Nuclear Electric Propulsion for Outer Planet Science Missions

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Nuclear electric propulsion (NEP) combines the high specific impulse of electric thrusters with a constant power source that can operate anywhere in the solar system. Current investments in fission surface power offer a starting point for development of an NEP capability for deep space science missions, with a mission to the Saturn system illustrating the potential of such a system. Minimum time of flight transits, maximum payload delivery, and a sample return from Enceladus are considered. The NEP system can deliver payloads to the Saturn system with similar transit times to the Cassini spacecraft without requiring the planetary flyby maneuvers, and when coupled with a heavy-lift launch vehicle an NEP-powered spacecraft can complete a Saturn transit significantly faster than Cassini. High payload masses can be delivered on a longer trajectory at the expense of transit time. Replacing a portion of the payload with propellant allows for a high degree of maneuverability upon reaching the Saturn system. An Enceladus sample return mission closes when utilizing the NEP system for the Saturn departure and Earth return burns.

Nomenclature

α_{ps}	=	power system specific mass
ΔV	=	change in velocity
C_3	=	characteristic energy
kW_e	=	kilowatts electric
V_∞	=	velocity at infinity

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I. Introduction

THIS paper explores the use of nuclear electric propulsion (NEP) for science missions to the outer planets, leveraging technologies developed for fission surface power (FSP) systems and improvements realized through additional technology maturation. Systems delivering 10-40 kW of electric power (kW_e) to the propulsion system are investigated. NEP has a low technology readiness level but is under active development by NASA's Space Nuclear Propulsion Program [1, 2]. Rather than performing a point design of a specific NEP system operating in a particular way, this paper illustrates the effect of operating NEP systems of various power level in different ways to optimize for different objectives. Recent interactions with planetary scientists indicate that reduced trip time, increased launch window flexibility and enhanced maneuverability at the destination sphere of influence are higher priority objectives than delivering significantly greater payload masses relative to past missions [3]. The science mission benefits provided by an initial NEP system of this type are explored in this context. These benefits will increase as NEP system performance increases through technology improvements and block upgrades.

One type of mission enabled by NEP in this power class is a sample return from an outer planet icy moon. Missions to these moons are highlighted in the 2022 decadal survey [4]. Sample return missions from icy moons will likely experience increased interest and activity over the next decade. This interest builds upon the results of the Cassini mission to Titan, the Galileo and Juno missions to Europa, and the upcoming Europa Clipper mission, and the possibility of life in the subsurface oceans of these bodies. NEP has a unique ability to enable such missions when the performance and lifetime of the technology reaches sufficient levels [3].

The interplanetary portion of the reference missions considered in this study uses as a starting point the "NEP Benefits Study" [5], which examined 10 kW_e NEP systems using highly-enriched uranium (HEU) fuel. For this paper, high-assay low-enriched uranium (HALEU) fuel is assumed for estimating reactor masses to maximize commonality with NASA's Fission Surface Power (FSP) project as well as emerging terrestrial microreactor developments. The power range was also selected to match that expected for the on-going FSP project. Many mission studies illustrate operations within outer planet spheres of influence and those previous studies were used to develop the ΔV budgets for the missions considered here. These include Cassini, the Enceladus Orbilander [6], Enceladus Orbiter [7], and Titan Saturn System Mission [8]. JIMO-era studies of missions to Jupiter, Saturn, and Neptune considered higher-power NEP systems and in many cases low-thrust spiral trajectories for escape from Earth and capture and orbit insertion at the destination system [9]. When using relatively low-power NEP (< 100 kW_e) and attempting to achieve shorter trip times, it is important to depart from Earth with appreciable energy ($C_3 > 0$) and arrive at outer planets using a small chemical propulsion burn or moon flyby (or propulsive moon flyby) for orbit insertion as opposed to using long-duration low-thrust spirals.

II. Approach

A set of modules has been developed to operate in concert with the Copernicus low-thrust trajectory modeling software [10] to enable optimization of NEP missions for various objectives (e.g., minimum trip time, maximum payload delivered, etc.). Outputs are a function of the estimated spacecraft and power system masses, the electric power delivered to the thrusters, and the thruster efficiency and specific impulse. Earth to Saturn and Saturn to Earth interplanetary trajectories are modeled in Copernicus, using modules to separately calculate Earth departure energy and to estimate the propulsive needs within the destination gas giant sphere of influence. The Earth departure module enables examination of the tradeoff between initial spacecraft mass and the characteristic energy of departure (C_3) provided by the launch vehicle. This sets an initial condition on the Earth departure leg of the Copernicus solution. Arrival at Saturn is modeled by constraining V_∞ (the asymptotic velocity at infinite distance from the planetary body). The maneuvers performed within Saturn's sphere of influence (SOI) are not modelled in Copernicus, instead being captured by a module that estimates a propellant mass budget for this phase of the mission. Departure from Saturn and the Saturn-to-Earth return leg is modeled using a low-thrust trajectory in Copernicus with the arrival at Earth modeled by constraining V_∞. The trajectory solution is complemented by modules that estimate spacecraft mass as a function of the power provided by the NEP system. In this manner all phases of the mission are modeled, with the results of previous phases serving as the inputs for subsequent mission phases to realize an end-to-end sample return trajectory, such as the example trajectory shown in Figure 1 that includes an Earth gravity assist (EGA). Copernicus optimizes the interplanetary transfers as a function of power supplied to the electric thrusters and C_3 at Earth departure, calculating the optimal thrusting durations and splits between powered and unpowered (coasting) phases of the transfers.

