------------------|--------------------------|--------------------|-------------------------------------------------|
| Case 1_Enceladus Sample Return CD-2025-225 | Launch and Commissioning | Transit- Thrusting | Communications/Science in Orbit at Saturn Moons |
|                                            | 24 hours                 | x hours            | x hours                                         |
|                                            | (W)                      | (W)                | (W)                                             |
| <b>Enceladus Sample Return</b>             | <b>1100.4</b>            | <b>39211.7</b>     | <b>2269.0</b>                                   |
| <b>NEP Vehicle</b>                         | <b>1100.4</b>            | <b>39211.7</b>     | <b>2269.0</b>                                   |
| <b>Payload</b>                             | <b>81.0</b>              | <b>81.0</b>        | <b>354.0</b>                                    |
| Attitude Determination and Control         | 169.4                    | 169.4              | 352.4                                           |
| Command & Data Handling                    | 47.2                     | 87.8               | 117.8                                           |
| Communications and Tracking                | 0.0                      | 0.0                | 500.0                                           |
| Electrical Power Subsystem                 | 585.0                    | 751.0              | 639.0                                           |
| Thermal Control (Non-Propellant)           | 6.8                      | 6.8                | 6.8                                             |
| Propulsion (Chemical Hardware)             | 88.0                     | 176.0              | 176.0                                           |
| Propellant (Chemical)                      | 0.0                      | 0.0                | 0.0                                             |
| Propulsion (EP Hardware)                   | 123.0                    | 37939.7            | 123.0                                           |
|                                            |                          |                    |                                                 |
| Bus Power, System Total                    | 977                      | 1272               | 2146                                            |
| 30% growth                                 | 293                      | 382                | 644                                             |
| Total Bus Power Requirement                | 1271                     | 1654               | 2790                                            |
| EP System power, System total              | 123                      | 37940              | 123                                             |
| 5% growth                                  | 6                        | 1897               | 6                                               |
| EP System total Requirement                | 129                      | 39837              | 129                                             |
| Total System power with growth             | 1400                     | 41490              | 2919                                            |

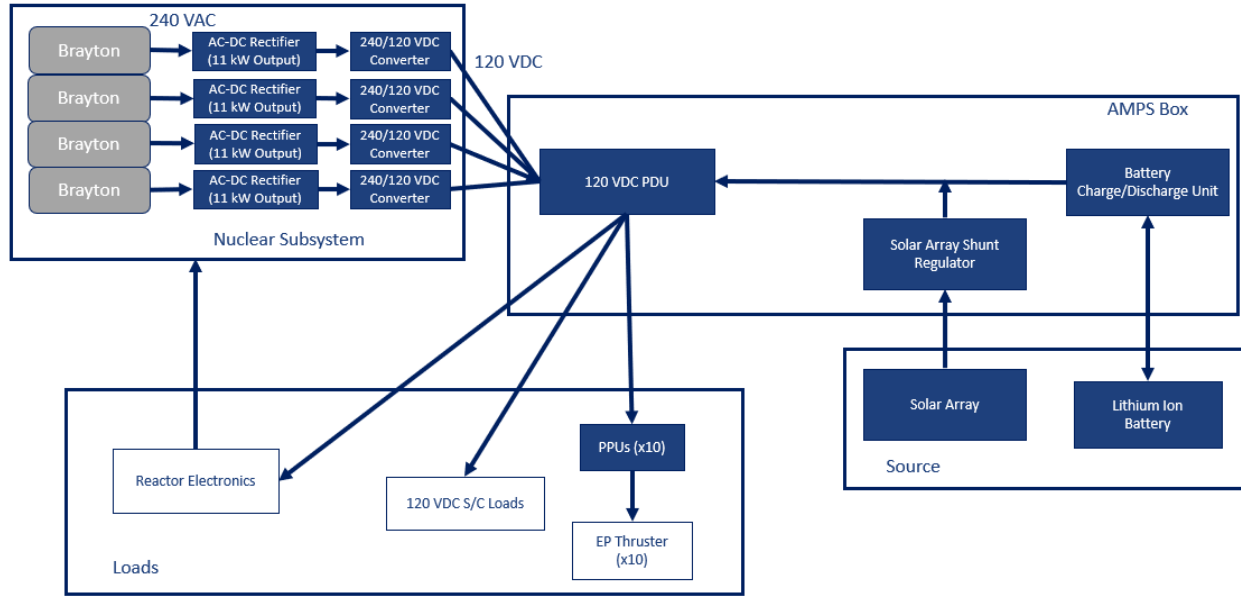


**Fig. 41 - Electrical Power System Architecture**

| Description                                       | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|---------------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| <b>Case 1_Enceladus Sample Return CD-2025-225</b> |     |           |               |            |              |               |
|                                                   |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Electrical Power Subsystem</b>                 |     |           | <b>4170.5</b> | <b>21%</b> | <b>865.3</b> | <b>5035.8</b> |
| <b>Power Generation</b>                           |     |           | <b>3861.0</b> | <b>20%</b> | <b>772.2</b> | <b>4633.2</b> |
| Solar Array                                       | 1   | 24.0      | 24.0          | 20%        | 4.8          | 28.8          |
| Reactor                                           | 1   | 1264.0    | 1264.0        | 20%        | 252.8        | 1516.8        |
| Shield                                            | 1   | 817.0     | 817.0         | 20%        | 163.4        | 980.4         |
| PCS                                               | 4   | 310.0     | 1240.0        | 20%        | 248.0        | 1488.0        |
| PMAD Electronics                                  | 4   | 121.0     | 484.0         | 20%        | 96.8         | 580.8         |
| PMAD Cabling                                      | 4   | 8.0       | 32.0          | 20%        | 6.4          | 38.4          |
| <b>Power Management &amp; Distribution</b>        |     |           | <b>212.0</b>  | <b>35%</b> | <b>73.6</b>  | <b>285.6</b>  |
| Power Electronics Box                             | 2   | 78.0      | 156.0         | 20%        | 31.2         | 187.2         |
| Harass                                            | 1   | 39.0      | 39.0          | 100%       | 39.0         | 78.0          |
| Aux Power                                         | 1   | 17.0      | 17.0          | 20%        | 3.4          | 20.4          |
| <b>Energy Storage</b>                             |     |           | <b>97.5</b>   | <b>20%</b> | <b>19.5</b>  | <b>117.0</b>  |
| Lithium-Ion Battery                               | 1   | 97.5      | 97.5          | 20%        | 19.5         | 117.0         |

**Fig. 42 - EPS MEL**

## 5.6 Propulsion

The propulsion system for SERENE consists of a primary EP system and a monopropellant RCS. The EP system consists of ten NEXT-C based thrusters operating in an 8+2 configuration with no cross-strapping among the thruster strings. They are mounted on two panels, with each thruster being mounted to the panels via its own individual commercial off-the-shelf (COTS) gimbal assembly. The EP thrusters utilize xenon propellant that is stored supercritically in a single 4,898 l composite overwrapped pressure vessel (COPV) that operates at a mean pressure of 119.6 bar (1,740 psia) with the current propellant load. It is constructed of a titanium alloy liner with a T-1000 overwrap, has 2 m diameter, elliptical endcaps, and mounts via a tabbed shirt. The xenon is routed from the tank to the thrusters via a nominal single fault tolerant feed system with dual flex lines at the panel gimbals which can be isolated via latch valves in case of damage to the lines. The xenon is then conditioned via individual flow control

modules mounted near each thruster. Each thruster is powered via a dedicated PPU that receives power from the 120 VDC spacecraft bus.

The NEXT-C thrusters have been extensively tested at NASA GRC for 50,000 hours and a total of 900 kg of xenon throughput. Initial lifetime estimates based on test data indicate that twice the tested throughput is possible [30]. For this design, it is assumed that these thrusters are an upgraded version utilizing enhancements that are currently under development. The primary enhancement is the use of flat carbon-based optics that have superior erosion resistance, low beam divergence (less than five degrees), and can minimize spacecraft plasma interactions. Additional enhancements to other thruster subcomponents are also in development to enable operation at high specific impulse ( $> 5,000$  s). The NEXT-C thruster is shown in Fig. 43 [31].

