

Electric Propulsion System Selection Process for Interplanetary Missions

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The disparate design problems of selecting an electric propulsion system, launch vehicle, and flight time all have a significant impact on the cost and robustness of a mission. The effects of these system choices combine into a single optimization of the total mission cost, where the design constraint is a required spacecraft neutral (non-electric propulsion) mass. Cost-optimal systems are designed for a range of mass margins to examine how the optimal design varies with mass growth. The resulting cost-optimal designs are compared with results generated via mass optimization methods. Additional optimizations with continuous system parameters address the impact on mission cost due to discrete sets of launch vehicle, power, and specific impulse. The examined mission set comprises a near-Earth asteroid sample return, multiple main belt asteroid rendezvous, comet rendezvous, comet sample return, and a mission to Saturn.

Nomenclature

I_{sp}	=	specific impulse, s
m_0	=	mass at Earth escape, kg
m_f	=	delivered mass to final target, kg
m_{C3}	=	launch mass, kg
m_N	=	non-SEP neutral mass, kg
m_p	=	propellant mass, kg
P_0	=	solar array power at 1 AU
TOF	=	flight time, yr.
χ_{m0}	=	launch vehicle cost coefficient, \$/kg
χ_{TOF}	=	flight time cost coefficient, \$/yr.
χ_p	=	propellant cost coefficient, \$/kg
χ_{p0}	=	power cost coefficient, \$/kW
μ_p	=	propellant mass coefficient, kg/kg
μ_{p0}	=	power mass coefficient, kg/kW

I. Introduction

Recent missions such as Dawn,¹ SMART 1,² and Hayabusa³ demonstrate the expanding role of solar electric propulsion (SEP) in the robotic exploration of our solar system. As future missions with electric propulsion are planned, traditional design techniques, largely developed for chemical propulsion systems, may prove inadequate. For example, with chemical propulsion the spacecraft trajectory can mostly be designed independently of the launch vehicle, which is selected to accommodate the spacecraft mass; power and propulsion system design have a secondary effect on the available mass. With electric propulsion missions the launch vehicle, solar array power, and number and type of thrusters combine to affect the spacecraft trajectory and delivered mass, requiring concurrent design of these systems.^{4,5,6} The flight time to the target also alters the trajectory and may effect the feasibility of a

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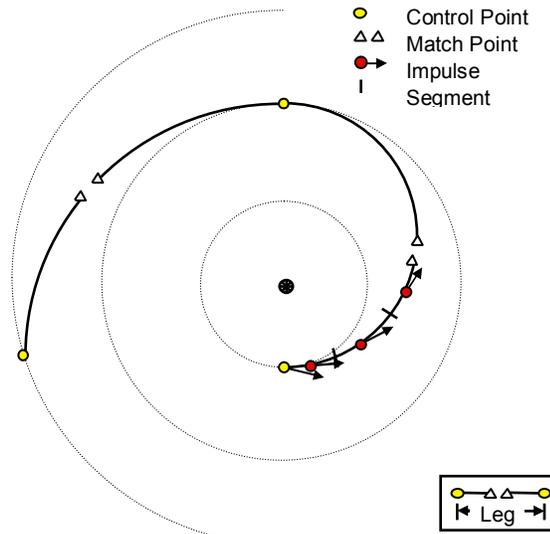
mission. We therefore explore how variations in these four key parameters (launch vehicle, power level, thrusters, and flight time) affect the design of an interplanetary mission. Moreover, the discrete levels of available systems can cause suboptimal performance (e.g. the actual engine thrust may be a lower than desired, thus increasing trajectory ΔV). The effect of discrete hardware choices on SEP system design is also addressed. Results are provided for a mission set comprising a near-Earth asteroid sample return, main belt asteroid rendezvous (à la Dawn), comet rendezvous, comet sample return, and a mission to Saturn.

Key issues to be resolved during preliminary design include selecting a launch vehicle, solar array size, thruster type, and flight time. Usually, the mission target (or short list of potential targets) and spacecraft neutral mass are chosen to satisfy scientific objectives. Neutral mass is defined as any mass that is unaffected by variations in the SEP system, i.e. mass of the payload, spacecraft bus, hydrazine, supporting structure, etc. Therefore the neutral mass allocation is treated more as a constraint than as a free design parameter. (Initial iterations between mission designers and science investigators ensure that the desired neutral mass range and destination set are feasible for the given timeframe and budget of the mission.) The problem then becomes to select a spacecraft system, defined by the four aforementioned parameters, that is cost effective and provides the required neutral mass while being robust to fluctuations in the design. A straightforward solution is to minimize mission cost for a given neutral mass. (The term “cost” always refers to the dollar cost of the mission.) This approach handles the “cost effective” requirement, while examining optimal cost solutions over a range of neutral masses addresses the “robust” requirement. The SEP system designs that emerge from this method are compared with designs that are indicated by mass optimization techniques. The desired result is a low cost design that is able to absorb variations in neutral mass and SEP system performance.

II. Trajectory and System Models

a. Trajectory model

The trajectories are optimized using JPL’s Mission Analysis Low-Thrust Optimizer (MALTO).⁷ The key aspect of MALTO is that it approximates continuous thrusting by a series of impulsive thrust vectors connected by conic arcs as illustrated in Figure 1. The magnitude of thrust on each segment is bounded by the power available, the engine model, and the segment duration. During the optimization MALTO propagates a trajectory forward and backward from control points (e.g. planets), and continuity is enforced by constraining the spacecraft states to be equal at the match points.



J. A. Sims and S.N. Flanagan

Figure 1 Schematic of MALTO trajectory model.⁷

Three different objective functions are used for the trajectory analysis: 1) maximize delivered mass with constrained TOF, P_0 , I_{sp} , and launch vehicle m_0 with free m_p , 2) maximize neutral mass with constrained TOF, I_{sp} , and launch vehicle m_0 with free P_0 and m_p (for given μ_{p0} and μ_p), and 3) minimize flight time with constrained delivered mass, P_0 , I_{sp} , launch vehicle m_0 , and m_p . This last mode is applied once the hardware and spacecraft mass

values have converged. Longer flight times deliver more mass, but cost more for extended operation time. While the exact cost depends on the mission, a TOF cost factor of $\chi_{\text{TOF}} = 10$ M\$/year is used to balance the benefit of additional mass against cost of operations.

b. Optimization of cost and neutral mass

MALTO does not explicitly optimize trajectories for minimum cost (i.e. χ_{TOF} , χ_{P_0} , χ_{m_0} , and χ_p are not optimization parameters), so cost is minimized by a combination of brute force trade studies and a simplex search on TOF, P_0 , and I_{sp} . The launch vehicle m_0 , P_0 , m_p , and m_f are simply scaled to provide a desired neutral mass. This scaling operation is possible because different spacecraft with the same acceleration history will travel along the same trajectory (with the same optimal launch date, C_3 , ΔV , etc.). The launch vehicle model is designed so that the launch mass at a given C_3 is proportional to m_0 , and the thrusters are modeled with constant I_{sp} (so thrust T and mass flow \dot{m} are proportional to P_0), which causes the thrust-to-mass ratio (acceleration) to be consistent. (More accurate launch vehicle and thruster performance curves replace these idealized models once a preliminary design is chosen.) This scaling property is particularly useful for examining a range of neutral masses. For example, if a mission with a specified m_0 , TOF, P_0 , and I_{sp} optimizes to launch mass = m_{C_3} , delivered mass = m_f , and propellant mass = m_p , then the neutral mass is given by Eq. (1)

$$m_N = m_f - \mu_{P_0} P_0 - \mu_p m_p \quad (1)$$

Then, if m_N^* is the desired neutral mass, the design parameters are adjusted by Eq. (2)

$$f = m_N^*/m_N, m_0^* = fm_0, P_0^* = fP_0, m_f^* = fm_f, m_p^* = fm_p \quad (2)$$

The acceleration history is the same because the launch vehicle and thruster models are designed to satisfy Eq. (3)

$$m_{C_3}^* = fm_{C_3}, T^* = fT, \dot{m}^* = f\dot{m} \quad (3)$$

Thus, a spacecraft with any desired neutral mass follows the same trajectory as the original spacecraft design. The mission cost (or at least the aspects of cost affected by the trajectory design) is then minimized by varying TOF, P_0 , and I_{sp} . The objective function is the relative cost, which accounts for variations from an arbitrary baseline.

$$\text{relative cost} = \chi_{m_0} m_0^* + \chi_{P_0} P_0^* + \chi_p m_p^* + \chi_{\text{TOF}} \text{TOF} \quad (4)$$

The terms in Eq. (1) do not account for the total system masses. For example, the power system term $\mu_{P_0} P_0$ only accounts for variations in mass, as the fixed mass does not affect the optimization. Instead any fixed masses are lumped with the neutral mass, which is defined as the mass that is unaffected by changes in the SEP system. So if the power system model is 100 kg + 10 kg/kW, the 100 kg is added to the neutral mass and μ_{P_0} is 10 kg/kW (and not the total specific mass of the system). Similarly, from Eq. (4) it is evident that the outcome of this approach is not to minimize the comprehensive cost of the mission, as there is no accounting for instrument or spacecraft bus costs. Moreover, this expression does not even account for the total cost that is affected by mission design choices. Instead it tracks the delta of varying any of the key mission design parameters, so that a stationary point may be found where any variation in flight time, power or specific impulse results in an increase in cost for a given neutral mass. Thus, just as a set of mass coefficients are readily applicable to maximize mass fractions,^{8,9} an extended parameter set that includes cost coefficients may be applied to the cost optimization problem.

c. Launch Vehicle

The launch vehicle model is designed to approximate the performance of existing launch vehicles, while allowing the launch mass to scale proportionally with m_0 . As demonstrated in Figure 1, the chosen polynomial in C_3 fits a range of launch vehicles and allows free choice of launch mass by selecting m_0 for a given C_3 (which is optimized in MALTO). The launch vehicle curves are published by KSC.¹⁰

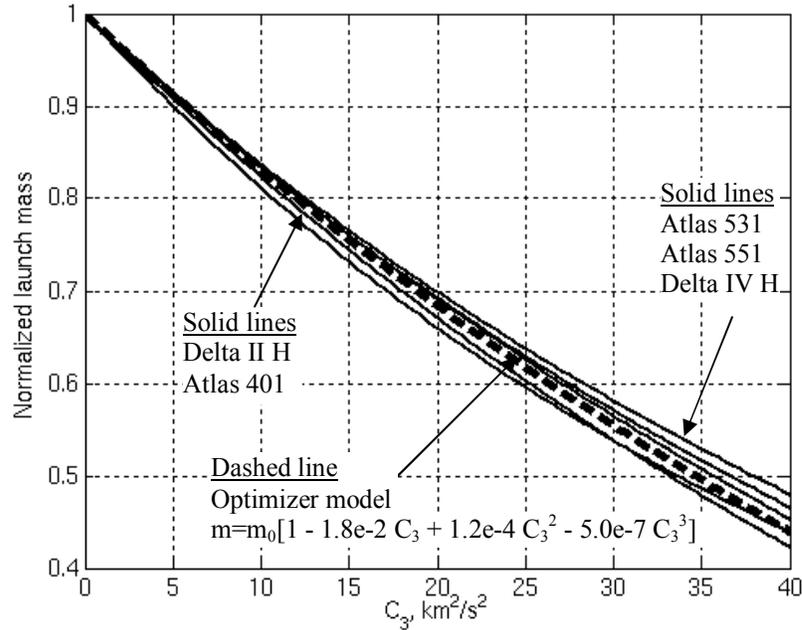


Figure 2 Launch vehicle model approximates actual performance.

