# Evaluation of Radioisotope Electric Propulsion for Selected Interplanetary Science Missions

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**In response to a request by the NASA Science Mission Directorate's Radioisotope Power Systems Program, the In-Space Propulsion Projects Office (ISPPO) conducted a study to assess the applicability of Radioisotope Electric Propulsion (REP) to science missions that are representative of mission classes relevant to the In-Space Propulsion Program (ISPP) including small body targets, medium outer planet class, and main belt asteroids and comets. REP was compared against state-of-the-art (SOA) chemical propulsion and solar electric propulsion. The three science missions evaluated were a Trojan Asteroid Orbiter, a Comet Surface Sample Return (CSSR), and a Jupiter Polar Orbiter with Probes (JPOP). Each mission has a unique payload, trajectory, and optimal specific impulse. The results show that REP offers a significant trip time reduction and increased target capture for the Trojan Asteroid Orbiter mission when utilizing a second generation radioisotope power system (RPS) and an advanced Hall thruster. Marginal benefits were realized with the use of first generation RPS. REP was found not to be a viable option for the JPOP mission primarily due to the increased propellant mass required for capturing into a final parking orbit. For the CSSR mission, REP may be a viable option, although more analyses need to be performed.**

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## Nomenclature

| | | |
|---|---|---|
| AU | = | Astronomical Units |
| $C_3$ | = | Launch Energy |
| $\Delta V$ | = | Delta Velocity |
| $I_{sp}$ | = | Specific Impulse |
| L4, L5 | = | Leading and Trailing Lagrange Points |
| $R_J$ | = | Radius of Jupiter, 71,492 km |

## I. Introduction

Previous studies[1,2] have suggested significant benefits to using electric propulsion powered by radioisotope power systems (RPS) for outer planetary exploration. The RPS, a key element needed for radioisotope electric propulsion (REP), has been successfully used in space for electrical power by the United States since 1961. The RPS operates independent of the orientation and distance from the Sun and is therefore capable of long-lived autonomous operations. RPSs generally provide higher power levels for science payloads.

This study assessed the benefits and applicability of REP to missions relevant to the In-Space Propulsion Program (ISPP) using first and second generation RPS with specific powers of 4 $W_e$/kg and 8 $W_e$/kg, respectively. Three missions representing small body targets, medium outer planet class, and main belt asteroids and comets were evaluated. Those missions were a Trojan Asteroid Orbiter, Comet Surface Sample Return (CSSR), and Jupiter Polar Orbiter with Probes (JPOP). For each mission, REP cost and performance was compared with solar electric propulsion system (SEPS) and SOA chemical propulsion system (SCPS) cost and performance. The outcome of the analysis would be a determinant for potential inclusion in the ISPP investment portfolio.

## II. Systems Analysis

The general analysis approach of this study consists of three primary steps. The first step is the development of propulsion systems and spacecraft systems models for each of the technologies investigated. This is the systems model building step. The second step consists of performing parametric mission analysis to determine the optimal power and propulsion requirements to perform the mission. This is the trajectory optimization step, and was generally done concurrently with the systems modeling step. The last step entails the use of the results from the mission analysis to make comparisons between the technologies based on a set of figures of merit (FOMs). The FOMs are chosen through a collaborative effort between the study team members including input from Marshall Space Flight Center (MSFC), Glenn Research Center (GRC), Jet Propulsion Laboratory (JPL), and Science Applications International Corporation (SAIC).

In the first step of the analysis, propulsion systems models were developed, in coordination with GRC and JPL, for the REPS, SEPS and SCPS. These models were developed in a flexible parametric way that allows for rapid modification and analyses. The models include subsystem level propulsion, power, structures, thermal, and spacecraft subsystem systems models for REP, SEPS, and SOA chemical bipropellant propulsion. The models are generally experience based (based on historical spacecraft and propulsion systems), with some physics based models such as for thermal control.

The next step, occurring concurrently with the first step, includes the selection of the reference missions to facilitate technology comparisons. The reference missions chosen for this study were a Trojan asteroid orbiter, a Comet Surface Sample Return, and a Jupiter Polar Orbiter with Probes. This step also includes trajectory generation and optimization for the chosen reference missions. Tools used to perform this trajectory optimization work include SEPTOP[3] for SEP Trajectories, VARITOP and DTOM[4,5] for REP Trajectories, and MIDAS[6] for chemical propulsion trajectories. During this phase of the study, launch vehicle trades were performed to ascertain the minimum launch vehicle capable of performing the mission.

The third and final step of the analysis process is composed of a technology assessment based on figures of merit. The figures of merit used in this comparison include the LV required to perform the mission, the trip time for delivery of reference payloads and the cost for system (all technologies assumed to be at >=TRL 6).

The RPS was simply modeled as a first and second generation of the power system technology. The first generation RPS was modeled with a specific power of 4 $W_e$/kg and the second generation RPS was modeled with a specific power of 8 $W_e$/kg. The first generation RPS specific power is generally an optimistic estimate of the technology being developed, but was used to insulate this study from changes resulting from the development of the two RPS technologies[7,8] The second generation RPS specific power estimate of 8 $W_e$/kg was chosen to represent studies performed for advanced RPS technology[9].

The EP system, which provides a significant proportion of the in-space Δv, was included in the optimization by means of a simple EP model. This EP system model used a theoretical performance model based on current best estimates of the performance of low-power EP systems. The performance model, shown in Figure 1, related efficiency to specific Impulse ($I_{SP}$) at power levels between 500 $W_e$ and 1000 $W_e$ into the EP system. These curves are representative of gridded-ion thrusters or Hall thrusters at these power levels. The specific thruster technology was chosen after an optimal $I_{SP}$ was determined for the mission. The EP system mass model was based on mass estimates of a low-power gridded-ion propulsion system, with heritage from the NSTAR[10] and NEXT[11] programs. The low power gridded-ion propulsion system is currently unfunded and therefore not developed.