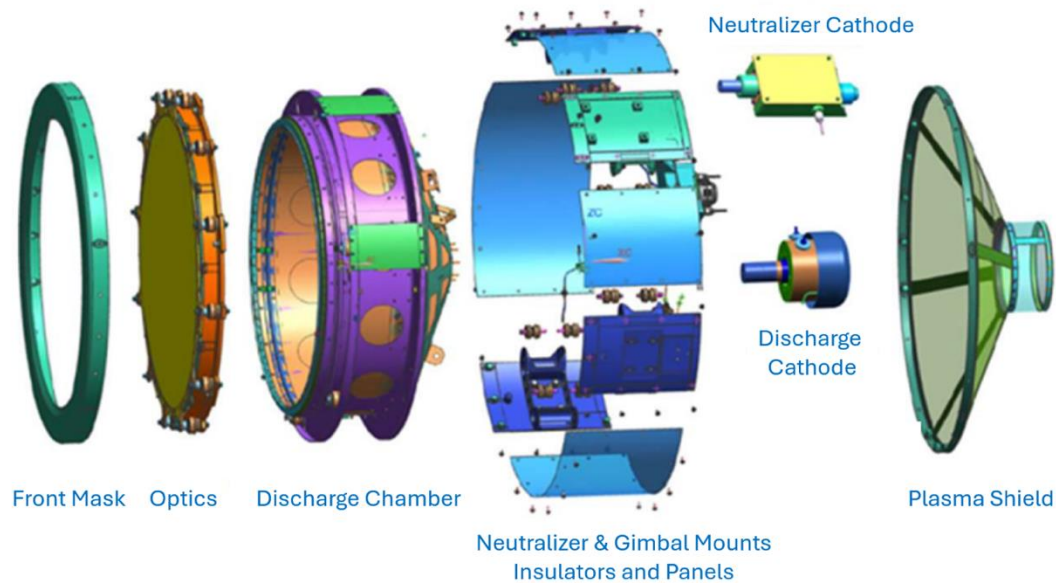


**Fig. 43 - NEXT-C Thruster Assembly**

The xenon flow control modules are based on COTS units that have integral 10-micron filters, triple redundant seals to prevent internal leakage, independent controls for anode and cathode flows, and an integrated closed loop flow controller. These modules are rated to inlet pressures up to 206.8 bar (3,000 psia), thus removing the need for a dedicated pressure regulator. There is one module per thruster string, and it receives data and power from that string's PPU. The PPU and thruster gimbal communicate with the main spacecraft computer, thus allowing it to coordinate both thrust level and thrust vector orientation across the EP thruster strings. This arrangement is shown in Fig. 44.

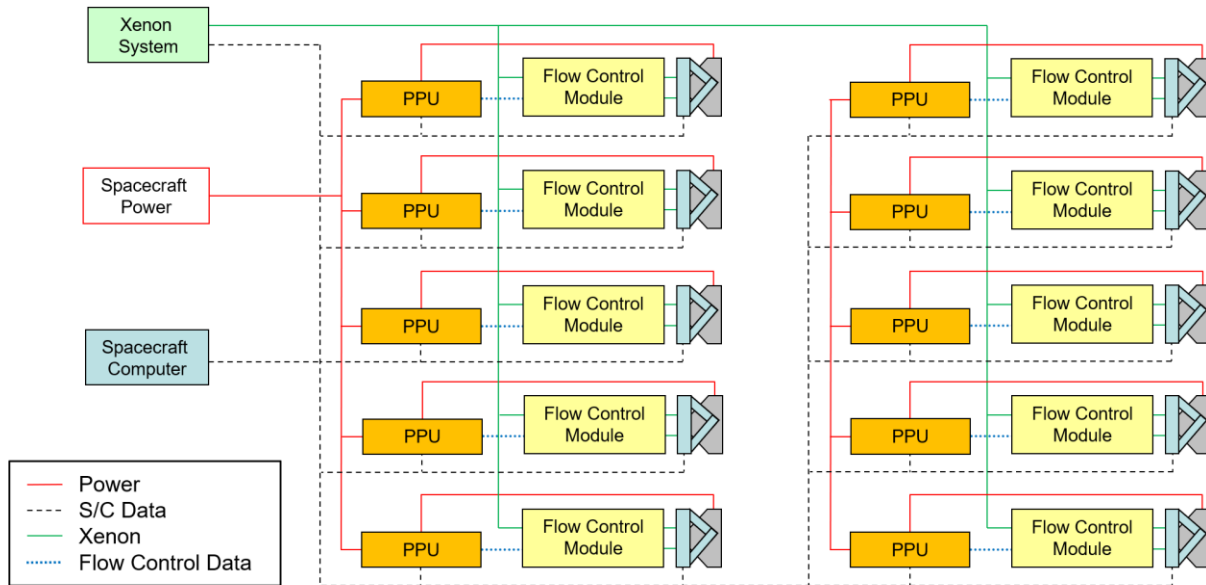


**Fig. 44 - EP System Diagram**

The RCS is a monopropellant hydrazine blow-down system consisting of sixteen COTS thrusters that develop a nominal 22N of thrust and 230 s of Isp. These thrusters are housed in four pods containing four thrusters each that are located at the aft of the spacecraft. The hydrazine propellant is stored in six helium pressurized COTS membrane tanks located in the aft portion of the bus. Four of the hydrazine tanks are 85.6 cm long and have a 58.3 cm diameter, while the remaining two tanks are smaller 56.2 cm diameter spheres. These tanks operate at a nominal 25.9 bar (375 psia), are sized to operate at a 3:1 blown ratio, and are at 82% capacity with the current propellant load. The propellant flows to the thrusters via a nominal single fault tolerant feed system that includes line heaters, isolation valves, and a nominal instrumentation suite.

| Description                                | QTY | Unit Mass | Basic Mass   | Growth    | Growth     | Total Mass   |
|--------------------------------------------|-----|-----------|--------------|-----------|------------|--------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |              |           |            |              |
|                                            |     | (kg)      | (kg)         | (% )      | (kg)       | (kg)         |
| <b>Propulsion (Chemical Hardware)</b>      |     |           | <b>115.7</b> | <b>5%</b> | <b>5.5</b> | <b>121.3</b> |
| <b>Primary Chemical System Hardware</b>    |     |           | <b>115.7</b> | <b>5%</b> | <b>5.5</b> | <b>121.3</b> |
| <i>Reaction Control System Hardware</i>    |     |           | 115.7        | 5%        | 5.5        | 121.3        |
| MR-106 Thrusters                           | 16  | 0.6       | 9.6          | 3%        | 0.3        | 9.9          |
| RCS Thruster Subassembly                   | 4   | 2.0       | 8.0          | 15%       | 1.2        | 9.2          |
| RCS Feed System                            | 1   | 9.2       | 9.2          | 15%       | 1.4        | 10.6         |
| Main Hydrazine Tanks                       | 4   | 19.1      | 76.2         | 3%        | 2.3        | 78.5         |
| Secondary Hydrazine Tanks                  | 2   | 6.4       | 12.7         | 3%        | 0.4        | 13.1         |

**Fig. 45 - Chemical Propulsion Hardware**

| Description                                | QTY | Unit Mass | Basic Mass   | Growth    | Growth     | Total Mass   |
|--------------------------------------------|-----|-----------|--------------|-----------|------------|--------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |              |           |            |              |
|                                            |     | (kg)      | (kg)         | (% )      | (kg)       | (kg)         |
| <b>Propellant (Chemical)</b>               |     |           | <b>473.1</b> | <b>0%</b> | <b>0.0</b> | <b>473.1</b> |
| <b>RCS Propellant</b>                      |     |           | <b>473.1</b> | <b>0%</b> | <b>0.0</b> | <b>473.1</b> |
| <i>Fuel</i>                                |     |           | 469.6        | 0%        | 0.0        | 469.6        |
| Fuel Usable                                | 1   | 412.5     | 412.5        | 0%        | 0.0        | 412.5        |
| Fuel Margin                                | 1   | 41.2      | 41.2         | 0%        | 0.0        | 41.2         |
| Fuel Residuals (Unused)                    | 1   | 15.9      | 15.9         | 0%        | 0.0        | 15.9         |
| <i>RCS Pressurant</i>                      |     |           | 3.5          | 0%        | 0.0        | 3.5          |
| Misc# 06.1.8.b.c.a                         | 1   | 3.5       | 3.5          | 0%        | 0.0        | 3.5          |