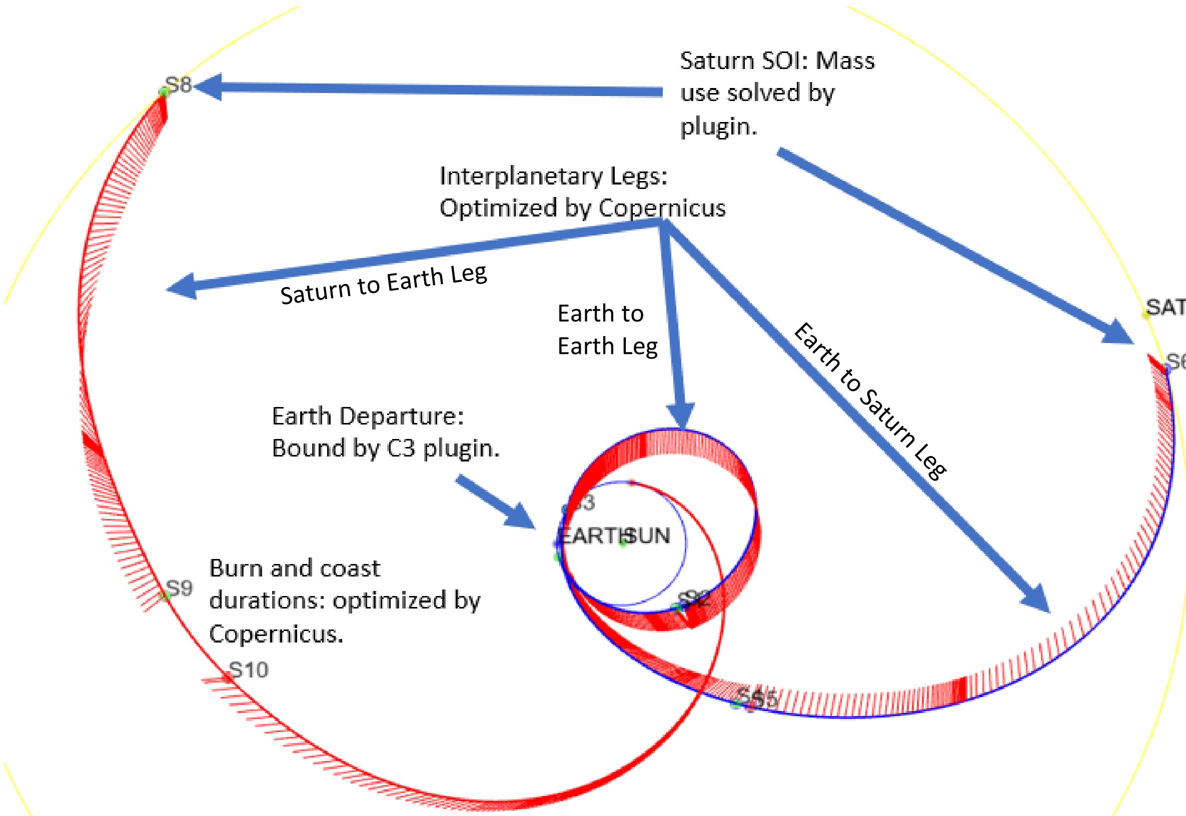

Figure 1. Saturn Sample Return Trajectory with EGA on Outbound Leg

Optimization of the NEP mission for a specific objective starts with the generation of a reference trajectory that is manually converged in Copernicus at the end of the parameter space where the Earth departure C_3 is lowest. These low C_3 trajectories are typically the hardest to computationally close. This converged solution is an ideal starting point for generating additional solutions in the parameter space. A Python code suite was developed to automatically develop converged solutions for additional points in the trade space using the Copernicus Python API. The Python code uses the reference trajectory as a starting point from which the next point in the trade space can be solved. Subsequent solutions use the next nearest solution as a starting point to ensure more rapid and accurate convergence.

The results presented in this paper use the methods described above to investigate the possible trade space for planetary science missions using NEP as the primary propulsion system for the interplanetary trajectory. Minimum trip time and maximum payload to Saturn are investigated to illustrate the bounding cases of the mission capabilities. Additionally, for sample return missions the payload to Saturn is combined with a ΔV budget for first reaching Enceladus once the spacecraft arrives in the Saturn system and then for returning from Enceladus to interplanetary space and ultimately to Earth. In this case, the delivered payload mass is traded for additional propellant, tankage mass and the number of thrusters required to provide the requisite ΔV budget and thruster lifetime to complete a sample return mission.

A. Mission Trajectory Modeling Approach

1. Minimum Time of Flight (TOF)

Given the science community's prioritization of reduced times of flight (TOF) to outer planet systems and a desire for more flexible launch windows, a direct Earth-to-Saturn trajectory was modeled and optimized for minimum TOF. Such a trajectory has a launch window each year, eliminates any inner solar system cruise, and avoids an Earth gravity assist maneuver, giving more mission planning flexibility. The variation in the ΔV required from year to year impacts the specific TOF for a given payload; an example launch year of 2029 +/- one year is used here. Trajectories were solved for various NEP system power levels and science payload masses. These types of direct trajectories are possible for several launch vehicles; Falcon Heavy-Expendable (FHe) and Space Launch System (SLS) are discussed as examples in this paper. Optimization for minimum TOF is also possible for a Saturn trajectory that uses an Earth gravity assist (EGA) (as shown in Figure 1); however, the Earth flyby adds a constraint to return to Earth about two years after launch, which limits the ability to minimize TOF in some instances.

To minimize the TOF the launch and arrival dates are both unconstrained so Copernicus can move the launch date forward or back in time to arrive at a more optimal trajectory. Multiple initial guesses were tested for launch date, TOF, C_3, and the thrust-to-weight ratio (T/W) of the spacecraft, resulting in local minima that occur roughly every 6 months. By utilizing multiple initial guesses and comparing the local minima, the global minimum can be found. Once the launch energy and the T/W are large enough, the trajectory subtends less than 180 degrees around the sun and the best initial guess is simple. For lower-energy launches, trajectories that travel greater than 180 degrees or even over 360 degrees around the sun may possess a more optimal TOF for a given payload.

The mass is adjusted on each iteration of Copernicus by the spacecraft mass sizing module described in Section II.B. The spacecraft mass sizing module utilizes a Python script containing a master equipment list (MEL) to calculate the mass of every spacecraft subsystem based upon the power level and required ΔV from each Copernicus iteration. The MEL in the model is discussed in greater detail in the next subsection.

The initial state of the NEP vehicle is constrained by the launch vehicle mass capability (e.g., a heavier spacecraft will begin in a lower energy orbit than a lighter one). Therefore, the initial orbital energy of the spacecraft is determined by the launch vehicle. Accounting for this changing initial orbit is possible with a simple script module for Copernicus. Equations were developed for different launch vehicles with each representing the relationship between spacecraft mass and that launch vehicle's capability to impart a given C_3 to that mass.

2. Maximum Final Mass

The other type of optimization investigated for this paper was maximizing the final mass of the spacecraft at Saturn. After some post-processing that accounts for the mass of the spacecraft, this allows for the calculation of the maximum possible payload delivered to Saturn for a given NEP system power level. This additional mass could be used for several purposes, including, for example, the provision of additional ΔV-capability while the spacecraft is in the Saturn system or the increase in the size of a lander. To maximize spacecraft mass at Saturn, an Earth-Earth-Saturn trajectory was used, with the single Earth flyby after a two-year orbit. The Earth gravity assist provides additional energy enabling a larger payload while maintaining launch window flexibility by only relying on the relative orientations of Earth and Saturn. This optimization is easily performed in Copernicus by setting maximum mass as the objective function while leaving initial mass and time of flight unconstrained. The division of mass between vehicle systems and payload is determined in post-processing script rather than concurrent with the solution. As in the previous subsection, the maximum possible spacecraft mass and departing C_3 are constrained as a function of launch vehicle, again using a Falcon Heavy Expendable or SLS.