While much insight is gained by examining the mission design problem with free choice of launch vehicle size, there are only a handful of launch vehicles that will fit a specific mission. Therefore the design problem is also addressed with a launch vehicle set limited to $m_0 = 1.5, 3.5, 5, 6.5,$ and 9 metric tons. The launch vehicle cost factor χ_{m_0} is assumed to be 20 M\$/t. It is emphasized that this value is not used to determine the absolute cost of a launch vehicle, but rather to estimate the relative costs of switching between launch vehicles.

d. Thrusters and propellant

The thrusters are modeled as an ideal engine, meaning that the thruster operates at any input power, and the efficiency η and specific impulse I_{sp} are assumed to be constant. These assumptions result in a thrust T and mass flow rate \dot{m} that are determined by Eq. (5), where the efficiency is assumed to be 60% , g is the standard acceleration of (Earth's) gravity, and P is the input power to the power processing unit(s) (PPU).

$$T = 2\eta P / g I_{sp}, \quad \dot{m} = 2\eta P / (g I_{sp})^2 \quad (5)$$

A specific impulse range of $1,000\text{--}6,000$ s is used to survey the design space. However, actual thrusters can only throttle over a limited range, so the missions are also examined with an I_{sp} of $1,500\text{--}2,500$ s (for Hall effect thrusters) or $3,500\text{--}4,500$ s (for ion thrusters).

While a single “super engine” provides a convenient model for mapping the design space, the actual propulsion system is comprised of multiple thrusters. It is often assumed that the power level P_0 drives number of required thrusters as extra available thrust is often beneficial. However, for missions to destinations beyond Earth's orbit, most of the actual thrusting is done at a much lower power level than P_0 . Instead, the propellant throughput m_p is a much better indicator to the number of required thrusters. As the engines process propellant, they degrade, and additional thrusters must be available to replace the old ones. This effect suggests that the mass and cost factors for propellant and thrusters may be combined. Table 1 provides a breakdown of the estimated mass and cost associated with the thrusters and propellant. It is stressed that these mass and cost values do not reflect the actual mass or cost of the system, but instead account for variations within the system (i.e. relative comparisons).

Table 1 Mass and cost factors for thrusters and propellant

Component	Mass	Cost	Notes
Thruster	30 kg	3 M\$	Typical 5 kW thruster. Higher power limit involves additional mass and cost. Includes gimbal, etc.
Throughput	300 kg of m_p	—	Typical 5 kW thruster. Higher power limit or Hall may process more xenon.
Propellant	—	1 M\$/t of m_p	Cost of xenon.
Tankage	100 kg/t of m_p	1 M\$/t of m_p	Includes support structure.
Total	$\mu_p = 200$ kg/t	$\chi_p = 12$ M\$/t	

e. Power and processing units

The solar array model is a simple inverse square curve, and does not account for any thermal efficiencies or radiation inefficiencies. In Eq. (6) P , is the power out of the solar arrays, P_0 is the array power at 1 AU, and r is the radial distance from the sun.

$$P = P_0 / r^2 \quad (6)$$

Spacecraft power is ignored, and for the purposes of this sizing exercise only SEP power into the power processing units (PPUs) is included in equation 6. The primary reason spacecraft power is omitted is to ease the scaling of the power system with launch vehicle and spacecraft mass [as in Eq. (2)]. If the spacecraft is powered by the arrays (as opposed to an RTG), then P_0 increases by one to a few kW depending on how far from the sun the spacecraft operates. As with the launch vehicle model, it is informative to examine the design space with free choice of P_0 , however solar arrays are not typically built to any desired power level (arrays could be sized to the nearest one or two kW). Moreover, it is often economical to incorporate a standard array, say from a previous mission or one developed for another spacecraft. The solar array size is therefore limited to power levels of 5, 10, 15, 20, and 25 kW to examine the implications of non-optimal P_0 . These power levels would be increased to account for spacecraft power drawn from the array.

While the propellant throughput can provide a rough estimate of the number of thrusters, the optimal power level P_0 provides an estimate of the number of PPUs. It is usually not necessary to have enough PPUs to process all of the available power at 1 AU, but designs that converge to higher power levels typically require additional PPUs to operate additional thrusters. Moreover, a single PPU can cross strap to multiple (or at least a couple) thrusters, partially decoupling the number of PPUs to the number of thrusters. In this sense, the PPU cost and mass factors may be book kept with the power factors. The key here is not to determine the absolute number of PPUs to be used (which can be chosen in subsequent design iterations), but rather to keep track of the relative mass and cost of the system for different designs (e.g. a 15 kW array is likely to use more PPUs than a 5 kW array).

Table 2 Mass and cost factors for Solar Array and PPU

Component	Mass	Cost	Notes
Solar Array	10 kg/kW	1 M\$/kW	Relative values—does not include fixed mass or cost.
PPU	30 kg	3 M\$	Typical for 5 kW thrusters. Higher power limit involves additional mass and cost.
Total	$\mu_{p0} = 16$ kg/kW	$\chi_{p0} = 1.6$ M\$/kW	

f. Summary

The coefficients used to track changes in neutral mass and mission cost are summarized in Table 3. There are constant mass and cost values associated with these coefficients that are essentially added to the neutral mass values. For example, the solar array mass may follow $100 \text{ kg} + 10 \text{ kg/kW}$ (so a 10 kW array would have a specific mass of 20 kg/kW). The 100 kg is simply added to the desired neutral mass (which includes non-SEP components such as

the payload and bus) and the 10 kg/kW is used to track mass variations due to P_0 among different designs as in Eq. (1). Further, if the array cost follows 5 M\$ + 1 M\$/kW the 5 M\$ is tacked onto the fixed neutral mass cost (e.g. cost for scientific instruments) and the 1 M\$/kW is used to optimize the design to a minimum cost solution using Eq. (4). These values do not necessarily reflect actual mass and cost parameters, but rather serve to illustrate a method of minimizing cost for an SEP mission.

Table 3 Mass and cost parameters used to select SEP systems

Component	Mass	Cost	Notes
Operations	—	$\chi_{\text{TOF}} = 10 \text{ M\$/yr.}$	—
Launch Vehicle	—	$\chi_{m_0} = 20 \text{ M\$/t}$	Also set to $m_0 = 1.5, 3.5, 5, 6.5, \text{ or } 9 \text{ t}$
Propellant	$\mu_p = 200 \text{ kg/t}$	$\chi_p = 12 \text{ M\$/t}$	I_{sp} also limited to 1,500–2,500 or 3,500–4,500 s
Power	$\mu_{P_0} = 16 \text{ kg/kW}$	$\chi_{P_0} = 1.6 \text{ M\$/kW}$	Also set to $P_0 = 5, 10, 15, 20, \text{ or } 25 \text{ kW}$

III. Mission Descriptions

Optimal SEP systems can vary significantly for different missions; therefore, it is prudent to include a range of destinations and spacecraft masses when addressing the general problem of designing SEP missions. Though the present mission set presented here is certainly not comprehensive (a target within Earth’s orbit is a notable omission), it does include a variety of mission characteristics including (among others) relatively close and more distant destinations, rendezvous and sample return, short and long flight times, small and large ΔV , and compact and massive spacecraft. The trajectories in Figure 3–Figure 8, illustrate the general characteristics for each mission, but do not reflect any actual design (i.e. the launch vehicle, flight time, power, and specific impulse are arbitrary). Indeed, the theme is a process to modify suboptimal, potentially infeasible designs (e.g. Figure 3–Figure 8) into cost-effective, robust missions. In the following figures, the spacecraft path is the solid line, encounter body orbits are dotted lines, thrusting is noted by arrows, and encounters (launch, flyby, etc.) are circles.

a. Near-Earth asteroid sample return

The destination for this mission is the asteroid 1989 ML, which is the closest target to Earth in the set. Near-Earth asteroid missions typically involve lower power levels, because the spacecraft is never too far from the sun. The neutral mass for this mission is 1600 kg, which is larger than a rendezvous spacecraft, but smaller than an outer-planet mission. Thus, there is an interesting combination of low ΔV with moderate mass. The mission duration is around 3 years with a minimum 90-day stay time.

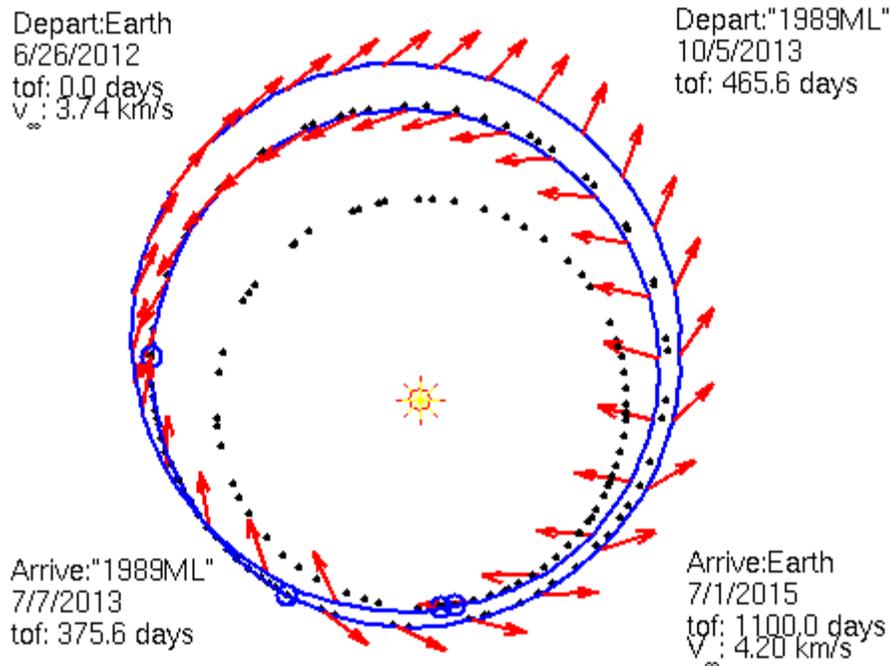


Figure 3 Encounter bodies and times with spacecraft trajectory for asteroid sample return mission.

b. Main belt asteroid rendezvous

The body sequence for this mission includes a Mars flyby, Vesta rendezvous and departure, and Ceres rendezvous (same path as Dawn). Because it is a rendezvous mission the neutral mass is 800 kg (no need for return hardware as with a sample return), but the ΔV is large (10–14 km/s) because there are two rendezvous. So, this mission is an example of a small spacecraft combined with a large ΔV . It also provides an example with multiple destinations. Missions to the main belt often allow a wide range of flight times (with different heliocentric revolutions), so a TOF range of 4–10 years is examined. The minimum stay time at Vesta is 270 days.