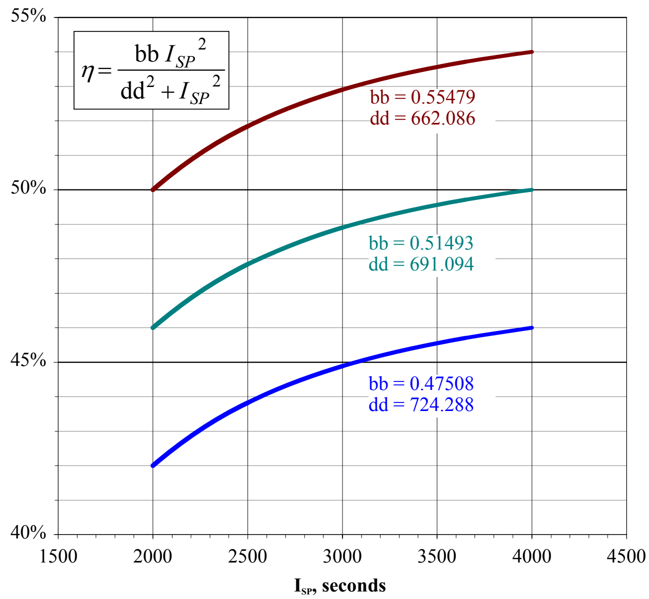



**Figure 1 . EP System Efficiency vs. $I_{SP}$**

## III. Mission Analysis

### A. Trojan Asteroid Orbiter Mission

Both Trojan asteroids and Centaur bodies have been identified as primitive body targets of interest for solar system exploration.[12] The Trojan asteroids are asteroids that sit at (or near) Jupiter's L4 and L5 Lagrange points. The L4 and L5 points, illustrated in Figure 2 are 60 degrees in front of and behind Jupiter in its orbit. They are stable points in the 3-body system where asteroids have tended to gather in the Sun-Jovian system.

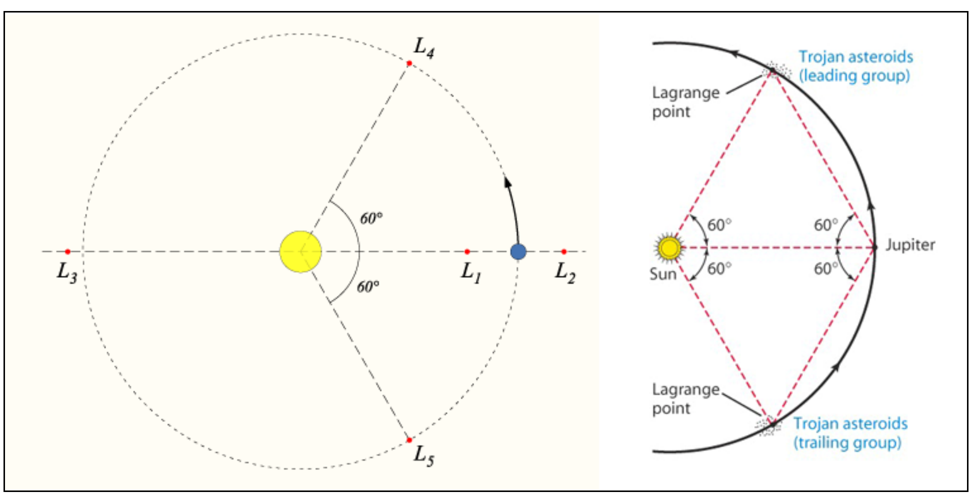



**Figure 2. Illustration of the Lagrange points and the corresponding locations of Jupiter's Trojan asteroids**

Although every planet has an L4 and L5 point with respect to the Sun, Jupiter is the only one that has a significant number of known asteroids (at least of significant size) that have settled into these stable orbits. Over 1700 asteroids populate Jupiter's stable Lagrange points. Figure 3 provides a picture of the main asteroid belt (green) and the obvious lumping of asteroids at Jupiter's L4 and L5 points (white).

Small body rendezvous missions are propulsively challenging missions. Because asteroids have almost no gravity, the target body's gravitational field can not be used to reduce the $\Delta V$ required to orbit the body. As a result, chemical propulsion missions to asteroids generally require relatively complicated gravity assist trajectories and long times of flight or require a large on-board propulsion system to accomplish orbit insertion at the target body.

The Dawn spacecraft addresses these constraints by using solar electric propulsion to rendezvous with two main belt asteroids. The Trojan asteroids are located farther from the sun, making it relatively difficult to use solar electric propulsion (SEP) for the final rendezvous maneuvers. In this section, we consider the use of radioisotope powered electric propulsion (REP) for a potential Trojan asteroid rendezvous mission and compare missions using REP to alternate architectures using SEP and chemical propulsion.