The same interplanetary solution is then patched to the operations in the sphere of influence of the destination. A few options are considered in this study. The simplest is inserting a spacecraft into an elliptical orbit possessing a roughly 215-day period. In this case, excess delivered mass can be allocated for science instruments or for additional propellant/tankage to perform maneuvers in the Saturn system. Another case considered is for a payload delivered into Enceladus orbit, with a lander descending to the Enceladus surface. An SLS-launched case is considered that carries enough usable mass to allocate the propellant required for reversing the series of maneuvers to Enceladus and returning the spacecraft to Earth.

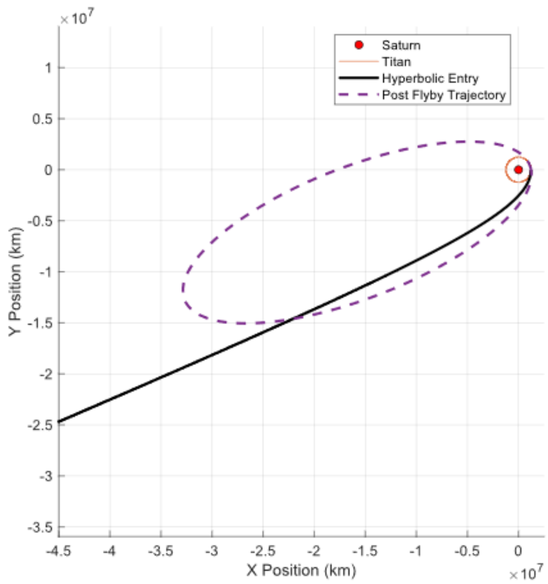

Figure 2. Example Saturn orbit insertion by Titan flyby from V_∞ = 2 km/s into a 218-day period orbit.

For all the missions examined, the spacecraft arrives with 1.2-2.0 km/s V_∞ at Saturn, depending whether the mission is performing a simple Saturn orbit insertion or is targeting a Saturn moon tour. This value can be further optimized in future work. Higher arrival velocity can shorten the trip by months, but it comes at the cost of a higher ΔV orbit insertion requirement. With a 2 km/s arrival, a 150 m/s burn at a periapsis radius of 71,000 km inserts the spacecraft into a highly elliptical roughly 215-day period orbit. From here, a periapsis raise maneuver can be performed by either chemical RCS or the EP system (~500 m/s for an impulsive chemical burn) to put the spacecraft on a trajectory to intercept Titan's orbit near periapsis. This would use an appreciable amount of the available excess delivered mass at Saturn. Alternatively, a Titan flyby can be targeted to accomplish the Saturn orbit insertion. With a V_∞ at Saturn of 2 km/s, an unpowered Titan flyby can capture the spacecraft into a Saturn orbit with a roughly 2.6 km/s V_∞ with respect to Titan. This possibility is enabled by the low V_∞ arrival to Saturn that is possible using low thrust electric propulsion on the interplanetary transfer.

A leveraged moon flyby sequence is used to transfer to Enceladus orbit once in Saturn sphere of influence. Similar sequences can be found in the Titan Saturn System Mission concept [8] (which eventually reduces to an orbit of Titan followed by a Titan-Enceladus transfer) or that proposed by Campagnola *et al*. [11] (to reduce V_∞ with respect to Enceladus to a level sufficient to perform a roughly 130 m/s orbit insertion at Enceladus). After an initial moon flyby, these types of leveraging trajectories use small burns at apoapsis or periapsis to target subsequent flybys of moons in the planetary system. These flybys need not happen at periapsis or apoapsis, so the V_∞ of the spacecraft with respect to each moon's SOI is different on each flyby. The leveraging maneuvers allow the spacecraft orbit to be adjusted over multiple flybys to alter its overall orbit about Saturn. The magnitude of the leveraging maneuvers is small enough (on the order of 10-30 m/s per maneuver) that the NEP system has sufficient thrust to perform the maneuvers. The disadvantage of this approach is the time it takes to perform the multiple lunar flybys.

Campagnola *et al*. [11] uses a V_∞ with respect to Titan of roughly 1.4 km/s to begin the leveraging tour. Connecting the NEP-powered spacecraft to such a tour requires either budgeting for up to 750 m/s of ΔV provided by a chemical propulsion system for a powered Titan flyby or employing an additional Titan flyby leveraging orbit. The latter option is more propellant-efficient but comes at the cost of at least another six months in a highly elliptical orbit of Saturn.

The round-trip missions utilize a module that calculates the mass expended within the Saturn sphere of influence, which sets the spacecraft mass at Saturn departure for Earth. This includes the moon tour down from the initial highly-elliptic Saturn orbit, the Enceladus orbit insertion and departure, and a reversal of the moon tour through escape from the Saturn system. The module utilizes the rocket equation to solve for the propellant mass expended by the spacecraft while in the Saturn sphere of influence, using the spacecraft mass of arrival at Saturn and then solving for the mass at Saturn system departure. This sphere of influence module assumes specific entry and exit V_∞ values at Saturn, which are used as constraints on the Copernicus model. There are also assumed the I_{sp} values for both the reaction control system (RCS) thrusters and electric propulsion system. Should a different moon tour be selected, the module can be easily updated with a new in-system ΔV budget and arrival V_∞. It can also be used for different electric propulsion systems by changing the assumed I_{sp} value.

B. Spacecraft Sizing Approach

All spacecraft subsystems are modeled, at various levels of fidelity, with reference to the various studies cited in this paper. Table 1 provides a MEL for an example point design. An RCS budget is carried for orbit maintenance and other maneuvers (like moon orbit-insertion). A routine converges all the subsystem mass and RCS propellant parametrically as a function of overall vehicle mass. A 25% margin on wet mass is carried in the launch vehicle module when calculating the departure C_3. The predicted spacecraft mass is then flown through the trajectory.

The NEP power system mass per unit power produced (power system specific mass or α_{ps}) is a key input to the model. Two curves of power system alpha as a function of generated power are considered, representing a Generation 0 system (FSP-derived) and a Generation 1 system (incorporating technology advancement beyond the FSP-derived system). These curves are provided in Figure 3. (In the solutions presented in this paper, the power system mass is treated fully parametrically and represented by a smooth curve.) An actual system would have a reactor designed at a specific power level and power conversion system

Table 1. Example spacecraft MEL for 10 kW$_e$ NEP system with 300 kg payload.