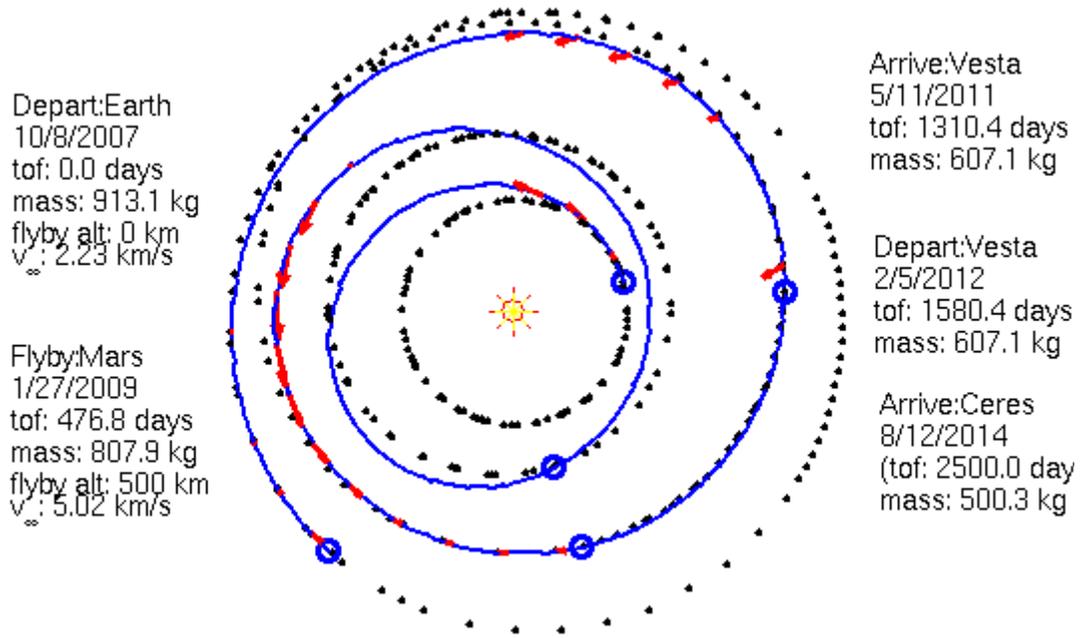


Figure 4 Encounter bodies and times with spacecraft trajectory for asteroid rendezvous mission.

c. Comet rendezvous

The rendezvous target is Tempel 1, which is a Jupiter-family comet (aphelion is near Jupiter's orbit). The neutral mass is the same as the asteroid rendezvous example, 800 kg. This mission provides an example of a small spacecraft with a moderate ΔV target. Rendezvous usually occurs within a year after perihelion, and the range at rendezvous is between 1.5–3 AU. This mission is also notable because the spacecraft thrusts almost the entire way to the comet.

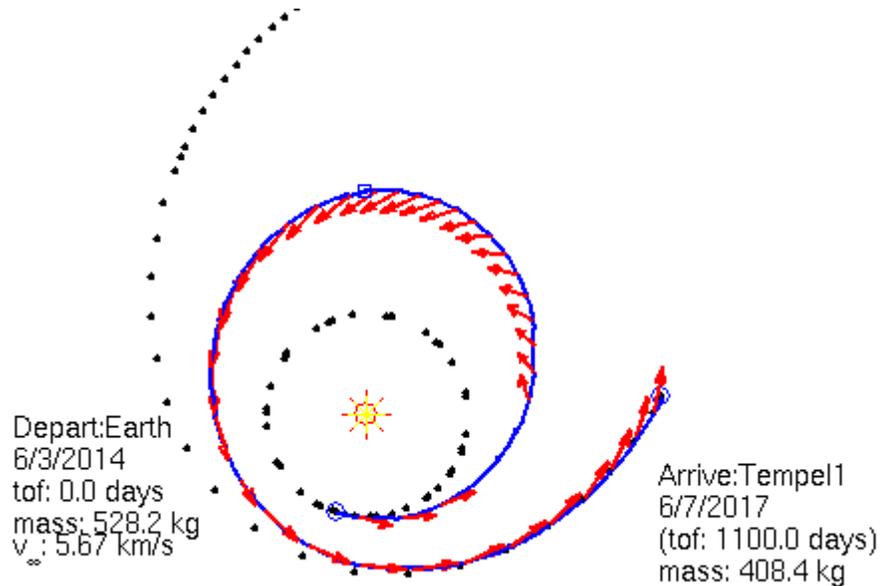


Figure 5 Encounter bodies and times with spacecraft trajectory for comet rendezvous mission.

d. Comet sample return

The target for this example is another Jupiter-family comet, Tuttle-Giacobini-Kresak. This mission combines a moderate sized spacecraft (neutral mass for sample returns assumed to be 1600 kg), with a large ΔV . Comet sample returns stress an SEP system because the spacecraft must often thrust at far distances from the sun (up to 4 AU) to depart the comet. As a result these missions typically involve the largest solar arrays. Also entry speeds at Earth can become prohibitively high, so the arrival V_{∞} is limited to 9 km/s. This constraint is almost always active. The mission duration is around 8 years with a minimum 180-day stay at the comet.

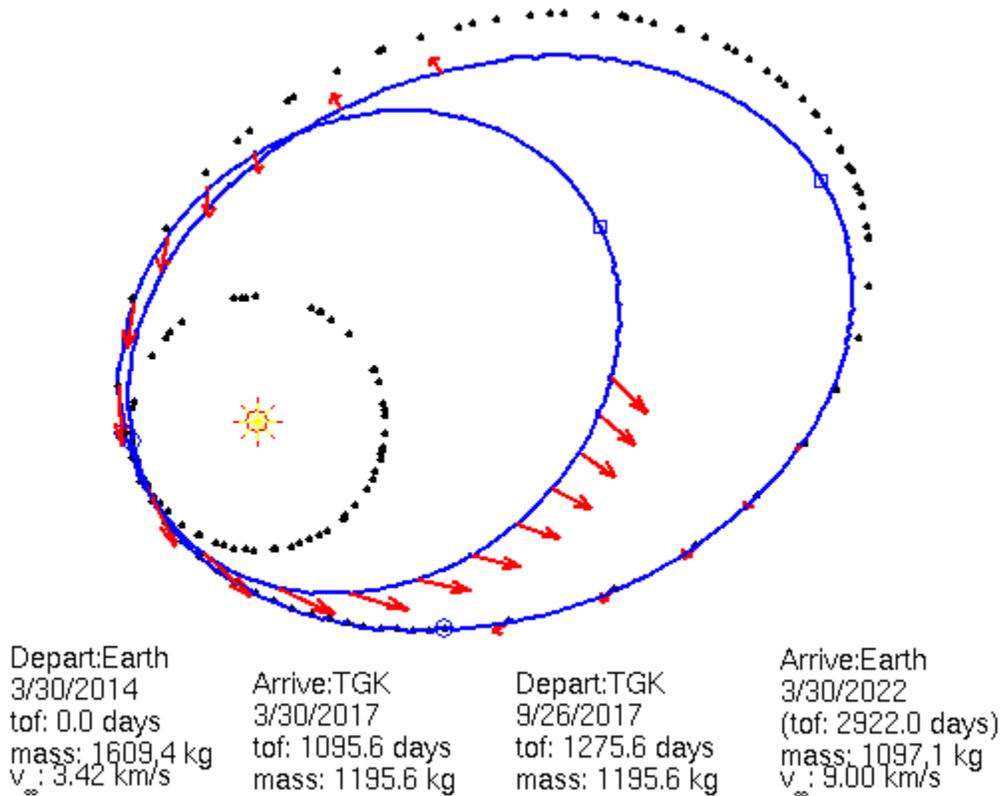


Figure 6 Encounter bodies and times with spacecraft trajectory for comet sample return mission.

e. Outer planet

The final mission in the set is to an outer planet, Saturn. Because of Saturn's great distance from the sun (9–10 AU) an SEP stage is impractical for orbit insertion. Instead, the means of rendezvous at Saturn is not specified (e.g. chemical stage or aerocapture), but the arrival V_{∞} is limited to 9 km/s. This constraint becomes active at shorter flight times. Most of the ΔV to reach Saturn is achieved by gravity assists, making this example unique in the set. (The body sequence is Earth-Earth-Venus-Venus-Earth-Saturn.) Moreover, while the destination is distant, most of the thrusting occurs near Earth's orbit (.7–2 AU), so this mission also provides an example where most of the ΔV is imparted near the sun. A neutral mass of 4500 kg is chosen to reflect a flagship class mission, providing an example of a massive spacecraft with small propulsive ΔV and large gravity-assist ΔV .