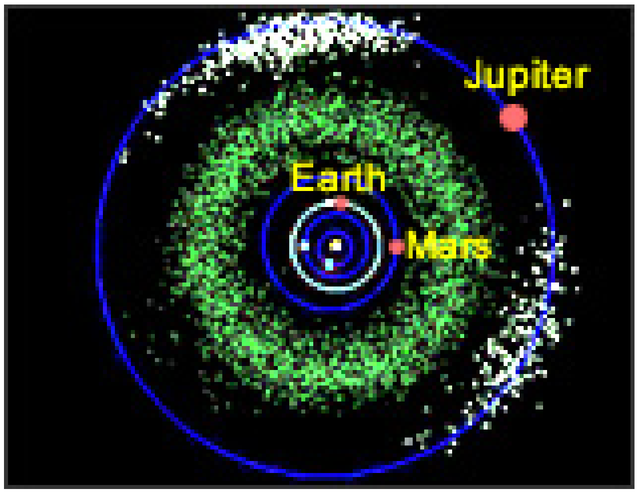



**Figure 3. Illustration of asteroid belt and the Jupiter Trojan asteroids**

*1. Analysis Approach and Overview*

The objective of the Trojan asteroid orbiter mission would be to put a small scientific payload into orbit around a Trojan asteroid. Several different architectures using combinations of REP, SEP, and chemical propulsion were considered as a means to meet this objective. For this study, all architectures were required to deliver a scientific instrument payload with a mass of 42 kg, standby power consumption of 30 W and peak power consumption of 100 W. These values match the mass and power requirements of the payload for the Dawn spacecraft.[13] The overall analysis approach used in this study is summarized in Figure 4.

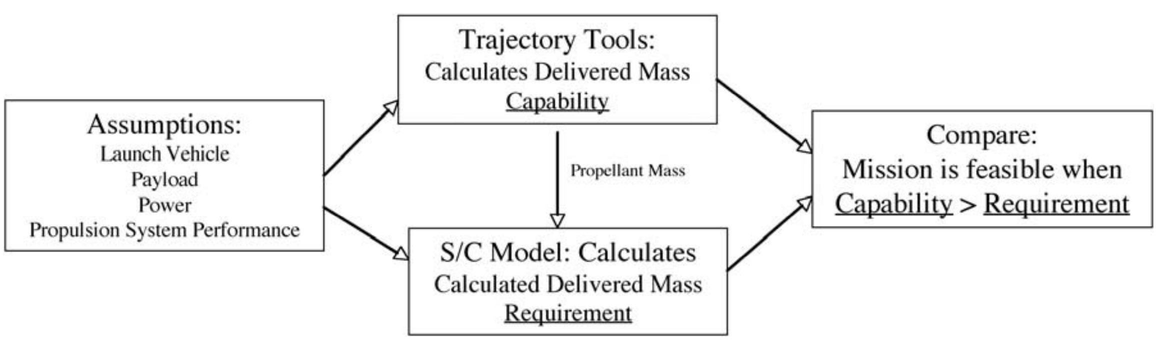



**Figure 4. System Analysis Approach**

First, a series of assumptions was made about the performance of the launch vehicle and on-board power and propulsion systems. In all cases, it was assumed that the spacecraft would be launched directly to an Earth escape trajectory using a medium class launch vehicle in the Atlas V and Delta IV families. This assumption is generally consistent with the requirements of medium class cost-capped missions (the "New Frontiers" mission class). The power and propulsion assumptions are specific to each architecture.

Once the assumptions were established, the system's delivered mass *capability*, defined as the total mass delivered to the final orbit around the destination asteroid, was calculated using trajectory optimization tools based on the launch vehicle, power, and propulsion system's performance parameters. The delivered mass *requirement*, defined as the mass of the spacecraft at the end of the mission including the payload, spacecraft bus, and residuals propellants, was calculated separately using a spacecraft dry mass model. Inputs to the model include the power consumption and the mass of the propellant required to meet the mission objectives. The resulting mass *requirement* was compared to the calculated mass *capability* to determine if the mission option was feasible or infeasible. In cases where the capability exceeded the requirement, the architecture was judged to be feasible. In cases where the requirement exceeded the capability, the mission was judged to be infeasible.

Because the scientific payload used in these models was based on Dawn, the payload requirements are representative of a low cost Discovery class mission. It is not clear that the science provided by this payload would be sufficient justification for a medium cost "New Frontiers" class mission. Therefore, this payload should be considered a "science floor" payload suitable for a feasibility analysis, but not necessarily suitable for a New Frontiers class mission proposal.

## *All-Chemical Mission Architectures*

Two types of all-chemical mission architectures were considered in this study. The first option was a direct ballistic trajectory with chemical propulsion used for deep space maneuvers and for orbit insertion at the target asteroid. The second option was similar, but included a Jupiter gravity assist to reduce the spacecraft's approach velocity to the destination asteroid. The mass requirements for both options were calculated using the spacecraft mass model shown in Table 1 below.

### Table 1. All-Chemical Trojan Orbiter Spacecraft Mass Model

| | | |
|---|---|---|
| **Nominal Transit Chem Propellant Ma** | 620 | kg |
| **RPS Alpha** | 8 | W/kg |
| **Power Degradation** | 1.15 | % of BOL/year |
| **Degradation Period** | 15 | years |
| **Max Power end of Degradation Peric** | 330 | W |