	Basic Mass	Predicted Mass (with MGA)
NEP Power System	1251 kg	1438 kg
EP Thrusters	69 kg	72 kg
EP Tanks	146 kg	160 kg
RCS Tanks	50 kg	63 kg
Structure	327 kg	392 kg
Solar Power System	187 kg	225 kg
Thermal Management	60 kg	78 kg
Comms	54 kg	65 kg
Cmd & Data Handling	35 kg	42 kg
Guidance, Nav, Control	25 kg	30 kg
Payload	300 kg	300 kg
Spacecraft Dry Mass	**2504 kg**	**2865 kg**
EP Propellant	1067 kg	1067 kg
RCS Propellant	331 kg	331 kg
Spacecraft Wet Mass	**3902 kg**	**4263 kg**

units sized accordingly. The number of active units at any one time could be tailored to a specific mission to provide near-optimal power throughout the mission.

The power system specific mass curves represent a parametric scaling based on best available technology examples currently in development or from past development efforts [12, 13, 14, 15, 16, 17]. The masses of the electric thrusters and their power-processing units (PPUs) are bookkept separately from α_{ps}. The thruster mass also scales with power and is dependent upon the number of thruster units needed. For this paper, we assume Xe-fed ion thrusters operating at 4000 sec I_{sp} and 63% system efficiency (including PPU efficiency and performance margin), with the mass taken as that of the NEXT thruster and PPU.

For long duration missions, such as the sample return mission, it is likely that additional redundancy or mass allocation will be required to increase the margin on lifetime of the NEP system. In the analysis that follows, mission performance is calculated by determining the amount of payload deliverable with a NEP system of the assumed specific mass. Further work is required in more detailed mission studies to refine the specific mass that can be achieved for a given life and reliability threshold and to optimize missions using that information. The results in the present paper are intended to demonstrate what is possible if the assumed NEP specific mass curves can be achieved through technology maturation efforts, and to motivate further refinement as that maturation occurs.

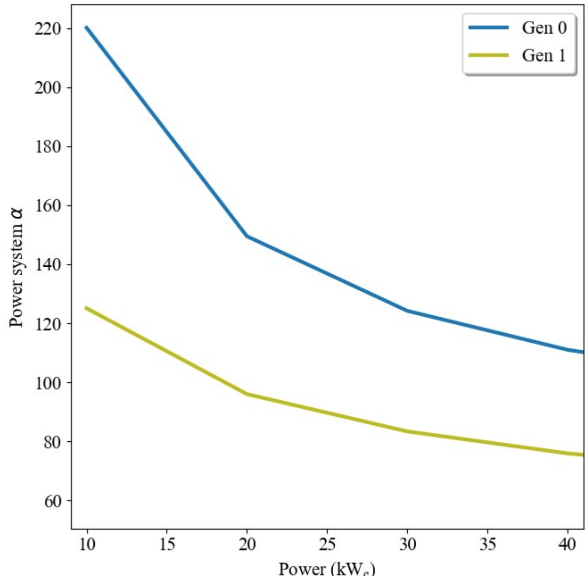

Figure 3. NEP power system specific mass (α_{ps}) in kg/kW$_e$ as a function of generated power.

III. Results

Presented in this section are results of mission analyses optimized minimum trip times to Saturn, maximum payload delivery to an orbit around Saturn, and maximum delivery of payload to Enceladus within the Saturn system. An Enceladus sample return mission is also assessed using the Generation 1 power system described in the previous section. Mission analysis results include the ΔV budget for the mission, spacecraft mass estimates, and trajectory information such as durations of thruster operation for each segment, number of coast phases, and total mission duration. Multiple launch vehicles are considered, as the launch vehicle capability affects the tradeoffs in mission design.

A. Minimum Time Transits to Saturn

By operating the NEP system to minimize trip time at power levels up to 40 kW$_e$, a payload can be delivered to the destination system more quickly than past missions such as Cassini without the need for planetary flybys. Figure 4 shows an example direct trajectory with the red lines pointing in the thrusting direction during the interplanetary transfer. This trajectory takes advantage of the high I_{sp} EP system all the way to the destination. A chemical orbit insertion burn (or Saturn orbit insertion using a Titan flyby) is needed to avoid spiral-down time at the destination. The magnitude of the chemical burn is relatively small because a low V_∞ at arrival can be efficiently targeted by the EP transit maneuvers. A higher arrival V_∞ corresponds to an even faster transit time, but above 2.5 km/s the Titan flyby for orbit insertion is replaced by a Saturn orbit insertion burn at a low periapsis, requiring more propellant mass for the insertion burn, which negates any transit time advantage.

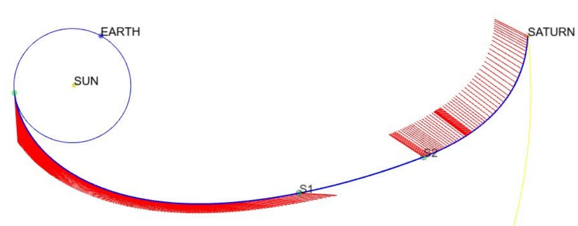

Figure 4. Example direct NEP trajectory to Saturn – 40 kW$_e$ with SLS launch.

To minimize trip time, some of the usable mass available at destination is traded for additional NEP system power and a commensurate increase in EP propellant. The EP system provides more total ΔV to the spacecraft, shortening the trip time. There is an optimal power to minimize trip time, above which the additional power system mass reduces the C_3 provided by from the launch vehicle enough to outweigh the increased ΔV a higher power NEP system can provide.