Fig. 46 - Chemical Propellant MEL

| Description                                | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |            |              |               |
|                                            |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Propulsion (EP Hardware)</b>            |     |           | <b>1059.2</b> | <b>12%</b> | <b>131.4</b> | <b>1190.6</b> |
| <b>Primary EP System Hardware</b>          |     |           | <b>674.2</b>  | <b>12%</b> | <b>78.9</b>  | <b>753.1</b>  |
| <i>EP System Hardware</i>                  |     |           | 674.2         | 12%        | 78.9         | 753.1         |
| Next-C Thrusters                           | 10  | 14.0      | 140.0         | 10%        | 14.0         | 154.0         |
| Thruster Gimbals                           | 10  | 6.0       | 60.0          | 12%        | 7.2          | 67.2          |
| Xenon Tank                                 | 1   | 449.2     | 449.2         | 12%        | 53.9         | 503.1         |
| Flow Control Modules                       | 10  | 1.4       | 14.0          | 15%        | 2.1          | 16.1          |
| Xenon Distribution System                  | 1   | 11.0      | 11.0          | 15%        | 1.7          | 12.7          |
| <b>Power Processing Unit (PPU)</b>         |     |           | <b>385.0</b>  | <b>14%</b> | <b>52.5</b>  | <b>437.5</b>  |
| Integrated PPU/DCIU                        | 10  | 35.0      | 350.0         | 10%        | 35.0         | 385.0         |
| Cabling (PPU to thrusters)                 | 10  | 3.5       | 35.0          | 50%        | 17.5         | 52.5          |

Fig. 47 - EP Hardware MEL

| Description                                    | QTY | Unit Mass | Basic Mass    | Growth    | Growth     | Total Mass    |
|------------------------------------------------|-----|-----------|---------------|-----------|------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225     |     |           |               |           |            |               |
|                                                |     | (kg)      | (kg)          | (% )      | (kg)       | (kg)          |
| <b>Propellant (EP)</b>                         |     |           | <b>9088.5</b> | <b>0%</b> | <b>0.0</b> | <b>9088.5</b> |
| <b>Main Propellant (EP)</b>                    |     |           | <b>9088.5</b> | <b>0%</b> | <b>0.0</b> | <b>9088.5</b> |
| Primary EP Propellant Used                     | 1   | 8492.6    | 8492.6        | 0%        | 0.0        | 8492.6        |
| Primary EP Propellant Residuals (Unused)       | 1   | 86.3      | 86.3          | 0%        | 0.0        | 86.3          |
| Primary EP Propellant Performance Margin (Unus | 1   | 509.6     | 509.6         | 0%        | 0.0        | 509.6         |

Fig. 48 - EP Propellant MEL

### 5.7 Attitude Determination and Control (AD&C)

The Attitude Determination and Control (AD&C) subsystem design philosophy is centered around satisfying the notional pointing accuracy requirement of 0.28° 3-σ while maintaining a pointing knowledge of 0.028° 3-σ (roughly 100 arc-sec). This pointing accuracy enables the radar mapping of Saturn's moons and coarse pointing of the communication antenna.

The pointing knowledge is accomplished using two sets of “blended mode” star trackers with specifications based on Sodern Hydra Star Trackers [32]. A blended mode configuration allows measurements from two star trackers which are pointing in independent directions to be combined, in effect reducing the error about a single star tracker boresight axis to levels approaching the off-boresight values of ~5 arc-sec. Redundancy is incorporated in the form of a second set of star trackers.

The spacecraft has two reaction wheels in the two largest moments of inertia axis directions, one in the smallest, and one spare with partial control. At periapsis when in orbit of Enceladus, max gravity gradient torques exceed the control authority of the reaction wheels. Maintaining a gravity gradient stable attitude with the reactor towards zenith will be required unless RCS is used for attitude control at this time. The radar array is placed on the vehicle such that a stable gravity gradient points the array towards the moon’s surface.

Several items of forward work are recommended including assessing the need for fast slews during the moon tour and further refining RCS usage throughout the mission. Given the mission length, the degraded pointing performance due to a failed reaction wheel should be assessed.

The equipment list for the AD&C system is shown in Fig. 49.

| Description                                | QTY | Unit Mass | Basic Mass  | Growth     | Growth      | Total Mass  |
|--------------------------------------------|-----|-----------|-------------|------------|-------------|-------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |             |            |             |             |
|                                            |     | (kg)      | (kg)        | (% )       | (kg)        | (kg)        |
| <b>Attitude Determination and Control</b>  |     |           | <b>82.8</b> | <b>16%</b> | <b>13.5</b> | <b>96.3</b> |
| <b>Guidance, Navigation, &amp; Control</b> |     |           | <b>82.8</b> | <b>16%</b> | <b>13.5</b> | <b>96.3</b> |
| IMU                                        | 2   | 0.8       | 1.6         | 3%         | 0.0         | 1.6         |
| Star Tracker Optical Head                  | 4   | 1.4       | 5.6         | 5%         | 0.3         | 5.9         |
| Star Tracker Electronics Unit              | 2   | 1.8       | 3.6         | 5%         | 0.2         | 3.8         |
| RW                                         | 6   | 12.0      | 72.0        | 18%        | 13.0        | 85.0        |

**Fig. 49 - AD&C MEL**

### **5.8 Command and Data Handling (C&DH)**

The Command & Data Handling (C&DH) subsystem provides command, control, health management, and data storage for the spacecraft. All components in the design are assumed to be radiation tolerant, rated at 100 kRad total ionizing dose (TID), and single fault tolerant. The C&DH system is separated into two enclosures. The first is for data collection, processing, and command, and includes two single board computers (SBCs). One SBC controls science operations at Saturn by communicating with the science instruments, collecting data, and storing data. The other SBC assumes command and monitoring of spacecraft operations, which includes communications with other subsystems and monitoring of health and status. Digital and analog data is transmitted and received through digital and analog I/O boards. There is also a 1-TeraByte storage unit in this enclosure for storing mission data. Since the maximum data estimate is around 3.1 Terabits (0.39 TeraBytes), there is margin to add additional data. The second enclosure is for instrumentation drivers and control. It includes an SBC to control and monitor the gimbal drivers and controllers for the RCS thrusters, latch valves, and heat valves. Both enclosure masses incorporate DC-DC converters, electromagnetic interference shielding, and any necessary filtering. They both house 3U cards and include a cPCI backplane for in-unit communication. The data harness mass estimate is derived from a Monte-Carlo simulation of TEFZEL wire density and a length range for each component and interface.