| **General Model** | **Mass, kg.** | **Mass Comments** | | **Power (W)** | **Power Comments** |
|---|---|---|---|---|---|
| **Instruments** | **42** | Same as Dawn CBE at Neutral Mass Review | | **60** | ~60% of Dawn Peak |
| **ACS** | **40** | Dawn:4 wheel+star tracker system | | **30** | Dawn |
| **C&DH** | **25** | 2 ACE's, 2 CEU's | | **70** | Dawn |
| **Comm** | **30** | Dawn: 2 TWTA's | | **12** | Dawn |
| **Harness** | **29** | Team X Harness model | | **4** | Estimate: 1.5% |
| **Mech/Structures** | **127** | Team X structure model | | **0** | NOTE: Power Budget is On Station |
| **Thermal** | **23** | Team X: thermal model | | **38** | Team X: thermal model |
| **Chemical Propulsion** | **94** | Scaled from Team X 755 kg. Dual mode bipropellant, 12 MIT + 4 22 N + 1 main | | **5** | assumption |
| **EPS** | **71** | | | **30** | Team X: power model (250W bus) |
| Electronics | 17 | Team X: power model (250 W bus) | | | |
| Battery | 6 | Team X: minimum battery for bus stability | | | |
| RPS | 48 | 350 W at 8W/kg. | | | |
| **LV Adapter** | **17** | 1.5% of wet mass, no contingency | | | |
| **Total Dry Mass, CBE** | **497** | | **Total Power, CBE** | **249** | |
| **Contingency** | 149 | 30% contingency | **Bus Power, CBE** | 249 | |
| **Total Dry Mass w/Contingency** | **646** | | **Bus Power w/Contingency** | 324 | 30% contingency |
| **Propellant, Chemical** | **688** | | **EP Power with Contingency** | 0 | 5% contingency |
| Chemical Propellant, Nominal | 620 | | | | |
| Chemical Propellant, ACS | 15 | half of Dawn | | | |
| Chemical Propellant, TCM | 3 | approx. 30 m/s | | | |
| Chemical Propellant, deltaV-Margin | 15 | approx 2% dV | | | |
| Chemical Propellant, Orbital Ops | 15 | approx 50 m/s | | | |
| Chemical Propellant, Residuals | 20 | Team X: 3% | | | |
| **Xenon Propellant** | 0 | | **Total Power w/Contingency** | **324** | |
| **Total Wet Mass w/Contingenc** | **1334** | | | | |
| **Total Delivered Mass to Astero** | **714** | | | | |

**= From Dawn Mass Budget**
**= Team X Design Model**
**= Model Specific to this Study**

The instruments, attitude and control system (ACS), command and data handling (C&DH) and communication subsystem mass and power were estimated as fixed values taken directly from the Dawn spacecraft. Power was provided by an advanced RPS with an output power of 350 W and a specific mass of 8 W/kg. This specific power is representative of a 2nd generation RPS, beyond the current state of the art. Although the assumption is aggressive, it is consistent with assumptions used in the REP architectures and therefore provides for a fair comparison of the two architectures. The RPS was oversized to account for 1.15% power degradation/yr over the life of the mission. The remaining subsystems were derived or scaled from subsystem parametric models developed by JPL's advanced projects design team ("Team X"). The Team X parametric design tools were used to generate a point solution for a nominal spacecraft configuration, and then the mass of the propulsion tanks and spacecraft structure were scaled to reflect changes in the mass of the on-board propellant. The propulsion system was modeled as a dual mode chemical system with one main thruster generating 450 N of thrust at a specific impulse of 325 seconds plus 16 small thrusters for attitude control and maneuvering. An overall dry mass margin of 30% was added to generate the total dry mass with contingency that is used as the basis for this analysis.

In addition to the deterministic propellant allocation for orbit insertion and deep space maneuvers, the model added 80 m/s of $\Delta V$ for targeting and orbital operations as well as 2% $\Delta V$ margin and 3% residuals. The level of residuals was consistent with values generated by the Team X propulsion model.

Delivered mass capability was calculated using MIDAS and assumed launch on an Atlas 551 launch vehicle. A script was used to calculate a delivered mass for each of the numbered Trojan asteroids with an inclination less than 20 degrees (about 700 targets), and the delivered mass varied widely depending on which Trojan asteroid was selected as the target. The calculated trajectories generally had launch $C_3$'s between 55 and 90 km$^2$/s$^2$, flight times between 3 and 5 years, and required between 4 and 10 km/s of on-board $\Delta V$ capability. The results of this analysis are summarized in Figure 5. Further details on trajectory calculations are available in reference 14.

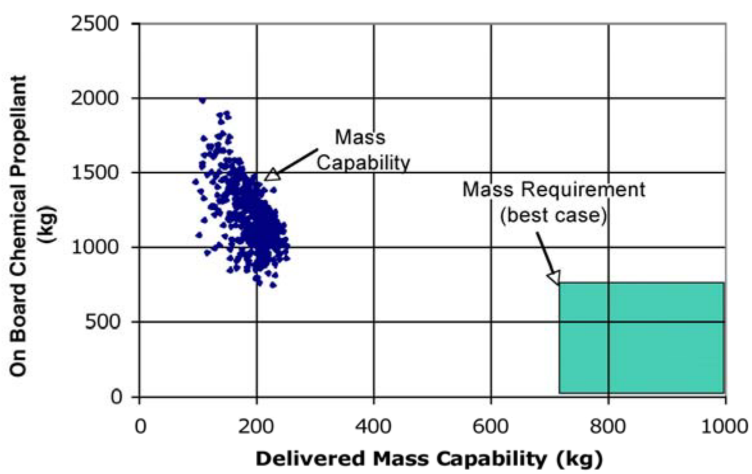



Figure 5. Trojan Orbiter Chemical Direct Trajectory Summary

The points on the left side of Figure 5 correspond to the delivered mass capability and propellant mass required to orbit each of the Trojan asteroid targets analyzed in this study. The propellant requirements are substantial; between 800 kg and 2000 kg of on-board propellant are required, and the maximum mass delivered to any destination is only ~250 kg. The delivered mass requirement, as derived from the spacecraft model, is indicated by the arrow on the right side of Figure 5. A spacecraft loaded with 800 kg of propellant has a delivered mass of about 725 kg. In order for the mission to be feasible, the delivered mass capability must fall within the shaded region at the lower right side of the graph. This is the range in which the delivered mass capability is greater than the mass requirement and the propellant used to reach the asteroid is less than the propellant load to calculate the requirement.