Figure 5 shows the relationship between NEP system-produced power and the time of flight (TOF) to Saturn for a Generation 1 NEP system with various payloads (payload defined as usable mass for maneuvering or science mission hardware at the destination planet). By way of comparison, the Cassini mission TOF was 7 years from launch to Saturn arrival and included two gravity assists at Venus, one at Earth, and a final one at Jupiter (VVEJ), necessitating a highly-constrained launch window with significant trip time penalties if the primary launch window was missed [18]. Also worth noting for reference is Cassini's orbiter science payload, which massed 351 kg and utilized 228-314 W of power, and the Huygens probe, which massed 343 kg. In the example presented in Figure 5, the V_∞ at arrival to Saturn is constrained to less than 2 km/s, which is low enough to allow orbit insertion using an unpowered Titan flyby. There is a larger option space to explore that includes powered flybys or typical chemical propulsion orbit insertion maneuvers that may allow slightly faster trip times with similar payloads. In general, these results demonstrate that an NEP system possessing the assumed α_{ps} can deliver highly capable payloads to the Saturn system times comparable to or less than the Cassini interplanetary flight without the need for planetary flybys, providing for much greater launch window flexibility. For Falcon Heavy expendable, trip times of about one year shorter are possible by adding an Earth flyby. If an SLS is used, a time of flight of less than four years can be achieved when coupled with a 40 kW$_e$ NEP system. While trip times of less than five years may not be quite fast enough to be "game-changing" for science missions ("game-changing" being defined in this instance as a 2-3 year time of flight [3]), the elimination of planetary flybys is a significant risk reduction that increases mission planning flexibility by providing launch windows at approximately one year intervals. Also of importance is that the nuclear power system is capable of providing almost two orders of magnitude greater power at the destination relative to Cassini, and once in-system that power is available for science instruments or high data-rate communication with Earth.

(a) (b)

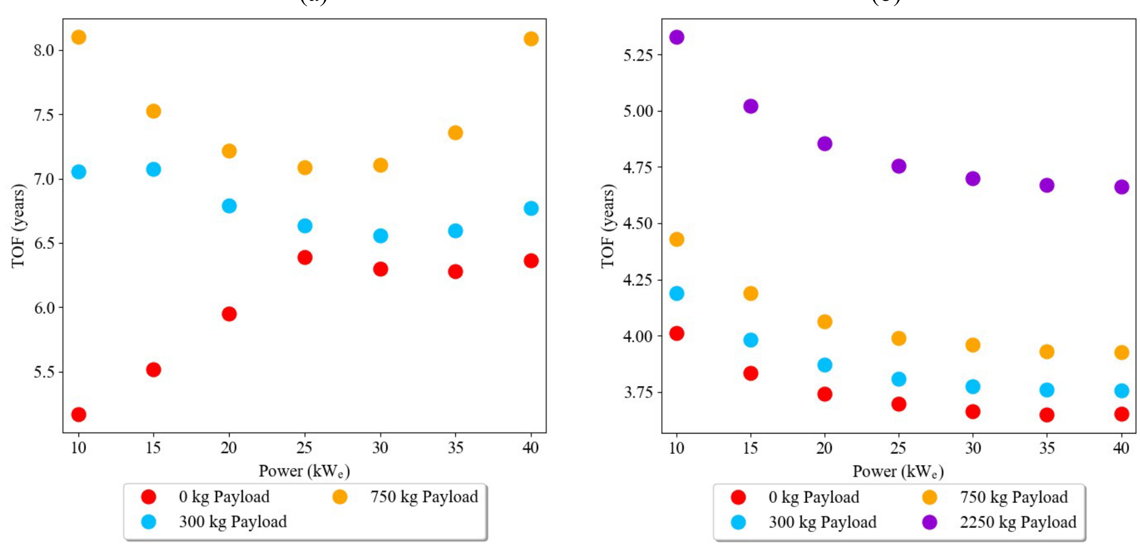

Figure 5. Time of Flight on an Earth-Saturn direct trajectory as a function of a Generation 1 NEP power system for various payloads (allocable mass at Saturn system) for (a) an FH-expendable launcher or (b) an SLS launch vehicle.

Worthy of note is that the minimum trip times when launched on SLS are achieved for a 40 kW$_e$ NEP system. With less capable launch vehicles and for moderate payloads (e.g., 300 kg in Figure 5(a)), the optimum power to minimize trip time falls within the middle of the 10-40 kW$_e$ range, but TOF is relatively insensitive to power over the entire range shown (all within half a year). There is a discontinuity in the trend (e.g., Figure 5(a) 0 kg payload below 25 kW$_e$) where the balance of available launch energy, spacecraft mass, and spacecraft thrust-to-weight ratio requires a longer (for higher power) or shorter (for lower power, case illustrated below 25 kW$_e$) spiral about the Sun to minimize the TOF. This aspect of the solution space is not relevant for missions launched by SLS, which have sufficient launch energy to minimize trip time without additional spiral time across the range of power investigated.

Figure 6 illustrates the corresponding ΔV provided by the NEP system for the minimum TOF solutions summarized in Figure 5. For this mission design exercise, TOF is minimized by maximizing the NEP-provided ΔV. This ΔV could be reduced for a mission, trading TOF for other considerations such as added margin. The discontinuity in trajectory geometry that affects Figure 5(a) is also seen in Figure 6(a).

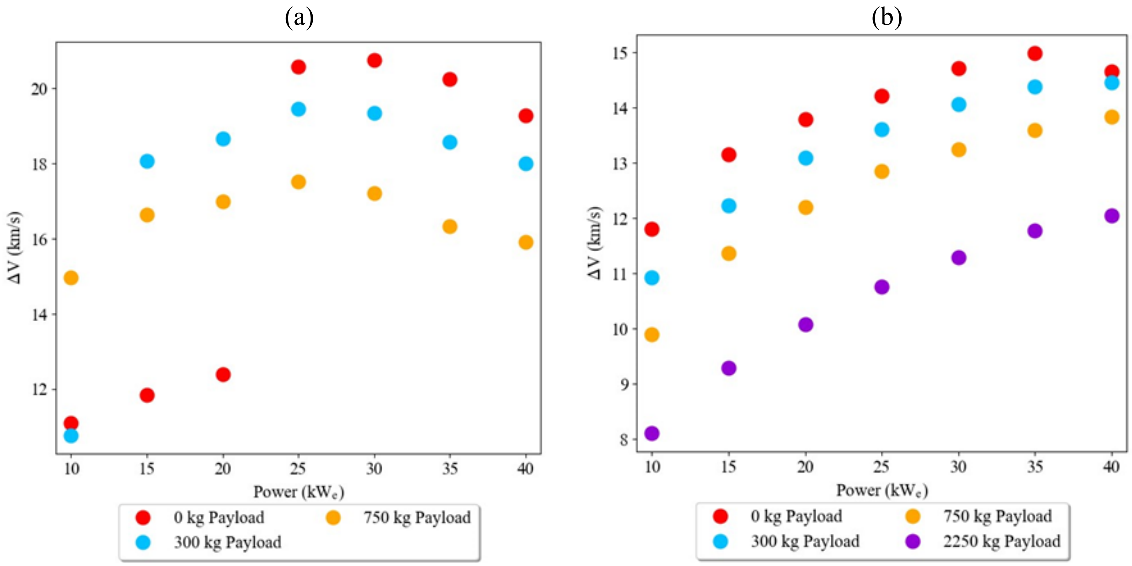

Figure 6. Total ΔV on an Earth-Saturn direct trajectory as a function of a Generation 1 NEP power system for various payloads (allocable mass at Saturn system) for an (a) FH-expendable launch or (b) SLS launch vehicle.