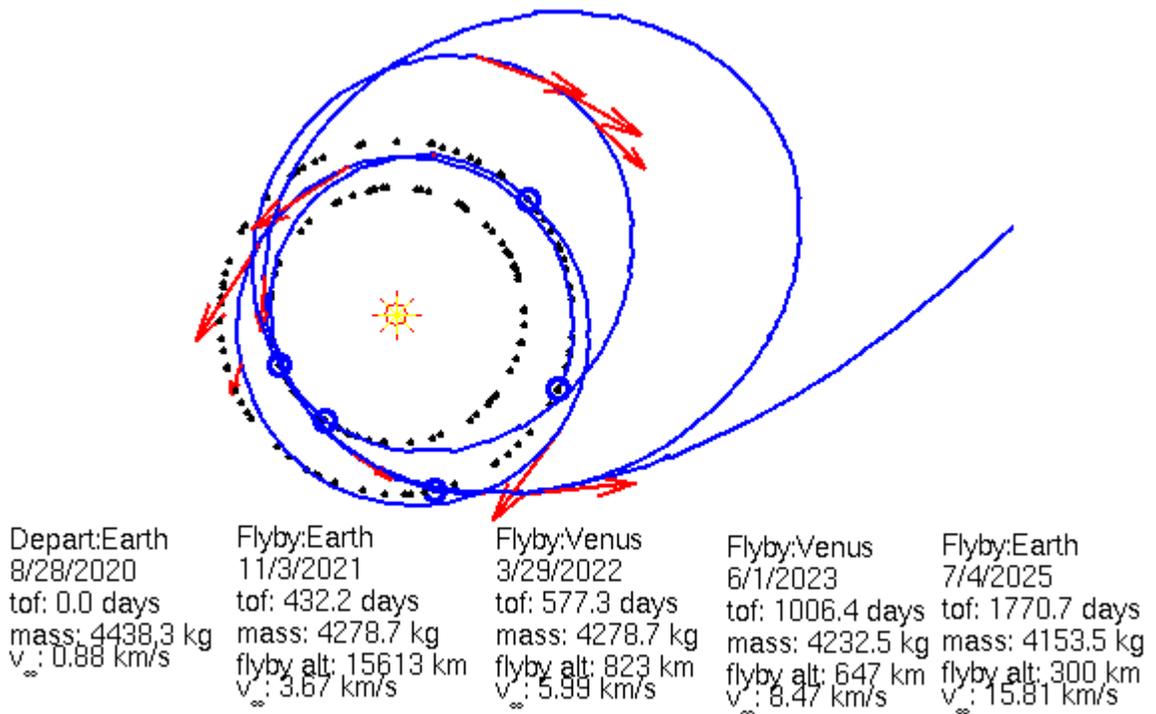


Figure 7 Flyby sequence and zoom on inner-planet portion of trajectory for mission to Saturn.

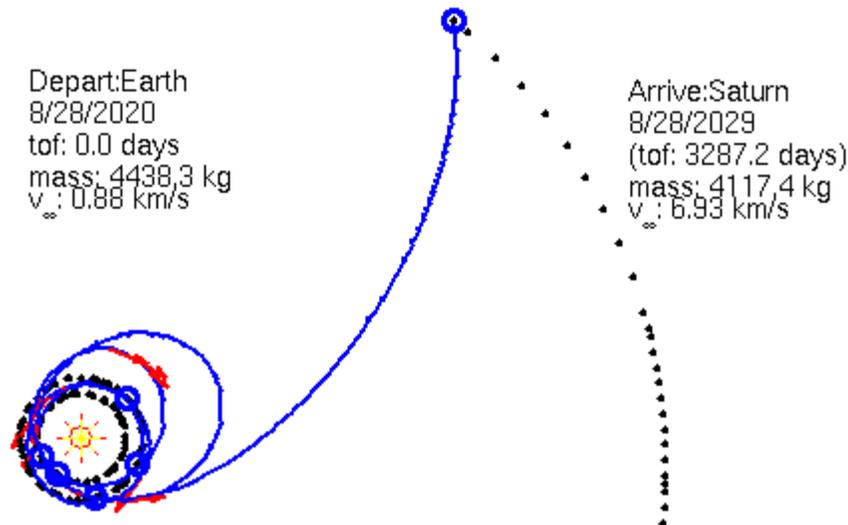


Figure 8 Spacecraft trajectory for mission to Saturn.

f. Summary

A list of pertinent characteristics of the mission set are combined in Table 4. The neutral masses do not reflect any actual spacecraft, but instead serve to offer a variety of SEP system designs.

Table 4 Summary of mission parameters

Mission	Destination	Neutral mass	TOF	ΔV	Thrust distance	Spacecraft distance
Asteroid sample return	1989 ML	1600 kg	~3 yr.	~3 km/s	1–1.6 AU	1–1.6 AU
Asteroid rendezvous	Vesta and Ceres	800 kg	4–10 yr.	10–14 km/s	1–2.5 AU	1–3 AU
Comet rendezvous	Tempel 1	800 kg	2.5–6 yr.	~8 km/s	1–3 AU	1–4 AU
Comet sample return	TGK	1600 kg	~8 yr.	12–16 km/s	1–4 AU	1–5 AU
Outer planet	Saturn	4500 kg	7.5–11 yr.	2–5 km/s	0.7–2 AU	0.7–10 AU

IV. SEP System Selection

When a mission is first proposed, the design space is wide. The study typically begins with a target of scientific interest and method of investigating the target. Major design choices such as launch vehicle, solar array power, number of thrusters, and engine type remain to be determined. (The propellant load is also a design variable, but it is largely a function of the selected specific impulse.) The required neutral (non-SEP dependent) mass of the spacecraft usually converges quickly based on the instrument package, and a cursory trajectory search can determine if the desired target is attainable in the timeframe of interest. From this initial search (e.g. beginning with the trajectories in Figure 3–Figure 8), the design team must find a system configuration that can reliably deliver the neutral mass to the target, and that is unlikely to require dramatic changes if the spacecraft mass fluctuates. This problem is twofold: first, how does one arrive at a robust design; then, how does one determine if the selected system is better than all of the other robust designs? Because of the multi-faceted nature of SEP missions, there are myriad solutions that provide the same mass margin (a key factor for mission robustness), and established methods that point the way to robust chemical systems (e.g. minimize ΔV) are insufficient to select one SEP design from another. For example, maximizing the delivered mass fraction is useful for selecting a launch vehicle and thruster specific impulse, but provides little guidance to size the power and propulsion system. Neutral mass optimization accounts for the power and propulsion system, but leaves the flight time as a free parameter. The proposed approach is to depart from mass optimizations and use dollar cost to unite the selection of launch vehicle, power and propulsion system, and flight time into a single figure of merit.

While this cost approach arrives at designs that balance the launch vehicle, SEP system, and flight time it does not necessarily ensure robust designs. For example, the cost optimization may arrive at a design point that is very near the maximum neutral mass fraction so that the spacecraft can fit on a small (and inexpensive) launch vehicle. But, if the payload or hardware mass increases, the launch mass may exceed the capability of the launch vehicle and a new design is required. One way to track the system sensitivity to spacecraft mass is to create a set of minimum cost solutions over a range of neutral masses. The increased margin on neutral mass also absorbs increases in hardware mass. If a system is designed for a neutral mass of 1,000 kg but the current neutral mass estimate is 900 kg, then a 100 kg increase in payload or a 100 kg increase in propulsion system mass are absorbed equally into the 1,000 kg allocation. In this way the minimum cost solution for any desired mass margin is known.

The SEP system selection process centers on the minimum cost performance and associated designs shown in Figure 9–Figure 18, where the neutral mass is varies by $\pm 30\%$ for each mission. The bold curve (also denoted by the \circ symbol) in these figures correspond to the minimum cost solution with discrete choice of launch vehicle, array size, and specific impulse (at the levels specified in Table 3). These results are compared with the continuous parameter set (e.g. free choice of launch vehicle size) and with solutions obtained by maximizing neutral mass fraction m_N/m_0 and delivered mass fraction m_f/m_0 . The discrete m_N/m_0 curves are denoted by a ∇ , while the discrete m_f/m_0 optimizations are denoted by a Δ , so when they happen to be equal they appear as a hexagram, and the continuous m_N/m_0 curves are denoted by a \times , while the continuous m_f/m_0 optimizations are denoted by a $+$, so they appear as an asterisk when they are equal. (There is nothing particularly special when the mass fraction curves are equal, but this is a common occurrence so the chosen symbols are a visual aid.)

a. Near-Earth asteroid sample return

As indicated by the flat bold curve in Figure 9, a variation in neutral mass allocation does not strongly affect the cost of this mission. From Figure 10, the launch vehicle, power system, and flight time are constant near the target neutral mass of 1600 kg. (It is important to note here that the launch vehicle and P_0 values in Figure 10 are the minimum required values to achieve the desired neutral mass. The actual launch vehicle m_0 is 3.5 t and the solar array is sized to 5 kW as specified in Table 3). The specific impulse curve indicates how the system is allocating more mass. The I_{sp} steadily drops from 4,500 s at 1,600 kg to 3,500 s at 2,000 kg, so it appears that the cost optimal mechanism to increase neutral mass is to burn more propellant to complete the mission. As noted in Table 4 the ΔV for this mission is quite low, so changes in the design should have little impact on propellant mass. The impact on cost appears in the very slight slope of the bold cost curve in Figure 9. Thus the SEP system could be designed to a neutral mass of 2,000 kg, providing plenty of margin with almost no impact on cost. The propellant load and tanks would be significantly (in a relative sense) larger than the 1,600 kg neutral mass design, but the increase in system mass is implicitly accounted for by the combination of systems in Figure 10.

If a neutral mass method were used instead of the cost optimization, the resulting system configuration would cost about 10 M\$ more (∇ curve in Figure 9). The difference in cost arises from the mass optimal configuration converging to a solution with more power. The high power solutions found via mass optimization is essentially used to build launch vehicle margin into the system as opposed to reducing cost. The launch vehicle subplot in Figure 10 shows that the spacecraft only uses a fraction of the launch vehicles capability (of 3.5 t at escape). If this margin is deemed excessive, the spacecraft mass can scale up [e.g. according to the relations in Eq. (2) and Eq. (3)] to provide more neutral mass. For example, there is plenty of launch vehicle margin to scale the 1,600 kg solution with maximum neutral mass fraction to provide 2,000 kg neutral mass (an increase of 25%). But, the power and propellant mass also increase, adding to the already excessive cost of the design. Instead, if a 25% margin on neutral mass is desired, then the cost-optimal solution that allocates 2,000 kg for neutral mass would be more appropriate.

Moreover, the flight time is constrained to be equal to the cost optimal TOF to make an even comparison. If the TOF is allowed to vary, the mass optimizations tend to increase the mission duration and cost of operations, which usually leads to an even bigger difference in cost. An additional trade study on flight time (e.g. maximizing neutral mass over a range of flight time), would provide some insight to help choose the right flight time, but the resulting SEP system is still suboptimal in terms of cost. Moreover, the cost optimization approach implicitly finds a good flight time without the need of additional trade studies (which can be conducted in later design iterations *after* a cost optimal solution is found as a baseline).

While the chosen search method has a significant effect on which combination of systems is deemed optimal, the limited set of hardware options can have an even larger effect on mission cost. As indicated by the continuous system curve (denoted by \bullet) in Figure 9 the mission could save about 30M\$ if the optimal configuration were available. (The \bullet point is 30 M\$ lower than the \circ point at the nominal mass of 1600 kg.) From Figure 10, the optimal launch vehicle size has a m_0 of around 2 t, and at 20 M\$/t would cost about 30 M\$ less than the available 3.5 t vehicle. The relatively low ΔV of this mission leads to little overhead in terms of launch vehicle and array size for the optimal design. But, with the launch vehicle cost set at the higher price, the cost optimization method automatically finds solutions that make use of the “excess” launch vehicle capability by keeping the power system small and reducing flight time slightly. The mass optimizations tend to converge to solutions with high power and high specific impulse, which suggests that the propellant mass fraction is a dominating term for the mass optimization, and which also accounts for the difference in cost of about 5 M\$ for neutral mass optimization and about 20 M\$ for delivered mass optimization. It is interesting to note that the optimal continuous specific impulse is lower than the discrete value while the I_{sp} decreases to the maximum available value (4,500 s) for the mass optimizations. (The I_{sp} in Figure 10 is actually at 4,400 s because 200 s increments were used in the MALTO runs. However, small changes in large I_{sp} values have little impact on spacecraft mass, and the 100 s difference could be considered as a margin on thruster performance.) High power solutions also tend to converge to higher I_{sp} , and in this case the cost optimization increased power to the 5 kW allotment and converged to a higher specific impulse solution.