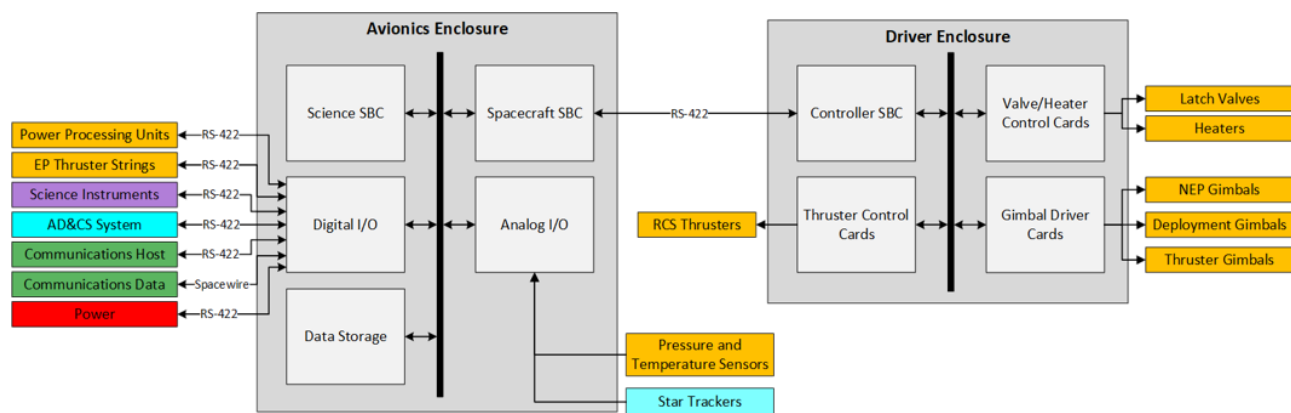


**Fig. 50 - Command and Data Handling System Block Diagram**

| Description                                | QTY | Unit Mass | Basic Mass  | Growth      | Growth      | Total Mass   |
|--------------------------------------------|-----|-----------|-------------|-------------|-------------|--------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |             |             |             |              |
|                                            |     | (kg)      | (kg)        | (% )        | (kg)        | (kg)         |
| <b>Command &amp; Data Handling</b>         |     |           | <b>84.2</b> | <b>76%</b>  | <b>64.1</b> | <b>148.3</b> |
| <b>C&amp;DH Hardware</b>                   |     |           | <b>25.2</b> | <b>20%</b>  | <b>5.1</b>  | <b>30.3</b>  |
| Spacecraft SBC                             | 2   | 0.5       | 1.0         | 5%          | 0.1         | 1.1          |
| Science SBC                                | 2   | 0.5       | 1.0         | 5%          | 0.1         | 1.1          |
| Controller SBC                             | 2   | 0.5       | 1.0         | 5%          | 0.1         | 1.1          |
| DMOAB                                      | 8   | 0.4       | 3.2         | 5%          | 0.2         | 3.4          |
| AMOAB                                      | 2   | 0.4       | 0.8         | 5%          | 0.0         | 0.8          |
| Valve Control Cards                        | 2   | 0.4       | 0.8         | 20%         | 0.2         | 1.0          |
| Thruster Control Cards                     | 4   | 0.4       | 1.6         | 20%         | 0.3         | 1.9          |
| Gimbal Driver Cards                        | 6   | 0.4       | 2.4         | 20%         | 0.5         | 2.9          |
| 1TB Memory Card                            | 2   | 0.6       | 1.2         | 10%         | 0.1         | 1.3          |
| C&DH Enclosure                             | 1   | 6.4       | 6.4         | 30%         | 1.9         | 8.3          |
| Driver Enclosure                           | 1   | 5.8       | 5.8         | 30%         | 1.7         | 7.5          |
| <b>Instrumentation &amp; Wiring</b>        |     |           | <b>59.0</b> | <b>100%</b> | <b>59.0</b> | <b>118.0</b> |
| Wire Harness                               | 1   | 59.0      | 59.0        | 100%        | 59.0        | 118.0        |

**Fig. 51 - Command and Data Handling MEL**

## 5.9 Communications

The communications system of the SERENE was designed to maximize science data return and operational flexibility across the mission's complex architecture. All communication links leverage 34 m DSN ground network services.

The primary communication modality consists of a 2.1-m gimbled Ka band high gain antenna co-bore sighted with an X-band medium gain antenna, both deployable and mounted on a two-axis gimbal mechanism. This configuration enables Earth pointing while operating on Enceladus' surface and supports 200-500 kbps Ka-band science data downlinks at 11 AU distance. Redundant solid-state amplifiers and software defined radios provide single fault tolerance wherever practical.

An array of eight X-band low gain patch antennas positioned strategically around the spacecraft provide omnidirectional coverage for command uplink capability even when the high gain antenna pointing solution to Earth is unknown. These antennas maintain 20 bps communication rates at 11 AU for emergency telemetry and control functions under any spacecraft attitude.

A UHF-band relay system supports communication between surface lander and the orbiting spacecraft, providing 4 Mbps data rates at 40 km altitude and enabling approximately 14 Gb of science data relay per orbit. This can be returned to Earth in a single 8-hour pass using the Ka-band link to DSN.

The communications system design addressed the unique challenges of deep space operations at Saturn while supporting both orbital and surface mission phases. Key design drivers included maintaining reliable Earth communication at extreme distances while providing flexible relay capabilities for surface assets.

The MEL for the SERENE communications system is shown in Fig. 53 and the MEL for assumed modifications to the existing Orbilander communications system is shown in Fig. 54.

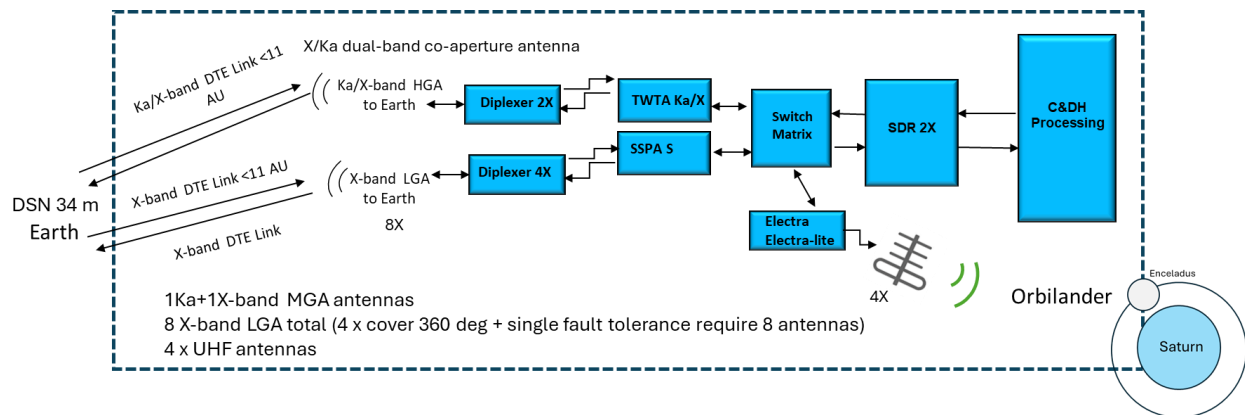


**Fig. 52 - Notional Communications Block Diagram Ka/X-band System**

| Description                                | QTY | Unit Mass | Basic Mass  | Growth     | Growth      | Total Mass  |
|--------------------------------------------|-----|-----------|-------------|------------|-------------|-------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |             |            |             |             |
|                                            |     | (kg)      | (kg)        | (% )       | (kg)        | (kg)        |
| <b>Communications and Tracking</b>         |     |           | <b>48.9</b> | <b>33%</b> | <b>16.1</b> | <b>65.0</b> |
| <b>Ka Band System</b>                      |     |           | <b>23.5</b> | <b>11%</b> | <b>2.5</b>  | <b>26.0</b> |
| 2m HGA Ka-band Dish                        | 1   | 10.0      | 10.0        | 10%        | 1.0         | 11.0        |
| dual-axis gimbal                           | 1   | 4.4       | 4.4         | 10%        | 0.4         | 4.8         |
| Ka-band TWTA                               | 2   | 2.0       | 4.0         | 10%        | 0.4         | 4.4         |
| Ka-band SDR radio                          | 2   | 2.2       | 4.4         | 10%        | 0.4         | 4.8         |
| diplexer 1 unit/2port                      | 1   | 0.5       | 0.5         | 10%        | 0.1         | 0.6         |
| cables + connectors                        | 1   | 0.2       | 0.2         | 100%       | 0.2         | 0.4         |
| <b>X Band System</b>                       |     |           | <b>6.4</b>  | <b>13%</b> | <b>0.8</b>  | <b>7.2</b>  |
| X Band MGA antennas                        | 1   | 1.0       | 1.0         | 10%        | 0.1         | 1.1         |
| X-Band SSPA                                | 2   | 1.4       | 2.8         | 10%        | 0.3         | 3.1         |
| diplexer                                   | 4   | 0.2       | 0.8         | 10%        | 0.1         | 0.9         |
| cables + connectors                        | 1   | 0.2       | 0.2         | 100%       | 0.2         | 0.4         |
| X Band patch antennas                      | 8   | 0.2       | 1.6         | 10%        | 0.2         | 1.8         |
| <b>UHF System</b>                          |     |           | <b>19.0</b> | <b>67%</b> | <b>12.7</b> | <b>31.7</b> |
| UHF-Band Antenna                           | 2   | 0.6       | 1.2         | 10%        | 0.1         | 1.3         |
| Electra/Electra-Lite                       | 2   | 2.7       | 5.4         | 10%        | 0.5         | 5.9         |
| cables + connectors                        | 2   | 6.0       | 12.0        | 100%       | 12.0        | 24.0        |
| misc. structural                           | 2   | 0.2       | 0.4         | 10%        | 0.0         | 0.4         |