B. Maximum Payload to Saturn System Destinations (Enceladus)

NEP, coupled with modern launch vehicles, can deliver a large amount of usable mass to the Saturn system. The large delivered mass is enabled by a combination of attributes of the NEP trajectory and the Saturn system. A system using NEP can fly a trajectory that has a low arrival V_∞ (< 2.5 km/s) at Saturn without any time-sensitive gravity-assist maneuvers (such as a Jupiter flyby). An Earth-Earth-Saturn trajectory, as shown in Figure 7, illustrates the upper bound of deliverable payload with NEP. With a low arrival velocity, only a small chemical orbit insertion burn (< 200 m/s) is required. A significant periapsis raise maneuver (500-800 m/s) is required to begin a moon tour flyby sequence, leveraging Titan to pump down the orbital energy in the Saturn system. However, with a low arrival velocity, orbital insertion by an unpowered or powered (0-750 m/s) Titan flyby is also a feasible option which additionally removes the need for a periapsis raise maneuver.

Once on an elliptical orbit with periapsis near Titan's orbital altitude, a moon tour can be executed with small maneuvers (1's-10's of m/s) performed by either chemical propulsion or electric propulsion to leverage and target subsequent flybys [11]. This results in a ΔV budget of roughly 300 m/s (impulsive) to reach Enceladus orbit. We apply margin to the budget if the EP system is used to perform this sequence, but even then the required (low-thrust) ΔV is only 500 m/s. The spacecraft then performs an Enceladus orbit insertion burn as previously shown in Refs. [6], [11]. The flyby sequence developed in past studies has the primary aim of minimizing the ΔV required to reach Enceladus. When able to perform the maneuvers with the high I_{sp} EP system, the goal of minimizing the in-system ΔV can be somewhat relaxed. The design of the tour could use larger leveraging maneuvers to

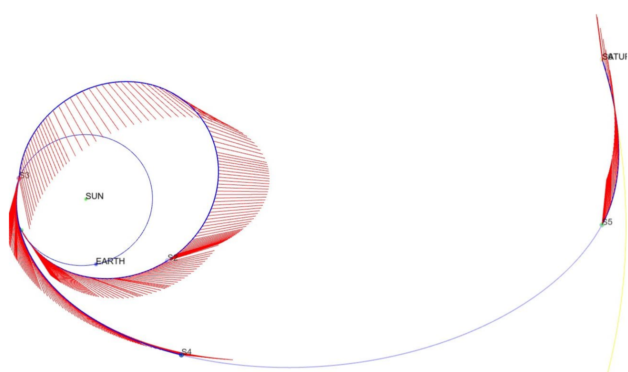

Figure 7. Example NEP trajectory to Saturn: 40 kW$_e$ NEP system using an SLS launch vehicle and Earth gravity assist.

reduce the number of flybys required and thus the total tour duration. This is another area for future study for any planetary system where an NEP mission may operate.

Shown in Figure 8 is the maximum payload mass at Saturn arrival and the maximum mass landed on Enceladus enabled by Gen 0 and Gen 1 NEP systems using either (a) a Falcon Heavy expendable launch vehicle or (b) an SLS rocket. Again, the baseline for comparison is the Cassini mission, which delivered a total mass of ~3500 kg to its initial Saturn orbit (after the orbit insertion burn and periapsis raise), which included roughly 700 kg of science payload (Cassini orbiter and Huygens probe) and roughly 1000 kg of propellant to provided additional ΔV for maneuvers and attitude control [19, 20]. Presented in Figure 9(a) are the spacecraft initial mass and its mass at Saturn arrival, while Figure 9(b) shows the corresponding transit times to Saturn based using either a Falcon Heavy Expendable (FHe) or SLS launch vehicle. For the FHe, the trade optimizes toward the lowest powers considered (10 kWe) because the trip

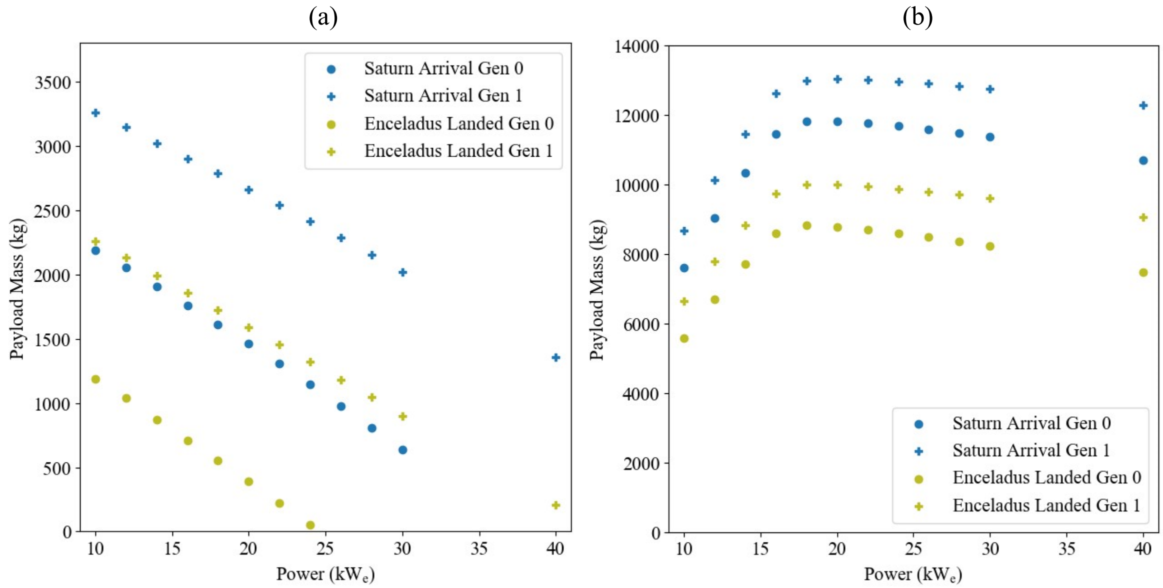

Figure 8. Maximum delivered payload mass at Saturn arrival and maximized mass landed on Enceladus for an Earth-Earth-Saturn trajectory using a Gen 0 or Gen 1 NEP system and (a) an FH-expendable launch vehicle or (b) an SLS launch vehicle.