As propellant mass increases, the delivered mass requirement also increases because the size of the propellant tanks and support structure increases. With chemical direct trajectories, the mass capability is always less than the mass requirement regardless of the size of the propellant tank. It is therefore concluded that:

- All chemical direct Trojan orbiter missions are infeasible using a medium class launch vehicle.

Addition of a Jupiter Gravity assist (JGA) to the transit trajectory lowers the on-board propulsion requirements and substantially increases the system's delivered mass capability. Figure 6 summarizes the JGA analysis. The trajectories generated by the optimizer generally have launch $C_3$'s between 75 and 90 km$^2$/s$^2$, flight times between

10 and 15 years, and require between 1 and 7 km/s of on-board ΔV capability. All options assume use of an Atlas 551 launch vehicle.

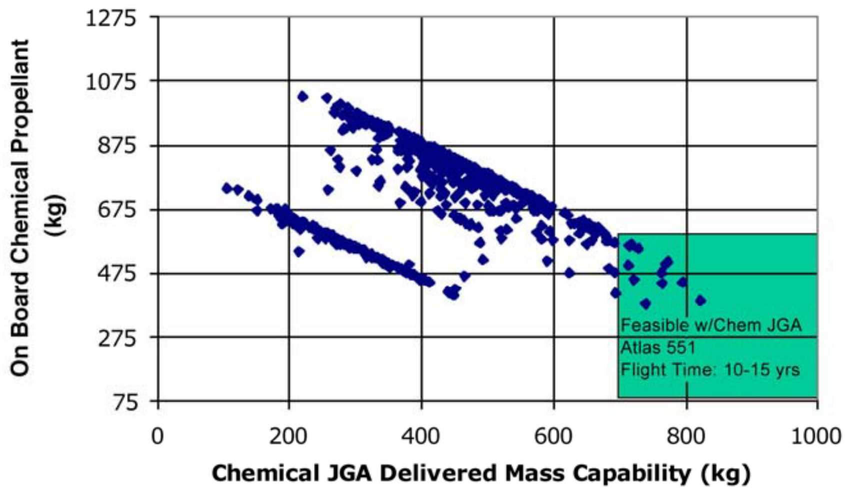



**Figure 6. Trojan Orbiter Jupiter Gravity Assist Trajectory Summary**

The JGA trajectories require much less propellant than the chemical direct trajectories and generally deliver much more mass to the final destination. The delivered mass requirements, indicated by the box in the lower right of the figure, are slightly less than in the chemical direct case because of the smaller propellant load. A small fraction of the missions examined fall within the shaded region, indicating that they are feasible with a JGA. The flight times are relatively long and the range of destinations is very limited, about 2% percent of the total targets considered. Based on these results, it was concluded that:

- A very limited number of Trojan asteroids can be reached using chemical propulsion with a Jupiter Gravity Assist.

It should be noted that this conclusion assumes that the spacecraft is powered by a second generation RPS. Use of a first generation RPS would further reduce the range of feasible targets. Alternately, use of a more aggressive design (incorporating advanced chemical propulsion for example) would increase the range of feasible targets for this architecture.

*Radioisotope Powered Electric Propulsion Options*

Two classes of REP missions were considered in this study, one based on 1st generation RPS technology and one based on 2nd generation RPS technology. The mass requirements for the REP missions were calculated using the mass model shown in Table 2.

The instruments, ACS, C&DH and communications subsystem mass estimates were again taken directly from Dawn. The harness, structure, thermal, EPS electronics, battery, and launch vehicle adapter were all derived or scaled from Team X subsystem parametric models. The electric propulsion subsystem was sized using the advanced Hall thruster option of the EP subsystem model described previously. The mass of the RPS was calculated assuming a specific mass of either 4 W/kg for 1st generation technology or 8 W/kg for 2nd generation technology and RPS was oversized to allow for 1.15% power degradation/yr over the life of the mission. The xenon tank was sized using a 4.5% tank mass fraction. This matches the mass fraction of the xenon tank used on Dawn. An overall dry mass margin of 30% was added to the total to generate the total dry mass with contingency used as the basis for this study.

In addition to the deterministic propellant allocation for orbit insertion and deep space maneuvers, the model included 7% xenon propellant margin to account for worst case flow rate, missed thrust periods, restarts, and residuals. The model also included 5 kg of xenon for orbital operations and 33 kg of chemical propellant for attitude control and other operations. These values are consistent with assumptions used for mission planning for Dawn.

## Table 2. Baseline Radioisotope Powered Electric Propulsion Spacecraft Model (Nominal 750 W Atlas 541/551 case shown)

| | | |
|---|---|---|
| **RPS Alpha** | 8 | W/kg |
| **Power Degradation** | 1.15 | % of BOL power/year |
| **Degradation Period** | 6.9 | years |
| **Max Power end of Degradation Period** | 1068 | W |