Fig. 53 - Communications MEL

| Description                                | QTY | Unit Mass | Basic Mass  | Growth    | Growth     | Total Mass  |
|--------------------------------------------|-----|-----------|-------------|-----------|------------|-------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |             |           |            |             |
|                                            |     | (kg)      | (kg)        | (% )      | (kg)       | (kg)        |
| <b>Communications and Tracking</b>         |     |           | <b>10.1</b> | <b>9%</b> | <b>0.9</b> | <b>11.0</b> |
| <b>UHF Communications System</b>           |     |           | <b>10.1</b> | <b>9%</b> | <b>0.9</b> | <b>11.0</b> |
| UHF-Band Antenna                           | 4   | 0.1       | 0.6         | 3%        | 0.0        | 0.6         |
| mounting hardware                          | 4   | 0.1       | 0.4         | 20%       | 0.1        | 0.5         |
| UHF-band SSPA                              | 2   | 1.4       | 2.7         | 20%       | 0.5        | 3.3         |
| UHF-band SDR                               | 2   | 3.0       | 6.0         | 3%        | 0.2        | 6.2         |
| misc. structural                           | 1   | 0.2       | 0.2         | 3%        | 0.0        | 0.2         |
| cables + connectors                        | 1   | 0.2       | 0.2         | 20%       | 0.0        | 0.2         |

Fig. 54 - Communications System Modifications for Orbilander Representative System MEL

### 5.10 Thermal Control System (TCS)

The spacecraft utilizes a nuclear reactor for providing power to the on-board electronics and control system as well as the PPU for the electric propulsion engines. The thermal system must maintain the temperature of the electronics and PPUs within their designated operating ranges throughout the mission. The thermal system also provides insulation and heaters to ensure the components don't fall below their minimum operating temperature. The main components that generate waste heat include the electric propulsion power production units (PPUs) as well as the waste heat from the electrical power, data, control and communications equipment for the spacecraft and propulsion system.

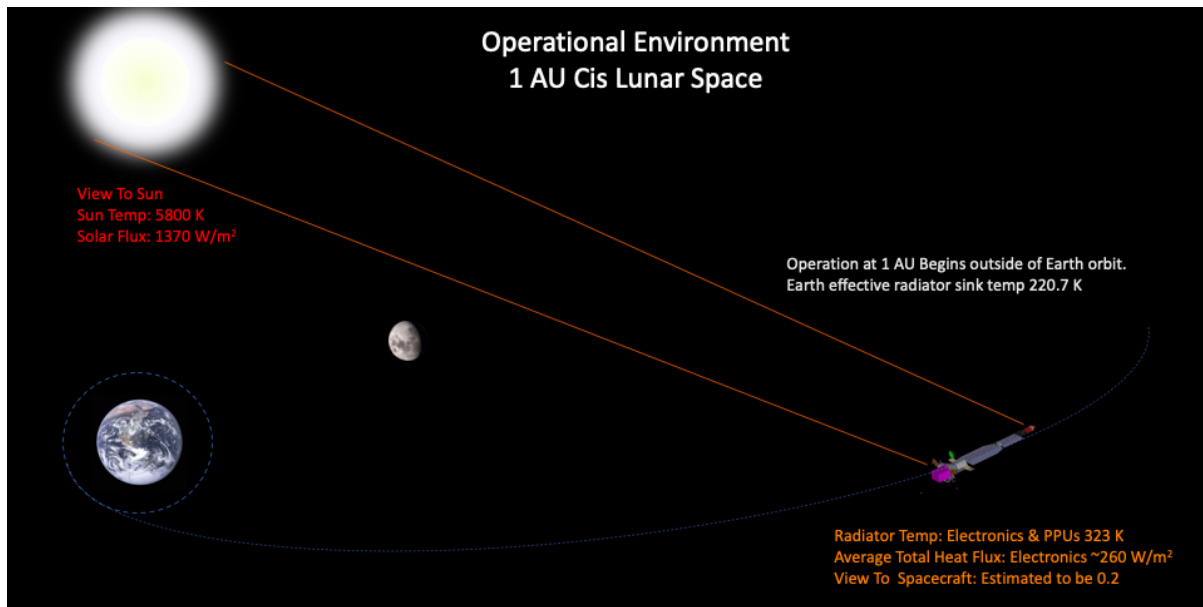


**Fig. 55 - Initial Operating Environment for the Enceladus Mission**

The initial spacecraft operation will take place at 1 AU, as shown in Fig. 55, which represents the worst-case thermal operation. The thermal system will be sized to operate within this environment. Solar Intensity and view angle are used to determine the worst case hot and cold conditions. The worst-case warm conditions will occur at 1 AU in sunlight with all equipment operation, whereas the worst-case cold conditions were also at 1 AU but with minimal equipment operating. The aspects of the thermal control and environment as well as the system components that were addressed or sized in the design and analysis are listed below.

#### **Systems Modeled:**

- Radiator Panels
- Spacecraft Insulation
- Heat pipes and cold plates for moving heat from the electronics and PPUs to the radiator
- Heaters for controlling the spacecraft components temperature
- Temperature Sensors, Controllers, Switches, Data Acquisition
- Heat leak through the insulation & insulation pass-through

Table 13 identifies the specification, assumptions and requirements for the thermal system design and operation.

Multi-Layer insulation (MLI) was used to cover the exterior of the spacecraft. The insulation was analyzed to determine the required number of layers and their corresponding mass and the heat leak out of the spacecraft. A tradeoff was performed between the insulation mass and the required heater power. The insulation model was based on radiation heat transfer analysis of the heat transfer from the spacecraft through the insulation to the surroundings. The specifications for the MLI are given in Table 13.

A deployable accordion-type radiator was used for rejecting the waste heat from the electronics and PPUs. The radiator was sized to remove the waste heat during the worst case hot operational conditions which occur while sunlit at 1 AU. The radiator sizing was based on a power balance analysis of the area needed to reject the identified heat load to space as illustrated in Fig. 56. From the area a series of scaling equations were used to determine the mass of the radiator. A single radiator system was used for both the electronics and PPUs since they are both operating at the same rejection temperature. The radiator specifications are given in Table 13.

Louvers were not used on the radiator since, when the PPUs are not operating, ample power is available to run heaters if needed to maintain temperature. A deployable radiator was used for the combined electronics and PPU heat load. A heat pipe/cold plate system to move the heat from the electronics and other components to the radiator.

**Table 13 - Specifications and Inputs Used for the Thermal Analysis**

| Variable/Component                  |                                               | Value/Description                                                                                                                                                              |
|-------------------------------------|-----------------------------------------------|--------------------------------------------------------------------------------------------------------------------------------------------------------------------------------|
| Spacecraft                          |                                               | Main Body: 3 m Diameter, 3 m Length                                                                                                                                            |
| Waste Heat Load to be Rejected:     |                                               | Electronics Bus & PPUs: 2912 W                                                                                                                                                 |
| Operating Temperature               |                                               | Electronics and PPUs: 323 K maximum temperature                                                                                                                                |
| Insulation Specifications           | Insulation (MLI was used for the electronics) | Spacecraft main body and endcaps: 25 layers of MLI, 1 cm                                                                                                                       |
|                                     | Spacecraft MLI Material                       | Aluminumized Mylar                                                                                                                                                             |
|                                     | Spacecraft MLI Material Aerial Density        | 0.11 kg/m <sup>2</sup>                                                                                                                                                         |
|                                     | Outer Covering                                | 0.05 kg/m <sup>2</sup>                                                                                                                                                         |
|                                     | Inner Covering                                | 0.0063 kg/m <sup>2</sup>                                                                                                                                                       |
|                                     | Spacer                                        | 0.055 kg/m <sup>2</sup>                                                                                                                                                        |
|                                     | Reflective Layer                              | 10 %                                                                                                                                                                           |
|                                     | Attachment & Seals Percentage                 |                                                                                                                                                                                |
|                                     | MLI Layer Spacing                             | 0.2 mm                                                                                                                                                                         |
|                                     | Passthrough Percentage                        | 2.84 %                                                                                                                                                                         |
|                                     | Insulation Heat Loss                          | At 1 AU: 154 W (12 W Insulation, 142 W Passthroughs)<br>At 9.6 AU: 338 W (19 W Insulation, 319 W Passthroughs)                                                                 |
| Environment                         |                                               | Operational Environment: 1 AU-9.6 AU                                                                                                                                           |
| Radiator Specifications             | Radiator & View Factors                       | Max 45° Angle to the Sun (for a 2-sided radiator represents sun normal to one radiator side)<br>View Factor to the Spacecraft or solar array 0.20<br>View Factor to Earth: 0.0 |
|                                     | Radiator Solar Absorptivity                   | 0.14                                                                                                                                                                           |
|                                     | Radiator Emissivity                           | 0.84                                                                                                                                                                           |
|                                     | Radiator Area                                 | 8.4 m <sup>2</sup>                                                                                                                                                             |
| Heat Pipes/Pump Loop Coolant System |                                               | Electronics and PPUs: Heat Pipe Based Coolant System                                                                                                                           |