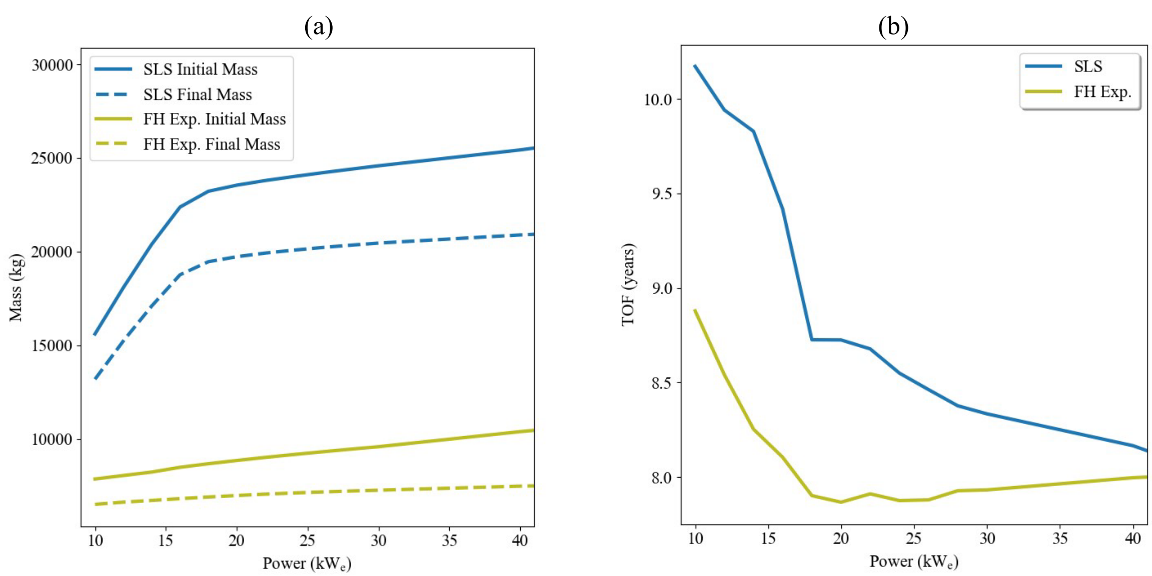

Figure 9. (a) Initial mass of spacecraft at Earth and maximized final mass at Saturn and (b) corresponding time of flight from Earth to Saturn for an FH-expendable or SLS launch vehicle.

time is unconstrained (up to 10.5 years to Saturn arrival). For the higher C_3 launcher (SLS), a higher power NEP system reduces the interplanetary ΔV enough to save propellant and compensate for the added power system mass up to roughly 20 kW$_e$. There may be a better trajectory solution this team did not find that would allow more payload using a lower power NEP system. However, the high energy launches possible with SLS led to the interplanetary transit optimizing such that no coast period was needed to maximize payload. More work is needed to design for a specific mission if such a high-energy launch is considered.

In the case of longer trip times, the total amount of payload deliverable to Enceladus when using an NEP system for the interplanetary transfer is potentially an order of magnitude higher than past missions. This paper does not consider specific mission designs, but with the potential deliverable payload, Enceladus landers at least as capable as the Orbilander concept [6] could be flown, with remaining margin to optimize trip time or allow for additional uncertainty in the NEP power system that may ultimately be available for such missions (e.g., an 20 kW$_e$ system flown using a FHe could also close such a mission despite being higher-than-optimal power). As before, a high data-rate communication system could be included on this type of mission to use the NEP power generation capabilities once the destination is reached, allowing for the transmission of significantly greater data compared to previous, very power-limited spacecraft.

C. Conceptual Closure of an Enceladus Sample Return Mission

Beginning from missions that deliver the maximum payload to Enceladus (examined in Section III.B), it is possible to consider the requirements for a sample return mission. Instead of a large science payload to Enceladus orbit, mass can be allocated for additional propellant to close a Saturn-to-Earth return journey. The assessment of closure here uses a Generation 1 assumption and is conceptual and preliminary. An end-to-end model of the interplanetary portion of the mission using an Earth gravity assist on the outbound leg (Earth-Earth-Saturn outbound and Saturn-Earth inbound) was used to calculate the ΔV budget for the interplanetary trajectory. The solver maximized the final mass at Earth return given an NEP power level, with spacecraft subsystem masses determined by post-processing the result in the manner described in Section II.B. For Saturn arrival to Enceladus orbit capture, the same mission as in Section III.B is assumed, with the small difference of a 1.2 km/s V_∞ at Saturn arrival to reduce the time and energy needed to begin the moon tour. For the mission phase starting with Enceladus departure and ending with the departure from Saturn's sphere of influence, the moon tour maneuvers are reversed. A detailed trajectory for this portion was not constructed, but there are no fundamental reasons why a roughly equivalent reverse series of maneuvers could not be performed. Feasible Saturn escape approaches include following an EP-powered low-thrust spiral trajectory for either direct escape or through the use of a Titan flyby, reversing the capture maneuver. Both options were examined, and the Titan flyby for escape is used in the analysis reported here. The ΔV budget for the mission is summarized in Table 2.

Table 2. Conceptual ΔV Budget for Enceladus Sample Return Mission with a 20 kW$_e$ NEP system.

Maneuver	ΔV Budget	Duration (Burn Length)
Earth to Saturn (series of 3-4 burns)	Launch C_3 = 44 km^2/s^2 EP to Saturn ~10 km/s	7.5 years (6.7 years)
Saturn orbit insertion (SOI)	0 m/s Titan flyby capture	
Moon Tour to Enceladus	380 - 1200 m/s EP Leveraging tour	2.7 - 5 years down
Enceladus Orbit Insertion/Departure	260 m/s chem for EOI and EOD	1-year orbital ops
Moon Tour Up	380 - 1200 m/s EP	2.7 - 5 years up
Saturn departure	0 m/s Titan flyby escape	
Saturn to Earth Burn 1	~ 8.5 km/sec EP	9.2 years (8.4 years)
Saturn to Earth Burn 2 (slow to Earth Arrival velocity)	~10 km/s EP (~ 12.5 km/s entry velocity [21])	
RCS for miscellaneous	75 m/s RCS for maintenance reserved to end of mission	
TOTAL	• **EP ΔV of roughly 29 - 32 km/s** • **Chemical bi-prop ΔV of 335 m/s**	**22.5 - 27 years**

Figure 10 shows the calculated NEP ΔV available to the spacecraft (based on available xenon propellant) as a function of required RCS budget and launch vehicle C_3, given the parametric vehicle model used for this study and an initial guess of the proportion of EP-provided ΔV required for the Earth to Saturn outbound versus the Saturn to Earth inbound legs. This diagram was used to inform initial closure analysis.