| | Mass, kg. | Sources | | Power (W) | Power Comments |
|---|---|---|---|---|---|
| **Instruments** | **42** | Dawn CBE, Neutral Mass Review | | **30** | Dawn |
| **ACS** | **40** | Dawn:4 wheel+star tracker system | | **30** | Dawn |
| **C&DH** | **25** | 2 ACE's, 2 CEU's | | **70** | Dawn |
| **Comm** | **30** | Dawn: 2 TWTA's | | **12** | Dawn |
| **RCS / Prop** | **14** | Dawn: 12 thruster hydrazine | | **2** | Dawn |
| **Harness** | **28** | Team X: structure model | | **15** | Estimate: 1.5% |
| **Mech/Structures** | **111** | Team X structure model (no LV adapter) | | **0** | |
| **Thermal** | **25** | Team X: thermal model | | **42** | Team X thermal model |
| **Electric Propulsion** | **55** | | | **750** | REP Assumption |
| Xenon Tank | 21 | Scaled using Dawn Tank Mass Fraction (4.5%) | | | |
| EP Subsystem | 34 | REP EP Mass Model 1.1 w/ADVANCED HALL | | | |
| **EPS** | **166** | | | **15** | Team X: power model |
| Electronics | 17 | Team X: power model (250 W bus) | | | |
| Battery | 6 | Team X: minimum battery for bus stability | | | |
| RPS | 144 | Calculated | | | |
| **LV Adapter** | **16** | 1.5% of wet mass, no contingency | | | |
| **Total Dry Mass, CBE** | **552** | | **Total Power, CBE** | **966** | |
| **Contingency** | 166 | 30% contingency | **Bus Power, CBE** | 216 | |
| **Total Dry Mass w/Contingency** | **718** | | **Bus Power w/Contingency** | 281 | 30% contingency |
| **Propellant, Chemical** | **33** | | **EP Power with Contingency** | 788 | 5% contingency |
| Chemical Propellant, Nominal | 30 | From Dawn | | | |
| Chemical Propellant Margin | 3 | Dawn margin 10% | | | |
| **Propellant, Xenon** | **509** | | | | |
| Xenon Propellant, Primary Transit | 471 | | | | |
| Xenon Propellant, Contingency/Residuals | 33 | Dawn equivalent 7% | **Total Power** | **1068** | |
| Xenon Propellant, Orbital Operations | 5 | Generous Estimate | | | |
| **Total Wet Mass w/Contingency** | **1260** | | | | |
| **Total Delivered Mass to Asteroid** | **789** | | | | |

**= From Dawn Mass Budget**
**= Team X Design Model**
**= Model Specific to this Study**

Because the optimization of low thrust trajectories is more labor intensive than the analysis of ballistic trajectories, it was not practical to analyze each of the 700 Trojan asteroids considered in the chemical propulsion study. Instead, the low thrust trajectory optimizer VARITOP was used to generate optimized trajectories to six asteroids chosen from the 700 targets considered in the chemical study. The analysis considered two power levels corresponding to 750 W and 1000 W PPU input power. Most of the cases use the generic EP thruster model described previously, but two cases assume use of an advanced Hall Thruster with a specific impulse of 2420 seconds and efficiency of 52% at 1000 W PPU power. The REP analysis results are summarized in Table 3. Further details about the selection of trajectories are available in reference 14. Although both 750 W and 1000 W trajectories were examined in many cases, only the best of the two power levels are shown in Table 3.

## Table 3. Trojan Asteroid REP Asteroid Summary

| Asteroid Name | 65211 | 60388 | Sarpedon (2223) | Achilles (588) | Hektor(624) | Aneas (1172) |
|---|---|---|---|---|---|---|
| PPU Input Power (W) | 750 | 750 | 750 | 1000 | 1000 | 1000 |
| Launch Vehicle | Atlas 541 | Atlas 541 | Atlas 551 | Atlas 551 | Atlas 551 | Atlas 551 |
| Thruster Model | REP Generic | REP Generic | REP Generic | REP Generic | Low Power Hall | Low Power Hall |
| C3 ($km^2/s^2$) | 69.5 | 69.2 | 74.2 | 67.6 | 75.0 | 77.1 |
| Transit Xenon Propellant Mass (kg) | 481 | 578 | 596 | 554 | 497 | 409 |
| Specific Impulse (sec)** | 1698 | 1535 | 1657 | 1783 | 2420*** | 2420*** |
| Trip Time (yrs) | 10.0 | 6.9 | 8.2 | 6.0 | 8.5 | 7.0 |
| Delivered Mass Capability (kg) | 838 | 861 | 834 | 1130 | 880 | 891 |
| **Second Generation RPS** | | | | | | |
| Mass Requirement (kg) | 813 | 826 | 833 | 905 | 819 | 859 |
| Feasible Mission? | Yes | Yes | Yes | Yes | Yes | Yes |
| **First Generation RPS** | | | | | | |
| Mass Requirement (kg) | 1034 | 1046 | 1056 | 1101* | 1031 | 1137 |
| Feasible Mission? | No | No | No | Yes* | No | No |

* Based on modified "ultralight" custom spacecraft mass model
** Optimum specific impulse shown unless otherwise indicated
*** Non-optimum, prespecified value

The optimized REP trajectories had $C_3$'s between 69 and 75 $km^2/s^2$ and flight times that varied from 6 to 10 years depending on the destination. Optimum $I_{sp}$'s were calculated for missions using the REP generic thruster. The calculated optimum was consistently below 2000 seconds, well within the Hall thruster's region of operation.

Feasible missions were found for all six targets when using the second generation RPS. No feasible missions were found when using the baseline spacecraft mass model with the first generation RPS. However, one feasible mission was found when the mass model was modified to represent a customized lightweight spacecraft. This lightweight model assumed use of a customized spacecraft structure that is 10% lighter than the baseline model, a customized lightweight C&DH system that is 40% lighter than the baseline, and a customized xenon tank that is 11% lighter than the baseline. Although mass reductions of this magnitude are probably feasible, they are likely to be expensive, to incur cost and schedule risk, and to require significant non-recurring engineering. The development of custom lightweight hardware is often not affordable on a cost-capped spacecraft program. The custom spacecraft model reduced the dry mass by 60 kg. The REP results and the chemical JGA results are compared in Figure 7 and Figure 8 below.