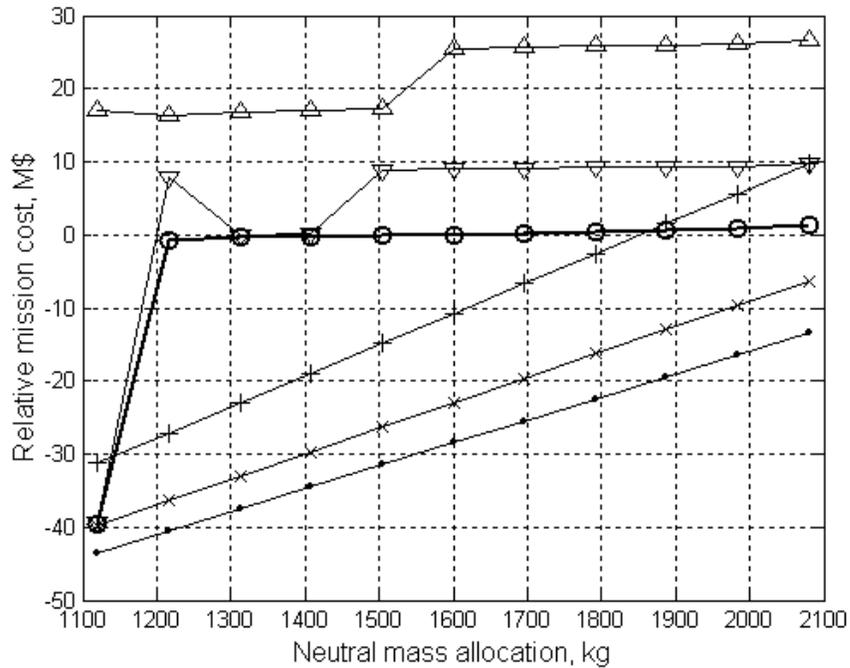


Figure 9 Mission cost for asteroid sample return. For discrete system levels $\circ = \text{cost}$, $\nabla = m_N/m_0$, and $\Delta = m_T/m_0$ optimizations, while $\bullet = \text{cost}$, $\times = m_N/m_0$, and $+= m_T/m_0$ solutions with continuous systems.

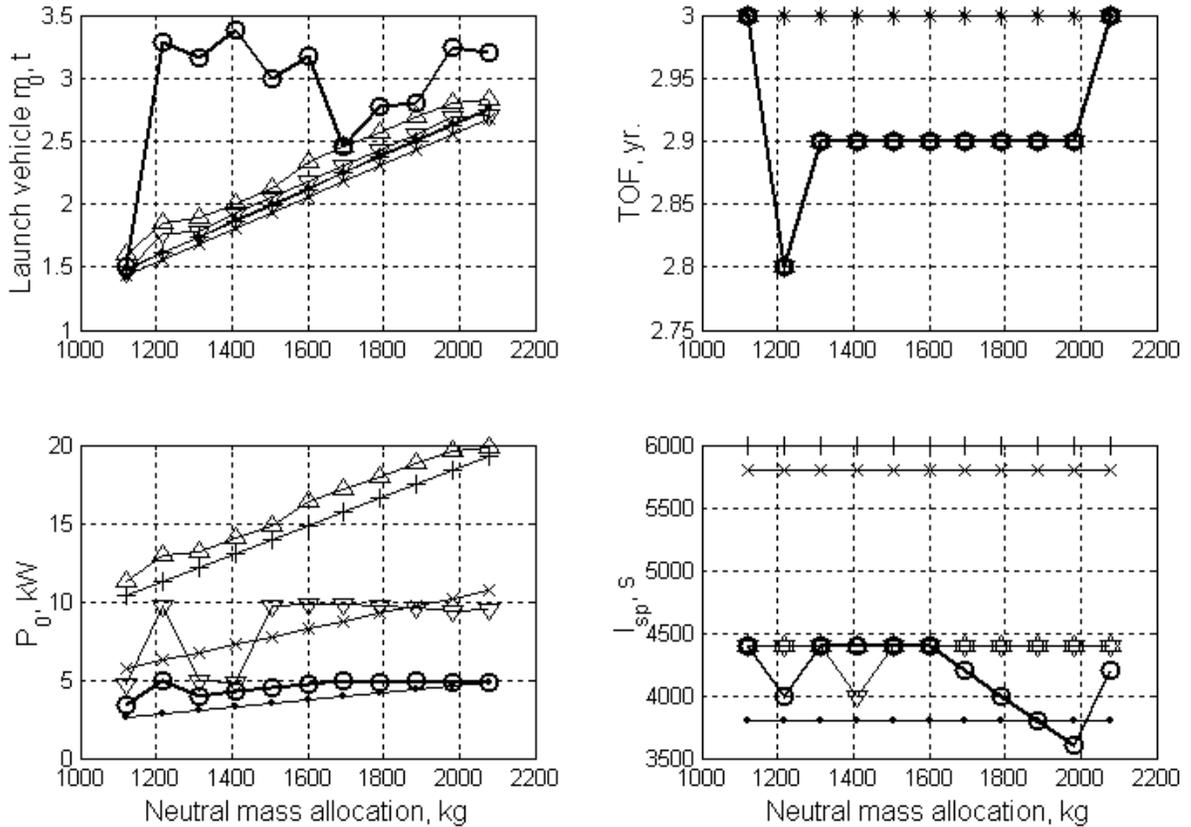


Figure 10 SEP system designs corresponding to Figure 9.

b. Main belt asteroid rendezvous

This mission exhibits a stronger correlation between neutral mass and mission cost than the asteroid sample return mission. The mission cost increases consistently up to about 900 kg neutral mass (800 kg is the design requirement). From Figure 12, the 15 M\$ jump in cost in the bold discrete system curve occurs because the launch vehicle switches from the 1.5 t to the 3.5 t model. Up to this point, the increase in neutral mass is handled by increasing flight time. Then when the mass becomes large enough, the cost optimal configuration switches from small launch vehicle with long flight time, to a large launch vehicle with short flight time. The 40 M\$ price tag for switching to the larger launch vehicle is mostly offset by the dramatic reduction in flight time. Also, the power system increases and the thruster I_{sp} decreases to provide more thrust to accelerate the larger vehicle on the short TOF trajectory. It is also noteworthy that there are 10 yr. TOF trajectories that allow more than 900 kg of neutral mass to launch on the small vehicle, but the cost optimal solution is to switch to short flight times with a larger launch capability. A similar trade occurs on the lower end of the neutral mass scale. Between 600 and 700 kg a 5 kW increase in power handles the mass increase, then from 700 to 800 kg the flight time is increased and the power drops back down to 10 kW. In this way, a cost optimization aids the decision on when it is more beneficial to increase power, flight time, or launch vehicle to build margin into the system. Since the SEP system configuration at the nominal 800 kg can absorb at least a 100 kg increase in neutral mass without redesigning the hardware, the 800 kg configuration is selected for this mission. This SEP system demonstrates reliability by handling mass growth with flight time margin.

The continuous system solution also increases flight time to absorb mass increases, but in this case the impact on cost is not as dramatic because the launch vehicle and power system adjust smoothly to balance against the change in TOF. The cost optimal SEP solution is to increase launch vehicle and power until the neutral mass reaches about 850 kg, then the launch and power systems go back to a smaller size once the flight time begins to rise. Thus the optimal cost curve in Figure 11 remains nearly linear across this boundary.

The cost implications of requiring discrete launch vehicle and solar array sizes is less pronounced for this mission than with the asteroid sample return because the optimal launch vehicle size (1–2 t) is close to the available size (1.5 t). In fact, when the optimal m_0 equals the available m_0 , the choice of optimization technique (discrete vs. continuous or cost versus mass fraction) has a smaller effect on the resulting SEP system. For continuous systems the optimal m_0 crosses the available m_0 between 650 kg and 700 kg neutral mass, then the optimal discrete m_0 converges with the available m_0 at 900 kg where the launch vehicle switch occurs. These points stand out in Figure 11, where the neutral mass solution converges with the optimal cost solution (\circ lines up with ∇). Moreover, all six curves are closest at around 650 kg, where the optimal m_0 corresponds with the available m_0 . The outlier is the optimal delivered mass solution, which has significantly higher cost across the neutral mass range. From Figure 12 it is apparent that a primary cause for the additional cost is high power levels. For a given launch vehicle, the largest delivered mass is attained by increasing power (without a power limit the thrust continues to increase and the maneuvers become impulsive). The concept of neutral mass optimization was introduced to account for the mass of the power and propulsion system while maximizing payload allocation. In this way the SEP system can be optimized to allocate more neutral mass, as indicated by the lower power levels (∇ vs. Δ) in Figure 12. However, for a given neutral mass, optimizing the neutral mass fraction leads to the smallest launch vehicle solution, which automatically maximizes mass margin without a built in control (30% margin could be desirable while 100% is overkill). Instead the extra margin could be “paid back” in reduced mission cost. So, just as delivered mass optimizations can lead to an underutilization of the spacecraft’s ability to maximize payload,⁹ neutral mass optimizations can lead to an underutilization of the launch vehicle’s ability to reduce mission cost.