Variable conductance heat pipes are used to move heat from the electronics and other components to the radiators where it is dissipated to the surroundings. A variable conductance heat pipe is a passive device that adjusts automatically to varying heat load inputs, thereby maintaining a constant operating temperature. The sizing for the heat pipes is shown in Table 14.

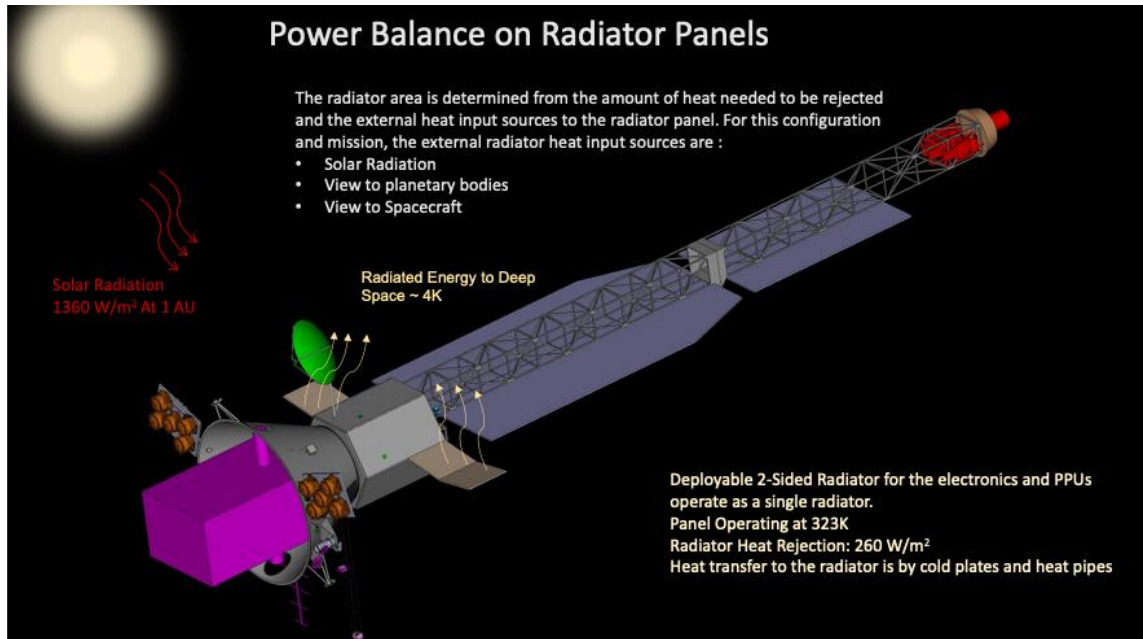


**Fig. 56 - Thermal Power Balance for the Electronics and PPU Radiator Panels**

**Table 14 - Heat Pipe Sizing Specifications**

| Heat Pipe   | Radius  | Effective Length | Heat Transfer Capability | Heat Pipe Specific Mass |
|-------------|---------|------------------|--------------------------|-------------------------|
| Electronics | 1.20 cm | 2.5 m            | 117.8 W/Heat Pipe        | 0.37 kg/m               |
| PPUs        | 1.5 cm  | 2.5 m            | 180.9 W/Heat Pipe        | 0.48 kg/m               |

The electronics and PPUs have two heat pipes run from each cold plate to the radiator. The complete heat load from each cold plate can be carried by one heat pipe. The two heat pipes per cold plate represent a redundant path for heat transfer to the radiator. The radiator panels are interconnected with heat pipes to balance the thermal load between them as well as provide a redundant thermal path in case of a failure.

Waste heat from the internal components as well as electric heaters are used to provide heat to the electronics and other components within the spacecraft. These heaters consist of either strip heaters or flat plate heaters which would be used for the electronics boxes or cold plates. Thermal control is accomplished by a network of thermocouples whose output is used to control the power to the various heaters. A data acquisition and control computer is used to operate the thermal system. Heater power varies with the mission operation from 0.0 W to approximately 338 W if all heat must be supplied by the heaters. The total heater power level will depend on the electronics operating power, and it is expected that the heaters will not need to be used for most of the mission.

The equipment list for the thermal control system (other than that directly related to the reactor) is highlighted below in Fig. 57.

| Description                                | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |            |              |               |
|                                            |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Thermal Control (Non-Propellant)</b>    |     |           | <b>1064.3</b> | <b>18%</b> | <b>191.6</b> | <b>1255.8</b> |
| <b>Active Thermal Control</b>              |     |           | <b>3.9</b>    | <b>18%</b> | <b>0.7</b>   | <b>4.6</b>    |
| Heaters                                    | 21  | 0.1       | 1.1           | 18%        | 0.2          | 1.2           |
| Thermocouples                              | 21  | 0.0       | 0.2           | 18%        | 0.0          | 0.2           |
| Data Acquisition                           | 2   | 0.3       | 0.5           | 18%        | 0.1          | 0.6           |
| Switches                                   | 21  | 0.1       | 2.1           | 18%        | 0.4          | 2.5           |
| <b>Passive Thermal Control</b>             |     |           | <b>1028.8</b> | <b>18%</b> | <b>185.2</b> | <b>1214.0</b> |
| MLI Insulation                             | 1   | 79.0      | 79.0          | 18%        | 14.2         | 93.2          |
| Electronics Cold Plates                    | 10  | 0.1       | 1.4           | 18%        | 0.2          | 1.6           |
| Electronics Heat pipes                     | 20  | 0.9       | 18.7          | 18%        | 3.4          | 22.1          |
| PPU cold Plates                            | 10  | 1.4       | 14.4          | 18%        | 2.6          | 17.0          |
| PPU Heat Pipes                             | 10  | 1.2       | 11.9          | 18%        | 2.1          | 14.0          |
| thermal paint                              | 1   | 0.5       | 0.5           | 18%        | 0.1          | 0.6           |
| PLR Shunt Radiator                         | 1   | 61.0      | 61.0          | 18%        | 11.0         | 72.0          |
| Reactor PMAD Radiator                      | 4   | 34.0      | 136.0         | 18%        | 24.5         | 160.5         |
| Reactor Heat Transfer Sys                  | 4   | 35.0      | 140.0         | 18%        | 25.2         | 165.2         |
| Reactor Radiator                           | 2   | 283.0     | 566.0         | 18%        | 101.9        | 667.9         |
| <b>Semi-Passive Thermal Control</b>        |     |           | <b>31.6</b>   | <b>18%</b> | <b>5.7</b>   | <b>37.3</b>   |
| Electronics & PPU Radiator                 | 1   | 31.6      | 31.6          | 18%        | 5.7          | 37.3          |

Fig. 57 - Thermal Control System MEL

### 5.11 Structures and Mechanisms

The spacecraft is based on a previous Compass design, the planetary defense interceptor [10]. The requirement for the structure is to contain the hardware for command and data handling; communication and tracking; electrical power; thermal control; guidance, navigation, and control; chemical propulsion; electrical propulsion; and nuclear reactor. The structure bears launch loads with a maximum launch vehicle axial acceleration of -6.0 g with a concurrent lateral acceleration of 1.0 g. The launch maximum lateral acceleration is 2.0 g with a concurrent axial acceleration of -3.5 g. The structure needs to provide minimum deflections, sufficient stiffness, and vibration damping while minimizing mass. The structures provide the backbone for the system assembly. Bonds and threaded fasteners are assumed for the assembly of the components. Materials consist of carbon fiber reinforced cyanate ester and aluminum.