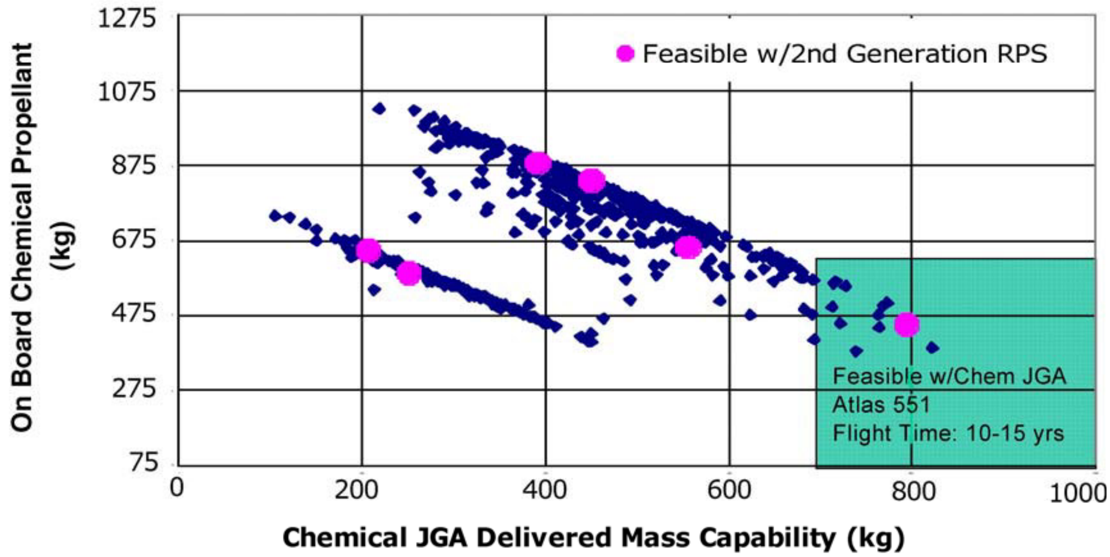



**Figure 7. Second Generation REP Missions Options Overlaid on Chemical JGA Results**

Figure 7 shows feasible 2nd generation REP destinations overlaid on the JGA results shown in Figure 6. The results show that 2nd generation REP allows access to a much wider range of targets than the combination of chemical propulsion with a Jupiter gravity assist. Based on the distribution of targets, it seems likely that most of the Trojan asteroids could be reached using 2nd generation REP. Figure 8 shows feasible and infeasible 1st generation REP destinations. First generation REP appears to have marginal applicability to this mission. Only one feasible case has been identified, and this case assumes the use of a customized lightweight spacecraft that is likely to be too expensive for use on a cost-capped mission. Based on these results, we draw the following conclusions regarding the use of REP for a Trojan asteroid orbiter.

- REP with 2nd Generation RPS enables a wide range of targets for a Trojan asteroid orbiter compared to chemical propulsion
- REP with 1st Generation RPS is marginal for a Trojan asteroid orbiter. With a customized lightweight spacecraft, it is probably feasible to reach a limited number of destinations.

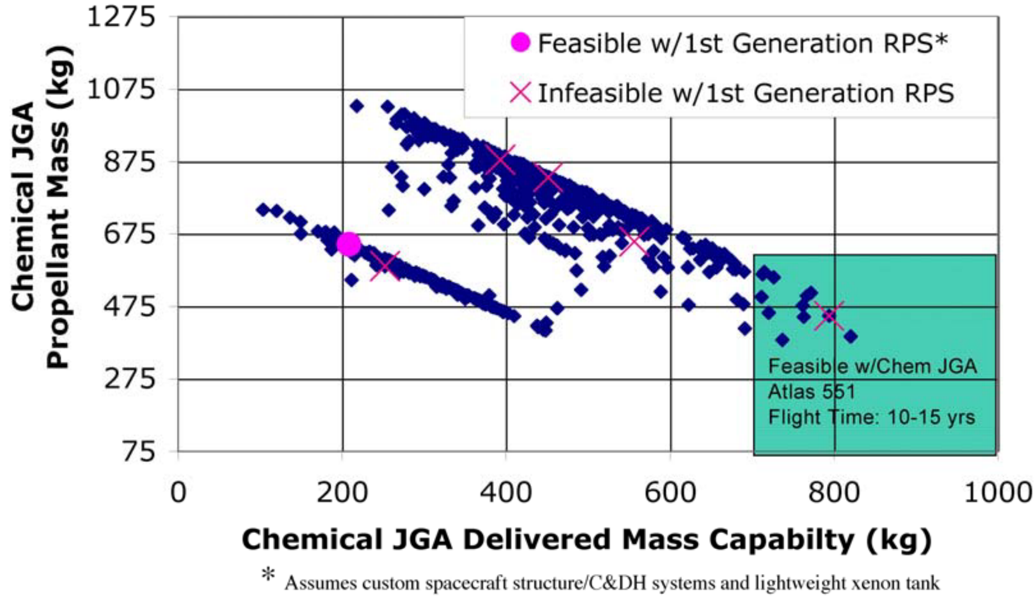



**Figure 8. First Generation RPS Mission Options Overlaid on Chemical JGA Results**

In addition to these general findings, some more specific findings can be made with respect to transit time. Figure 9 shows the calculated transit time for four of the feasible 2nd generation REP missions.