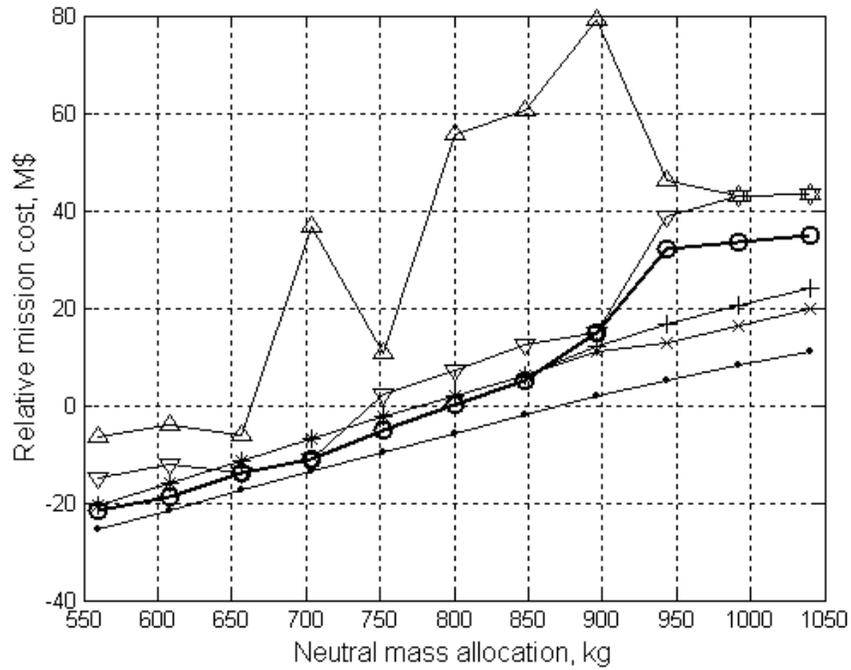


Figure 11 Mission cost for asteroid rendezvous. For discrete system levels \circ = cost, ∇ = m_N/m_0 , and Δ = m_T/m_0 optimizations, while \bullet = cost, \times = m_N/m_0 , and $+$ = m_N/m_0 solutions with continuous systems.

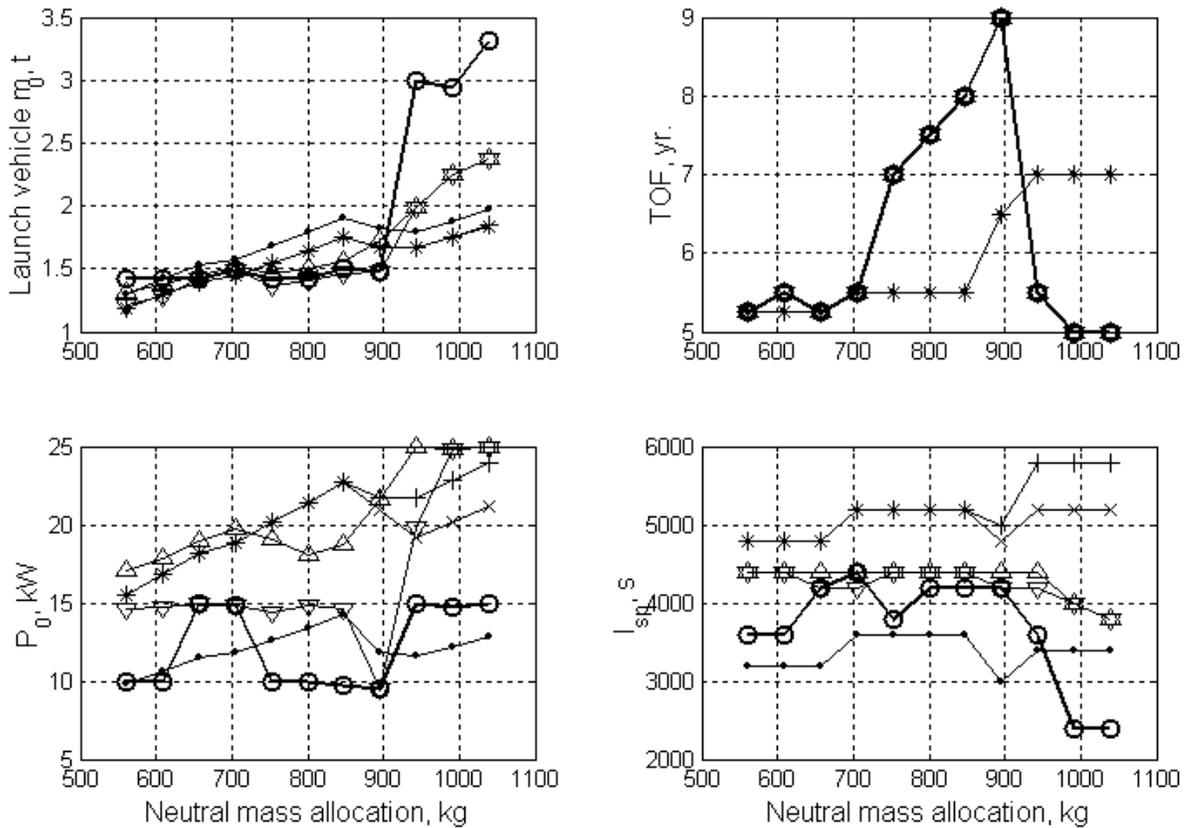


Figure 12 SEP system designs corresponding to Figure 11.

c. Comet rendezvous

This mission presents an interesting case where the cost optimal system has a dramatic shift almost right at the nominal neutral mass design point. In Figure 13, the bold optimal cost curve is relatively flat for neutral masses that are much more or much less than the nominal value. Figure 14 shows the design in transition at 800 kg neutral mass. Beginning at 550 kg the system tends to continue increasing power to account for the increasing neutral mass. Then around 800 kg the flight time must increase dramatically to accommodate the large neutral mass on the small launch vehicle. For larger spacecraft it is cost efficient to reduce the flight time around 2.5 years with more thrust (lower I_{sp}) on a larger launch vehicle. The mass fraction is dramatically increased at this transition as the trajectory switches from low ΔV with high I_{sp} to large ΔV with low I_{sp} , thus making full use of the launch vehicle capability. Since the neutral mass tends to increase as the design matures, the selected SEP system corresponds to the cost-optimal solution for large neutral masses. This case has the unique property of requiring a lot of cost for a little mass margin (e.g. going from 800kg to 850 kg), but then large increases in mass (e.g. going from 850 kg to 1050 kg) may be accommodated with relatively little cost.

The robust combination of low flight time with a large launch vehicle is recognized as a cost effective solution when examined in terms of optimal cost versus neutral mass. For example, if the initial design point is a neutral mass of 750 kg, a mission on a small launch vehicle with short flight time would appear to be ideal. However, when this design hits a wall at 800 kg and solutions with the larger launch vehicle are required, all of a sudden the design is a lot more expensive and provides a lot more mass. It would seem that a design step was skipped between the inexpensive design with slightly lower mass and the overkill design of providing a spacecraft mass that is more appropriate for a different mission class. One way to cut cost is to decrease power and flight time by trial and error until a satisfactory system is found, but it is difficult to tell if a design at 5 kW and 3 year TOF is better or worse than a design at 10 kW and 2.5 year TOF. Optimizing the neutral mass can lower the mission cost by 15 M\$ by decreasing the allocation from 1,000 to 900 kg (as demonstrated by the ∇ curve in Figure 13), suggesting that reducing the power is the way to go (corresponding P_0 curve in Figure 14). On the other hand, the cost optimization approach automatically finds a low-cost combination of power and flight time (and propulsion system) without the need to guess (or intuit) and check a series of intermediate points. Moreover a single trade study on neutral mass shows that the cost optimal solution that provides 1,000 kg dry mass is only a couple of M\$ more than the 900 kg system. Thus, a system that provides “excessive” margin does not necessarily correlate to excessive cost, as it is only slightly more expensive than SEP systems that provide more reasonable margins.

This cost-optimal design point with a 1,000 kg neutral mass allocation is significantly different than the five other “optimal” designs that provide the same mass margin in Figure 14. Specifically the launch vehicle is much larger and the power and specific impulse are lower than the other solutions. Indeed, the cost optimal designs are usually distinct from the mass optimal solutions, which are comparatively similar to each other. This trend is most evident in the continuous system curves, where the designs are not pushed towards any predetermined limits. For example, the delivered mass and neutral mass launch vehicle m_0 and specific impulse are nearly identical, and the mass optimal power levels are about double those of the cost optimal design. This behavior is also seen in the asteroid rendezvous mission (Figure 12), while the asteroid sample return mission (Figure 10) provides an exception. A basic purpose to optimize neutral mass is to find solutions that are fundamentally different than optimal delivered mass designs. However, for the mass parameters in Table 3, the two solutions are usually similar as the power and propulsion mass coefficients do not sufficiently differentiate the designs. A distinguishing characteristic of the cost approach is that there are more parameters that affect the optimization, which leads to distinct SEP system designs. In this way, the optimal solution space (i.e. the locus of designs found by the optimizer) is broadened, reducing the number of trade studies required to arrive at cost-effective solutions.

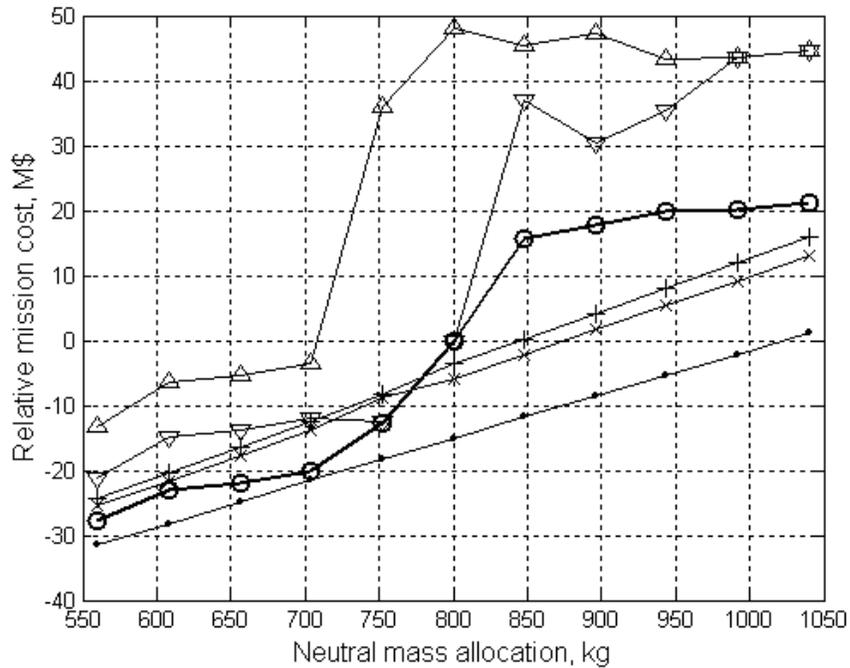


Figure 13 Mission cost for comet rendezvous. For discrete system levels \circ = cost, ∇ = m_N/m_0 , and Δ = m_P/m_0 optimizations, while \bullet = cost, \times = m_N/m_0 , and $+$ = m_N/m_0 solutions with continuous systems.