The structure consists of a space frame and truss with tubular members and shells with flanges. Decks are used to support equipment. The carbon fiber reinforced cyanate ester composite is M55J 6k/954 3 as noted in the Military Handbook MIL HDBK 17F [33]. The aluminum alloy is 7075 T6 as presented by Battelle Memorial Institute MMPDS [34]. The materials are at a minimum technology readiness of 6 (TRL6)[35]. The decks utilize a composite sandwich structure with M55J 6k/954 3 composite face sheets and an aluminum honeycomb core. Secondary structures consist of the installation hardware.

Stress analysis is by hand calculations and a spreadsheet. Quick checks are made under the given static launch loads. The 7075-T6 aluminum has an allowable stress of 364 MPa which is derived from the properties presented in the MMPDS and applied safety factors per NASA STD 5001B [36]. The laminated C/cyanate ester composite is laid up in a quasi-isotropic configuration, that is, [-45/0/45/90]<sub>2s</sub> and [45/0/45/90]<sub>4s</sub>. The allowable stress for the C/cyanate ester composites is 165 MPa using the MIL HDBK 17 properties, HyperSizer software [37], Tsai Hill failure theory, and NASA STD 5001B safety factors.

The maximum stress in the payload cone is 135 MPa under the maximum axial acceleration with a margin of 1.7 and 211 MPa under the maximum lateral acceleration with a margin of 0.7. The maximum stress in the Lander cylinder adapter is 21.9 MPa under the maximum axial acceleration with a margin of 15.7 and 52.3 MPa under the maximum lateral acceleration with a margin of 6.0.

Installation hardware mass is calculated as four percent [38] of the installed hardware mass and carried for all spacecraft subsystems.

Mechanisms are used to deploy hardware and to direct thrusters. The thruster panel gimbals are assumed to be scaled designs of a Moog Type 55. The radar boom uses a hinge with spring and lock. The nuclear reactor support boom has a hinge and a lock. Also, the nuclear reactor support boom has a deployment motor.

Structural risks may include excessive g loads or impact from a foreign object which may cause deformation, vibrations, or fracture of sections for the support structure. The long duration mission may raise the probability of such a hazard. Consequences include lower performance from mounted hardware to loss of mission. Mitigation would include designing to NASA standards to withstand expected g loads, a given impact, and to have sufficient stiffness and damping to minimize the probability of impact with foreign objects. Mechanisms face similar risks to that of the overall structure.

Recommendations for next steps include a higher fidelity structural analysis to determine modal frequencies and structural responses to static and dynamic loads from the launch cycle and operations. Greater use of advanced materials and architectures may help in increasing stiffness and reducing mass.

| Description                                | QTY | Unit Mass | Basic Mass    | Growth     | Growth       | Total Mass    |
|--------------------------------------------|-----|-----------|---------------|------------|--------------|---------------|
| Case 1_Enceladus Sample Return CD-2025-225 |     |           |               |            |              |               |
|                                            |     | (kg)      | (kg)          | (% )       | (kg)         | (kg)          |
| <b>Structures and Mechanisms</b>           |     |           | <b>1400.0</b> | <b>18%</b> | <b>252.0</b> | <b>1652.0</b> |
| <b>Structures</b>                          |     |           | <b>1142.5</b> | <b>18%</b> | <b>205.7</b> | <b>1348.2</b> |
| <i>Primary Structures</i>                  |     |           | 1056.2        | 18%        | 190.1        | 1246.3        |
| bus frame                                  | 1   | 32.3      | 32.3          | 18%        | 5.8          | 38.1          |
| bus side panels                            | 6   | 26.3      | 157.7         | 18%        | 28.4         | 186.1         |
| bus top deck                               | 1   | 53.5      | 53.5          | 18%        | 9.6          | 63.1          |
| bus bottom deck                            | 1   | 53.5      | 53.5          | 18%        | 9.6          | 63.1          |
| bus base flange to payload cone            | 1   | 32.3      | 32.3          | 18%        | 5.8          | 38.1          |
| stowed reactor support                     | 1   | 26.9      | 26.9          | 18%        | 4.9          | 31.8          |
| box boom base                              | 1   | 126.9     | 126.9         | 18%        | 22.8         | 149.7         |
| box boom top                               | 1   | 108.9     | 108.9         | 18%        | 19.6         | 128.5         |
| Payload Cone                               | 1   | 387.8     | 387.8         | 18%        | 69.8         | 457.6         |
| Lander Cylinder Adapter                    | 1   | 76.5      | 76.5          | 18%        | 13.8         | 90.2          |
| <i>Secondary Structures</i>                |     |           | 86.4          | 18%        | 15.5         | 101.9         |
| thruster wing                              | 2   | 33.3      | 66.6          | 18%        | 12.0         | 78.6          |
| RCS thruster mount                         | 4   | 1.6       | 6.4           | 18%        | 1.1          | 7.5           |
| Ka band antenna boom                       | 1   | 3.0       | 3.0           | 18%        | 0.5          | 3.6           |
| bus base flange to Lander cyl.             | 1   | 9.9       | 9.9           | 18%        | 1.8          | 11.7          |
| ice penetrating radar boom                 | 1   | 0.5       | 0.5           | 18%        | 0.1          | 0.6           |
| <b>Mechanisms</b>                          |     |           | <b>257.5</b>  | <b>18%</b> | <b>46.3</b>  | <b>303.8</b>  |
| <i>Power System Mechanisms</i>             |     |           | 17.0          | 18%        | 3.1          | 20.1          |
| boom hinge                                 | 1   | 5.0       | 5.0           | 18%        | 0.9          | 5.9           |
| boom lock                                  | 1   | 2.0       | 2.0           | 18%        | 0.4          | 2.4           |
| boom deployment motor                      | 1   | 10.0      | 10.0          | 18%        | 1.8          | 11.8          |
| <i>Science Payload</i>                     |     |           | 0.3           | 18%        | 0.1          | 0.4           |
| spring lock hinge, radar boom              | 1   | 0.3       | 0.3           | 18%        | 0.1          | 0.4           |
| <i>Adaptors and Separation</i>             |     |           | 123.1         | 18%        | 22.2         | 145.3         |
| Marman clamp ring, pass., payload cone     | 1   | 95.1      | 95.1          | 18%        | 17.1         | 112.2         |
| Marman clamp ring, pass., Lander cyl.      | 1   | 7.0       | 7.0           | 18%        | 1.3          | 8.3           |
| thruster panel gimbal, scaled type 55      | 2   | 10.0      | 20.0          | 18%        | 3.6          | 23.6          |
| antenna boom spring lock hinge             | 1   | 1.0       | 1.0           | 18%        | 0.2          | 1.2           |
| <i>Installations</i>                       |     |           | 117.1         | 18%        | 21.1         | 138.1         |
| C&DH Installation                          | 1   | 4.3       | 4.3           | 18%        | 0.8          | 5.0           |
| Comm. & Tracking Installation              | 1   | 1.5       | 1.5           | 18%        | 0.3          | 1.8           |
| Electrical Power Installation              | 1   | 10.0      | 10.0          | 18%        | 1.8          | 11.8          |
| Thermal Control Installation               | 1   | 53.4      | 53.4          | 18%        | 9.6          | 63.0          |
| Attitude Determination & Control           | 1   | 3.4       | 3.4           | 18%        | 0.6          | 4.0           |
| EP Propulsion Installation                 | 1   | 40.1      | 40.1          | 18%        | 7.2          | 47.3          |
| Chemical Propulsion Installation           | 1   | 2.5       | 2.5           | 18%        | 0.5          | 3.0           |
| Science Installation                       | 1   | 1.9       | 1.9           | 18%        | 0.3          | 2.2           |

Fig. 58 - Structures and Mechanisms MEL

## 6.0 Lessons Learned and Next Steps:

### 6.1 Lessons Learned

A 40-kW-class NEP vehicle using the Lunar/Mars Fission Surface Power system, when used in combination with new, commercial launch systems was found to *enable* an Enceladus mission. A comparison of NEP's performance compared to the workshop 'asks' from the introduction are shown in Table 15. A comparison to an all-chemical mission is included.