Figure 11 shows the payload returnable to Earth given these assumptions and Table 3 provides the summary parameters for a 20 kW_e Generation 1 NEP spacecraft performing the mission. A high-energy launch vehicle like SLS (or potentially a Starship with a kick stage) is needed to close the mission. With SLS, Earth return closes with a maximum of up to roughly 2000 kg usable payload mass available for a reentry probe, lander, and any other science instruments. The total mission time is between 22 and 27 years with this large payload, depending on the specific design of the Saturn moon tour up and down. With a mission that uses a 300 kg sample collection vehicle (for Enceladus operations) and 300 kg return entry vehicle (larger than Stardust [21]), this leaves significant usable mass that could be allocated to additional propellant and power system mass as needed to reduce the interplanetary transit times (by about 1 year) or the duration of the moon tour. More investigation is required to quantify this trade-off. Sufficient margin is left to consider the mission for any of the yearly launch windows for an Earth-Earth-Saturn trajectory. Alternatively, this represents 27% additional dry mass margin above the built-in launch vehicle margin. It is worth noting that given the large initial mass of the vehicle, 10 kW_e does not appear to be enough power to optimize the trajectory; the thrust-to-weight ratio is too low for the best EGA and an efficient Earth-Saturn leg. However, 20 kW_e appears to be optimal based on the assumed models, but this is sensitive to the uncertainty in α_{ps}.

Specific optimization of this mission is left to future work, but the authors expect a mission duration of under 20 years for an Enceladus sample return to be possible using a generation 1 NEP system with sufficient lifetime and reliability to close this mission. Future work includes a balancing of system lifetime, redundancy/reliability, and nominal power level, which will point to the specific attributes such a generation 1 system would require.

IV. Conclusion

Use of an NEP system enables high-specific impulse, constant low thrust propulsion to the outer solar system. This permits minimization of the

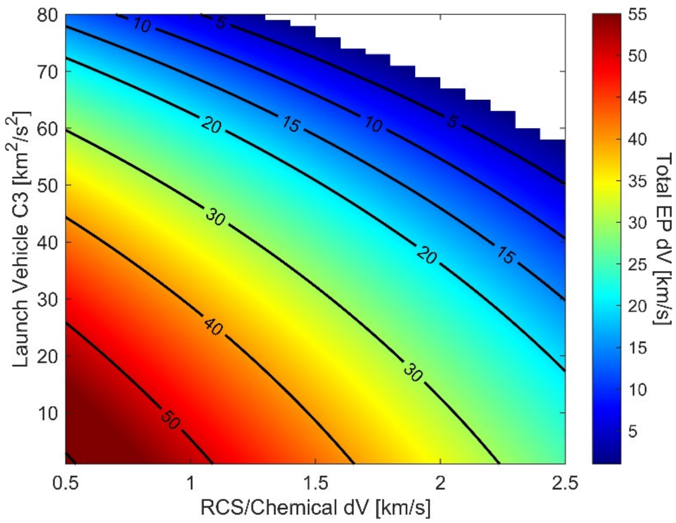

Figure 10. Possible EP ΔV with a Generation 1 NEP and 300 kg payload as a function of RCS ΔV requirement and LV C_3.

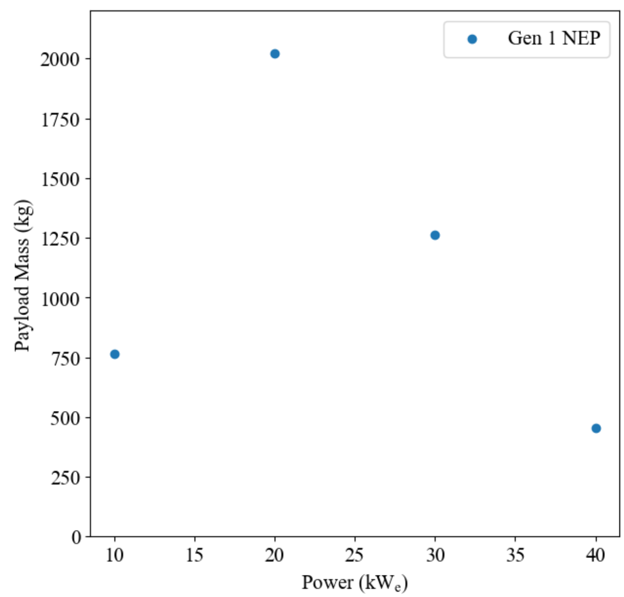

Figure 11. Payload returnable to Earth for Enceladus sample return mission as a function of NEP power level.

Table 3. Sample Return Spacecraft Summary

System	Description
Propulsion	20 kW_e, 4000 sec I_{sp} ion Thruster
Spacecraft Mass (without payload)	~16,100 kg Initial Mass (25% LV margin) ~4,700 kg Predicted Dry Mass (includes 15% MGA) 1000 kg RCS propellant 8400 kg EP propellant
Payload Example (2000 kg returned)	300 kg sample return entry vehicle 300 kg lander/collector 1,400 kg additional usable mass (27% additional dry mass margin)

arrival velocity V_∞, reducing the amount of chemical propulsion required for capture maneuvers (and in some cases allowing capture through a flyby of a planetary moon). NEP also enables high-I_{sp} maneuvering within planetary spheres of influence, greatly reducing the propellant required for tours of the planet's moons while simultaneously increasing the payload mass that can be delivered to specific destinations. In some cases, NEP can enable faster trip times to outer planet destinations using smaller launch vehicles compared to what would be required using all-chemical propulsion systems, such as was previously used by the Cassini mission. For Saturn missions, preliminary investigations indicate NEP systems with power levels between 20-40 kW_e are required to sufficiently reduce trip time. Lower α_{ps} leads to further reduction in trip time and increases mission flexibility. An NEP-powered spacecraft can also increase launch window flexibility by closing direct-transfer and single Earth-gravity-assist trajectories. Finally, NEP appears able to close a sample return mission from Enceladus. A low-α_{ps} power system operating at 20 kW_e is the most likely system to enable this mission based on current models, requiring an NEP system capable of 17-20 years of operation. Achieving this lifetime may require redundant lightweight, redundant propulsion and power systems. There remains significant uncertainty in many of the technology performance assumptions used in this paper, and further study is required. This includes NEP technology development performed with a goal of obtaining more accurate metrics on performance, power, and mass scaling and system lifetime and reliability.

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