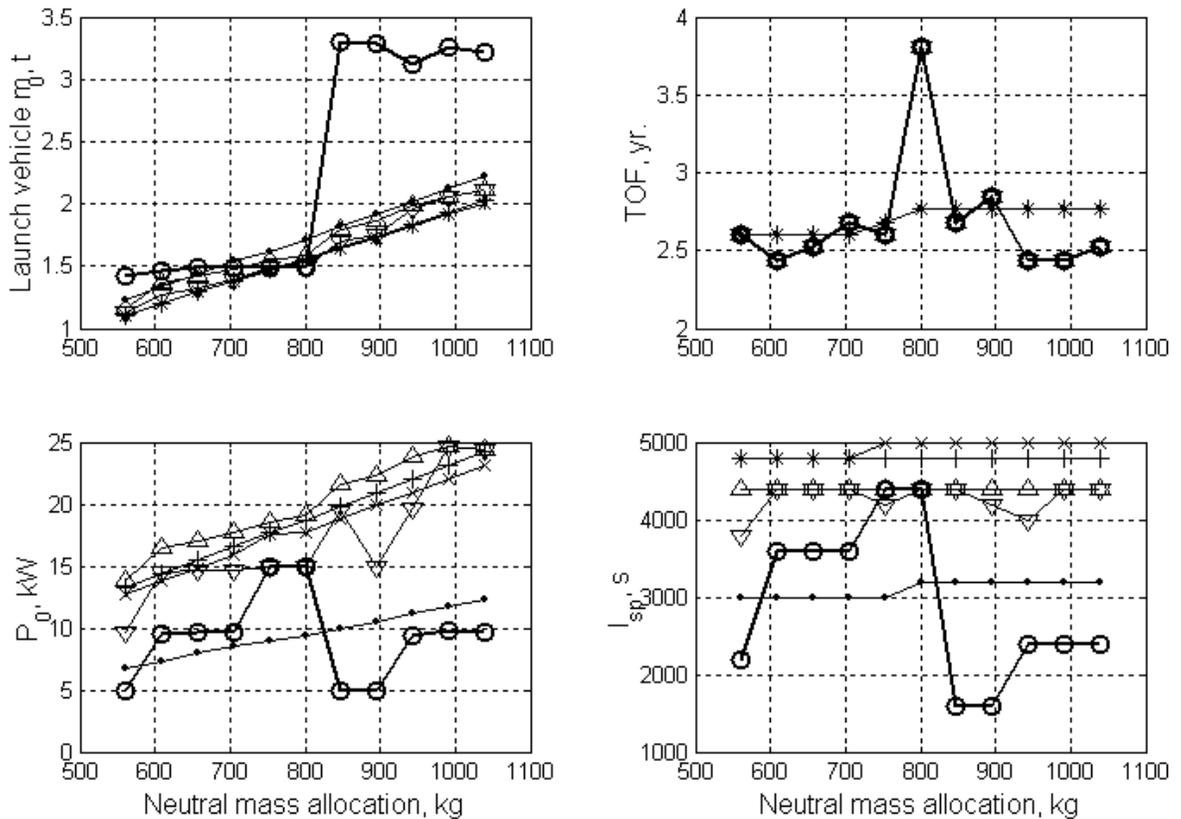


Figure 14 SEP system designs corresponding to Figure 13.

d. Comet sample return

The cost optimal design for this mission requires a dramatic shift in system parameters for any mass growth above the nominal 1600 kg neutral mass allocation. Just as with the comet rendezvous mission, increases in neutral mass are most easily accommodated by switching to a larger launch vehicle. But unlike the comet rendezvous example, the shift in cost for the discrete parameter set (bold line in Figure 15) does not level off after the launch vehicle change. Instead the mission cost rises proportionally with the neutral mass over a wide mass range. The cost-optimal design choices that allocate increasing neutral mass may be inferred from Figure 16. First, a 9 month jump in TOF provides room to fit the 1600 kg nominal mass on the smaller launch vehicle. Then, the design shifts to a bigger launch vehicle at 1700 kg; the power rises at 1800 kg; the propellant load increases (I_{sp} decreases) at 1900 kg; then the cycle is repeated by increasing TOF again at 2000 kg neutral mass. So, for this mission there is no single mechanism that consistently allocates larger masses, which helps explain why the cost does not level off at the launch vehicle switch. Since it appears that every 100 kg increase in neutral mass corresponds to a 10 M\$ increase in cost, the selected SEP system is very much a function of the desired mass margin. In this case, a 100 kg margin is assumed, placing the system on a larger launch vehicle with 20 kW of power and high specific impulse for the SEP system. The optimal TOF for this system is 8.0 years, which places the design right in the middle of the flight time range (noted in Table 4), so lowering TOF could help reduce cost at the nominal 1600 kg point, while there is room to increase the TOF to accommodate masses beyond 1700 kg (to an extent).

This mission is unique in that the optimal discrete system solution lines up with the optimal continuous parameter solution at the nominal point. From the \bullet curves in Figure 16 the cost optimal m_0 is around 3.5 t, the optimal P_0 is just above 25 kW, and the optimal I_{sp} is at 4200 s. The discrete system flight time is longer than the continuous solution to absorb the small variations from the optimal. Indeed, over the entire examined neutral mass range the discrete system cost is relatively close to the continuous model cost. The two solutions begin to diverge slightly after 1800 kg as the maximum allowable P_0 of 25 kW constrains the system. (A 30 kW array would lower mission cost at high masses). The optimal mass fraction solutions are also very similar in cost to the cost optimal solution. As expected, the launch vehicle m_0 for optimal mass fractions are smaller than and the P_0 and I_{sp} are larger than the optimal cost values, but in this case the associated cost deltas for the different design parameters almost cancel (flight times are constrained to be equal since the mass optimizations do not optimize TOF). (The difference in I_{sp} causes a large difference in propellant load due to the high ΔV of this mission, noted in Table 4.) Again the mass optimal discrete system design diverges in cost from the cost optimal design between launch vehicle shifts.

Because the discrete and continuous solutions converge at the nominal, it is easy to begin with the optimal continuous solution (which usually require less optimization work) and automatically see where the discrete solution should lie. However, the optimal continuous solution crosses the discrete level sets at only a few special points, so intermediate designs (between these points) are less straight forward. For this mission the discrete solution usually converged to the next highest launch vehicle, and the nearest available P_0 , I_{sp} , and TOF to the continuous solution. But for the asteroid sample return mission, all of the vehicle parameters increase, with a slight reduction in flight time, while the asteroid rendezvous usually decreased the vehicle parameters and increased flight time. The comet rendezvous mission had a more or less random combination of both. Though the best direction to move when switching from a continuous model to a discrete model is not known a priori, the optimal discrete variables usually bound the optimal continuous solution. Thus, the continuous design answer (solvable with local optimization techniques) would make a good initial guess for a global optimization of the discrete solution space.

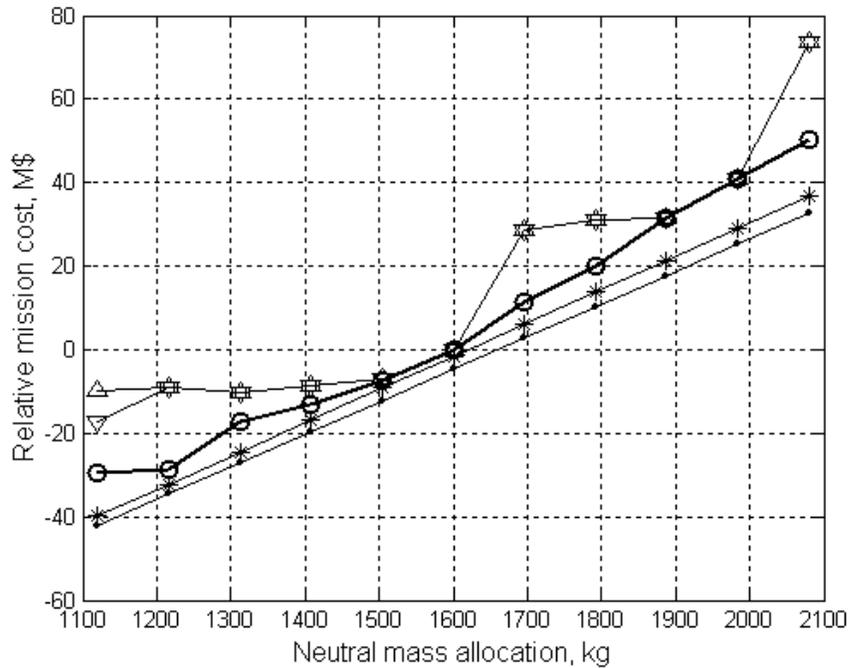


Figure 15 Mission cost for comet sample return. For discrete system levels \circ = cost, ∇ = m_N/m_0 , and Δ = m_T/m_0 optimizations, while \bullet = cost, \times = m_N/m_0 , and $+$ = m_T/m_0 solutions with continuous systems.

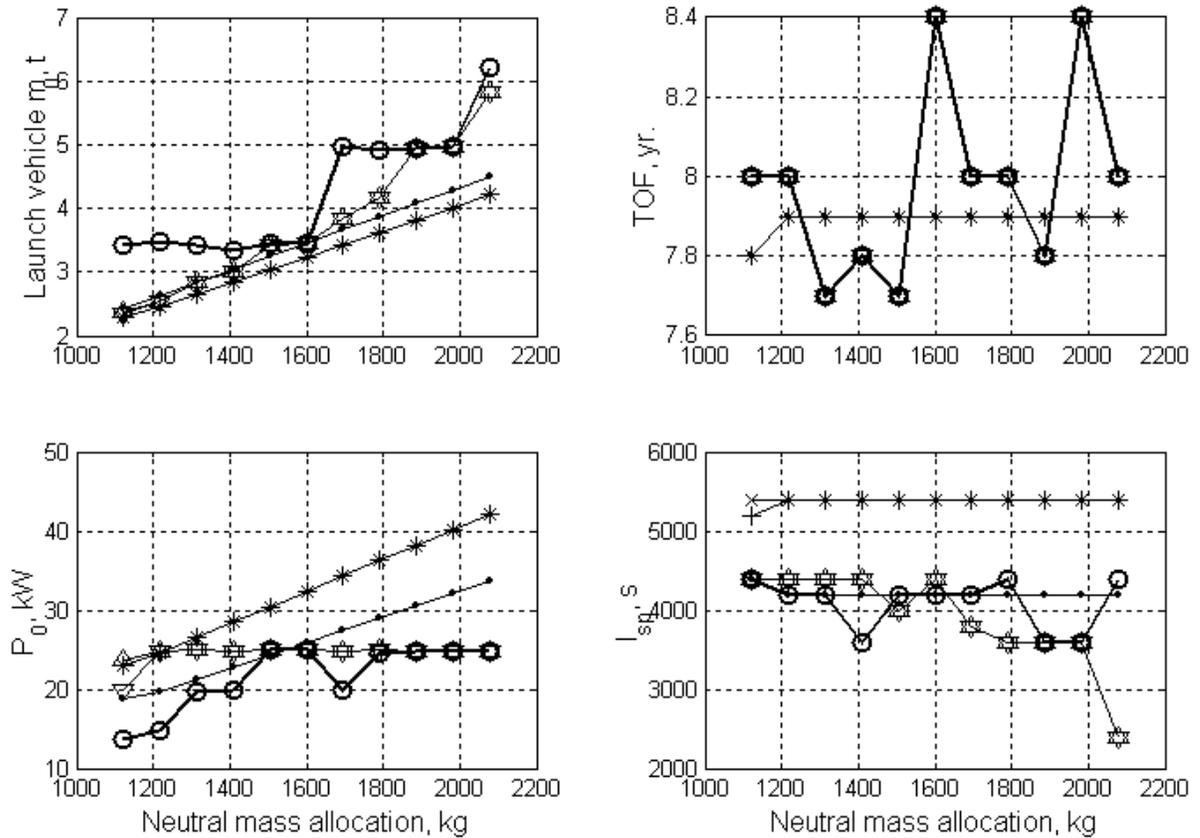


Figure 16 SEP system designs corresponding to Figure 15.