**Table 15 - Comparison of NEP and Chemical Enceladus Sample Return Missions to Demonstration Objectives**

| Option                                                                                      | NEP Round Trip  | NEP/Chem Split<br>(NEP w chem return & Moon<br>Orbiting tours) | Chem Round trip<br>(2-stage) |
|---------------------------------------------------------------------------------------------|-----------------|----------------------------------------------------------------|------------------------------|
| Use in concert with new large commercial rockets (Starship w x Tankers)                     | ✓ (15 tankers)  | ✓ (15 tankers)                                                 | ✓ (17 tankers)               |
| High DV destinations                                                                        | ~20 km/s*       | ~18/2.5 km/s*                                                  | 1.89 / 2.4 km/s              |
| Enabling much higher capability orbiters <i>and</i> landers                                 |                 | ✓                                                              |                              |
| Significantly reducing trip times                                                           | ✓ (~16.5 yrs)   | ✓ (~18.5 yrs)                                                  | (~21.5 yr)                   |
| Sample return                                                                               | ✓               | ✓                                                              | ✓                            |
| Increase launch window flexibility by reducing or eliminating the need for gravity assists. | ✓               | ✓                                                              | JGA (every ~ 20 yrs)         |
| Increase communication rates                                                                | ✓ (>5-12X)      | ✓ (>5-12X)                                                     |                              |
| Ground penetrating radar systems: icy moons/subsurface ice/liquid interface                 | ✓               | ✓                                                              |                              |
| Enabling the delivery and landing of capable vehicles on the ocean worlds                   | ✓               | ✓                                                              | ✓                            |
| 40 kW provided significant improvements in trip time and delivered mass                     | ✓               | ✓                                                              | N/A                          |
| Maximizing commonality between FSP and NEP subsystems (Time @ full power)                   | ✓ (9 yrs)       | ✓ (9 yrs)                                                      | N/A                          |
| Nuclear architectures may enable missions to reach multiple destinations                    |                 | ✓ (orbit multiple moons)                                       | N/A                          |
| Data Returned                                                                               | Sample & 0.7 Tb | Sample & 3 Tb**                                                | Sample & 0.5 Tb              |

NEP can either return a sample itself in ~**16 yrs** from launch OR return the sample chemically in ~**18 yrs** from launch while staying at Saturn and orbiting four more moons (including Titan). The NEP spacecraft has the capability to carry a large, ~6.5t lander. While not adding large amounts of science instruments, the NEP moon tour option can apply those instruments to five times the number of moons. The NEP can carry both ice-penetrating and surface mapping radars and increased communication rates/ data return by five to ten times.

The 40 kW NEP configuration relied on the flown NEXT ion propulsion system; eight strings were found to adequately propel the vehicle and process all the required xenon propellant. The impact of increasing reactor technology (temperature and materials) did not show notable improvement— *A quick mission investigation showed the same time of flight and approximately 700 kg less propellant.*

SERENE can be carried in a Starship, fully fueled and with only one boom fold, no additional reactor radiator deployments are required. The Starship with tankers is key in sending the ~ 27t NEP and lander vehicle to a high departure velocity.

#### 6.1.1 Comparison to Prometheus

Given the past promise of NEP for the 2000's Prometheus program and the resulting costs it is of interest to compare the SERENE design to Prometheus, primarily the Jupiter Icy Moons Orbiter (JIMO). While not a complete comparison and for a different target and mission, it still can show the potential impact of exploration's investment in a high power fission surface power system and large commercial launchers. Fig. 59 shows the top-level attributes of SERENE as compared with JIMO. At a top level, SERENE is significantly lower in cost than JIMO – mainly because SERENE assumed that the reactor and propulsion technologies would be available without novel technology development. The SERENE cost would need to be increased to add other technology developments (different radiators, large xenon storage, etc.) but the impact of not paying for reactor development is clear. The cost

of a Starship and 15 tankers is yet to be determined. All in all, these investments in FSP and Artemis launchers show promise for greatly expanding planetary science mission capabilities.

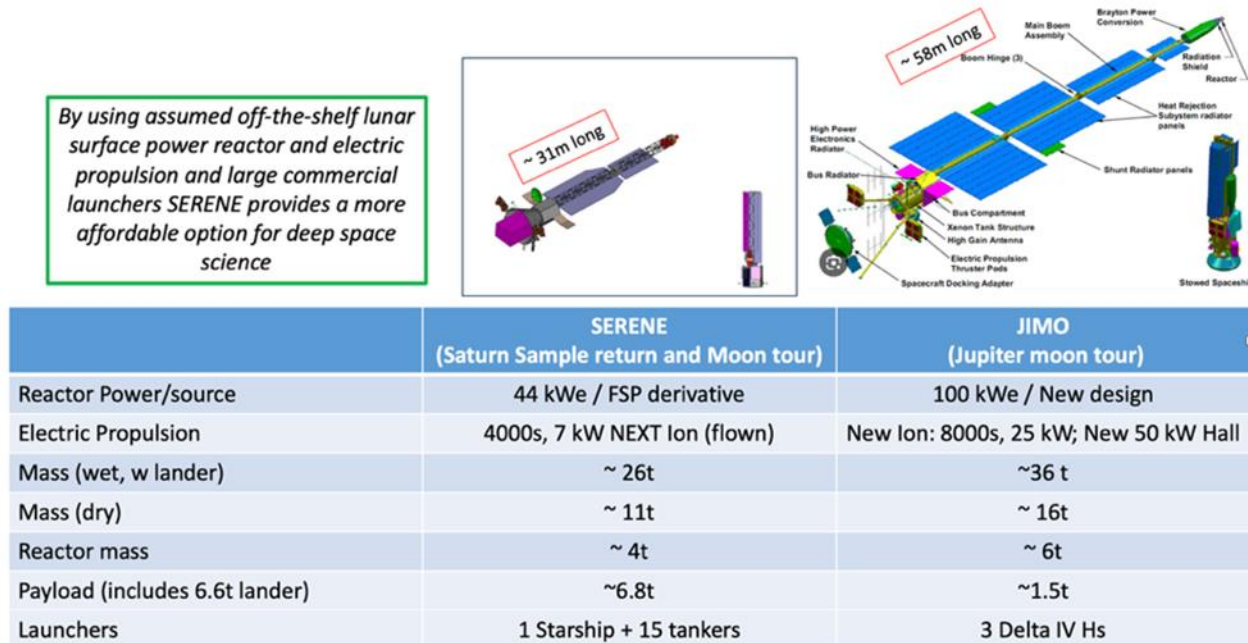


**Fig. 59 - Comparison of SERENE and JIMO**

## 6.2 Next Steps

Additional investigation is warranted if a mission such as SERENE is to be pursued. Primary next steps include refining the storable chemical equivalent design to better provide a comparison case, both developing a chemical 'stage' to transport the lander and a chemical return vehicle to Enceladus and reworking the Orbilander design to better serve as the return vehicle. Additionally, the use of cryogenic propellants for an all-chemical Enceladus vehicle should be examined, along with whether that will require a nuclear power system. Considering the use of a nuclear thermal propulsion stage for Earth departure may reduce the number of required Starships.

For the NEP sample return round trip mission, it would be beneficial to investigate the use of beamed power from the orbiter to the sample return lander. This may eliminate the need for radioisotope power systems on the lander.

Finally, a more thorough planetary protection analysis, both forward and backward, is appropriate.

## Acknowledgments:

The authors wish to thank the Space Nuclear Propulsion (SNP) office, and particularly Andrew Presby and Matthew Duchek, for commissioning this study and for their assistance throughout. Thank you as well to the Orbilander design team for providing an excellent reference for this design. Finally, thank you to Melissa McGuire for her technical review and feedback.

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