e. Outer planet

The final example combines the trajectory efficiency of gravity assist with the propulsive efficiency of SEP. As with the comet missions, a cost-optimal switch in launch vehicle occurs near the nominal neutral mass, but this time the switch occurs before the nominal value, so incorporating mass margin is not as costly. (The comet rendezvous mission required additional cost due to the larger launch vehicle, but the comet sample return mission increased cost independent of a change in launch vehicle.) From the bold curve in the TOF plot in Figure 18, the cost-optimal approach to increasing neutral mass before the launch vehicle switch is to increase flight time. Then at 4,500 kg, a larger launch vehicle becomes more cost efficient than increasing flight time, even though the launch vehicle costs 30 M\$ more than the smaller model and additional TOF up to 10.5 years is available. From Figure 17, the cost levels off for neutral masses above the nominal, and a 500 kg margin increases cost by only about 5 M\$. So the 5,000 kg neutral mass design is selected for its combination of low cost with adequate margin. This design has a P_0 and I_{sp} that are near the continuous solution values, but the launch vehicle is larger than optimal (which is usually the case), and the flight time is lower than the continuous solution to partially offset the launch vehicle cost. An interesting situation occurs at 5,600 kg where the continuous launch vehicle solution crosses the 6.5 t discrete m_0 level. At this point the discrete solution converges with the continuous solution allowing the limited parameter design to become as cost effective as the continuous design. While this design is cost-effective in terms of the ratio of cost to mass, it still costs 10 M\$ more than the nominal mass solution, and the extra 500 kg may not be necessary.

Because of the low ΔV of this mission (achieved via gravity assist) the mass fractions are naturally close to unity. Thus as seen in the launch vehicle plot in Figure 18, the m_0 for the cost-, neutral-mass-, and delivered-mass-optimal solutions are about the same. But, the high mass fraction also leaves little room for SEP system mass in the delivered mass allocation. In this case, the neutral mass optimization more closely corresponds to the cost optimal solution, where the difference in cost is only a few M\$ for continuous systems and 5–10 M\$ for discrete system levels. The optimal delivered mass solution with its bulky SEP system again requires significantly more cost to fit the required neutral mass.

f. Summary

The selected designs for the five example missions are compiled in Table 5. The designs correspond to a cost-optimal discrete value solution to minimize mission cost, while providing neutral mass allocation above the nominal to ensure a robust design. At the inception of a mission concept, before the major design parameters are known, a healthy mass margin is usually warranted. But once an SEP system is baselined, the mass margin can be kept as a component in the mass budget, and mission margin is spread among the various systems.¹⁰ The cost optimization approach still provides an efficient method of optimizing changes to the baseline mission (e.g. a change in payload or discontinuation of launch vehicle), but the selected design can correspond closer to the nominal neutral mass value.

Table 5 Selected SEP systems, launch vehicles, and flight times

Mission	Neutral mass, kg (margin)	m_0 , t	TOF, yr.	P_0 , kW	I_{sp}	Design cost M\$ above 0% margin	Cost-optimal margin parameter
Asteroid sample return	2000 (400)	3.5	3	5	high	1	propellant
Asteroid rendezvous	900 (100)	1.5	9	10	high	15	flight time
Comet rendezvous	1000 (200)	3.5	2.5	10	low	20	launch vehicle
Comet sample return	1700 (100)	5	8	20	high	10	launch vehicle (equally others)
Outer planet	5000 (500)	6.5	7.75	10	high	5	power and flight time

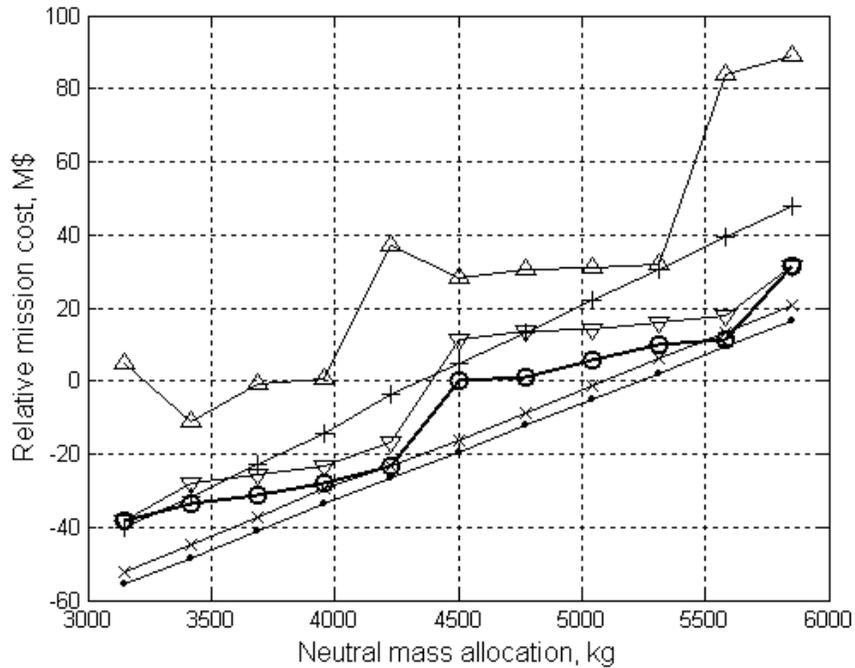


Figure 17 Mission cost for mission to Saturn. For discrete system levels $\circ = \text{cost}$, $\nabla = m_N/m_0$, and $\Delta = m_P/m_0$ optimizations, while $\bullet = \text{cost}$, $\times = m_N/m_0$, and $+$ = m_P/m_0 solutions with continuous systems.

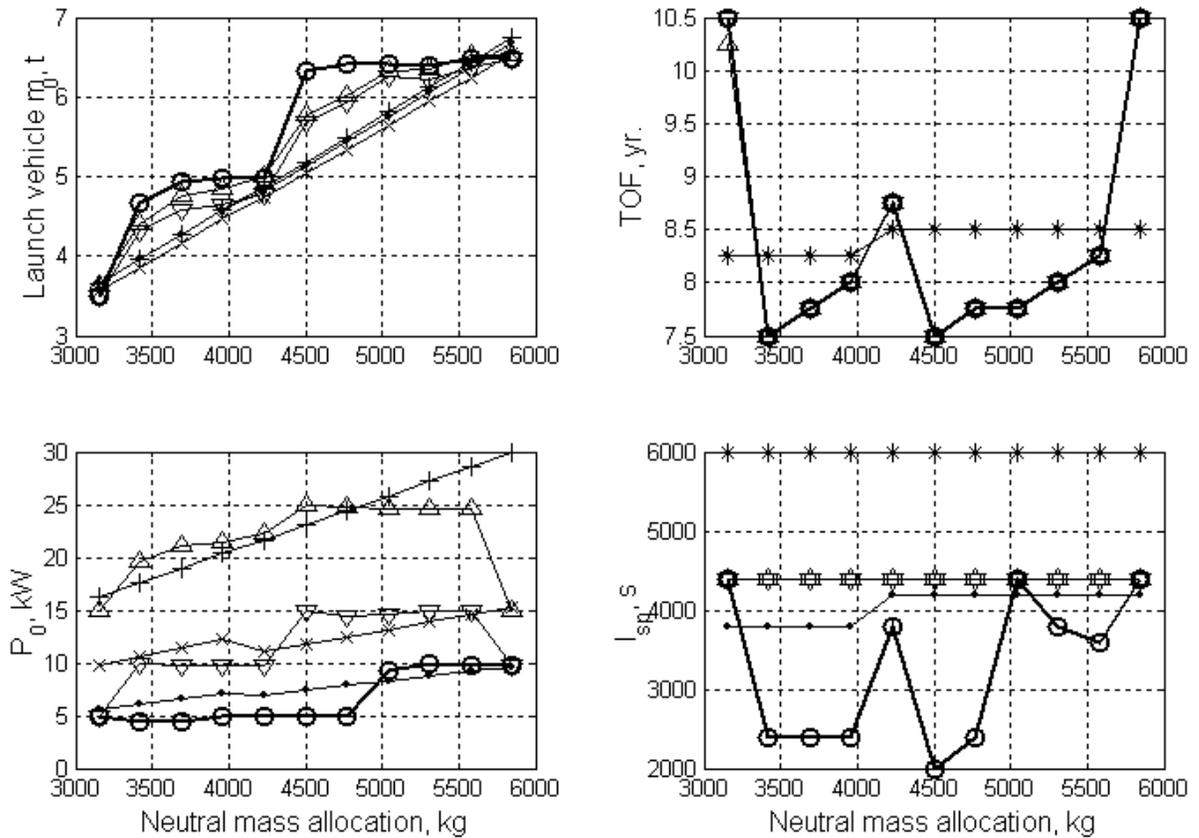


Figure 18 SEP system designs corresponding to Figure 17.

V. Conclusions

Mission cost provides a figure of merit that combines the selection of launch vehicle, flight time, power system, and propulsion system into a single optimization problem. A single trade study of minimum mission cost over a range of neutral mass values then indicates the optimal system configuration for any desired mass margin. When more mass margin is desired, the cost-optimal system parameter depends strongly on the mission type. Sufficiently large increases in mass must be accommodated by increasing the launch vehicle capability, but the considerable increase in cost associated with a bigger launch vehicle is offset by decreasing flight time, power level, and/or propellant mass. It is found that, for free choice of launch vehicle, flight time, and power and propulsion system, the cost-optimal configuration is generally not mass optimal, and mass-optimal designs are usually not cost optimal. Thus, mass fraction optimizations are usually insufficient to determine the minimum cost solution without brute force searches. Discrete levels in system parameters (e.g. gaps in launch vehicle size) can cause a dramatic increase in mission cost when compared to the optimal value for continuous systems. However, the optimal continuous system configuration supplies an initial design to optimize the discrete system set, thus producing a cost efficient design for the mission.

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