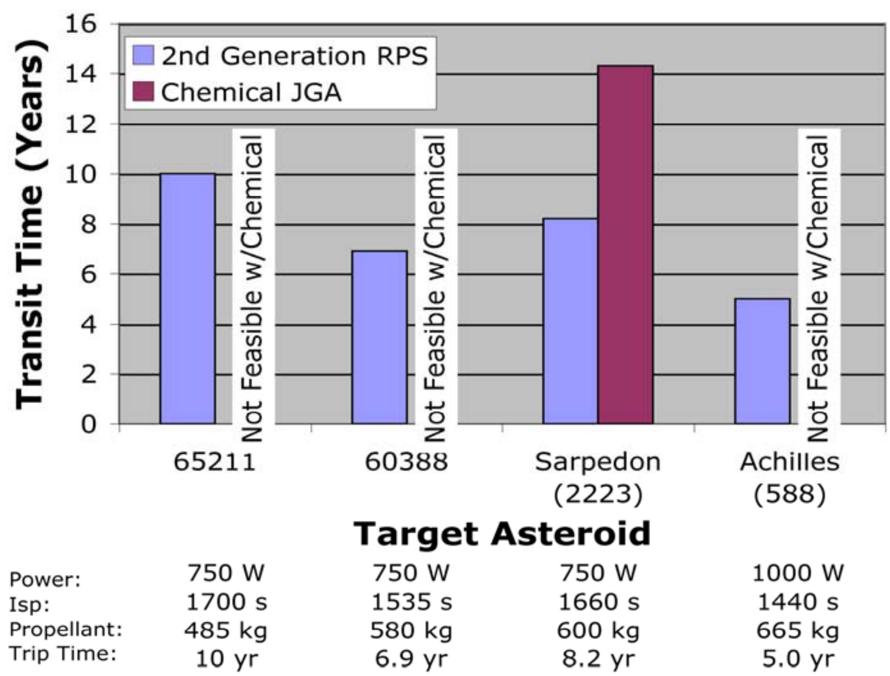



**Figure 9. Feasible REP Mission Flight Time Summary**

One of the targets, Sarpedon, is feasible with a chemical-JGA trajectory, but the flight time is much higher for a chemical-JGA option than for 2nd generation REP. The following observation can be made from this result.

- REP with 2$^{nd}$ Generation RPS can substantially lower trip time compared to Chemical Propulsion.

Identical 42 kg/100 W payloads were used in this comparison of chemical-JGA and REP mission architectures. However, the resulting REP spacecraft has considerable excess power capability because the 750 W to 1000 W dedicated to the electric propulsion subsystem during cruise is available for use in the science orbit. This power could support higher instrument duty cycles or, if mass is available, higher power instruments or higher power amplifiers for communications which would allow transmissions at higher data rates. All of these factors enhance the science returned by the mission and should be considered as part of the total system benefits of REP.

The xenon throughput requirement for the REP missions was very high, typically > 500 kg, which is well beyond the state of the art capability for low power Hall thrusters. Considerable development would be needed to achieve the throughputs necessary to support these missions. Finally, it should be noted that addition of a Star 48 upper stage to the launch vehicle may improve the mass performance of these missions, and should be considered in future work on both Trojan asteroid and Centaur object REP studies.

## *Solar Electric Propulsion*

Two types of SEP architectures were considered in this study, an all-SEP architecture and a SEP-chemical architecture. These options are discussed separately below.

### *SEP-Only Mission Options*

Another mission architecture considered in this study was the use of solar electric propulsion (SEP) to transit to the asteroid and to accomplish the orbit insertion maneuver. This is the strategy being used by the Dawn spacecraft for its rendezvous with the main belt asteroids Vesta and Ceres. The mass requirements for this option were calculated using the spacecraft model shown in Table 4.

**Table 4: Solar Electric Propulsion Powered Trojan Orbiter Spacecraft Model**

**Primary Transit Xenon Propellant** 1100 kg

| General Model | Mass, kg. | Mass Comments | Power (W) | Power Comments |
|---|---|---|---|---|
| **Instruments** | **42** | Same as Dawn CBE at Neutral Mass Review | **30** | Dawn |
| **ACS** | **40** | Dawn:4 wheel+star tracker system | **30** | Dawn |
| **C&DH** | **25** | 2 ACE's, 2 CEU's | **70** | Dawn |
| **Comm** | **30** | Dawn: 2 TWTA's | **12** | Dawn |
| **RCS / Prop** | **14** | Dawn: 12 thruster hydrazine | **2** | Dawn |
| **Mech/Structures** | **217** | Scaled from REP Team X Model | **0** | NOTE: power budget for EP ON |
| **Thermal** | **100** | Titan Orbiter SEP Module | **150** | Titan Orbiter SEP Module |
| **Electric Propulsion** | **275** | | **24750** | assumption |
| Xenon Tank | 50 | Scaled using Dawn Tank Mass Fraction (4.5%) | | |
| EP Subsystem | 225 | Discovery Analysis Model: 5 thruster NEXT system | | |
| **EPS** | **291** | | **15** | Team X: power model (250W bus) |
| Electronics | 36 | Dawn EPS | | |
| Solar Array Drive | 8 | Titan Orbiter SEP Module | | |
| Battery | 27 | Dawn Battery | | |
| Solar Array | 220 | Ultraflex Array (120 W/kg) | | |
| **Harness** | **45** | Titan Orbiter SEP Module Harness | **15** | Guess: 1.5% |
| **LV Adapter** | **34** | 1.5% of wet mass, no contingency | | |

| | | |
|---|---|---|
| **Total Dry Mass, CBE** | **1113** | |
| **Contingency** | 334 | 30% contingency |
| **Total Dry Mass w/Contingency** | **1447** | |
| **Propellant, Chemical** | **33** | |
| Chemical Propellant, Nominal | 30 | From Dawn |
| Chemical Propellant Margin | 3 | Dawn margin 10% |
| **Propellant, Xenon** | **1182** | |
| Xenon Propellant, Primary Transit | 1100 | |
| Xenon Propellant, Contingency/Residuals | 77 | Dawn equivalent 7% |
| Xenon Propellant, Orbital Operations | 5 | generous guess |
| **Total Wet Mass w/Contingency** | **2662** | |
| **Total Delivered Mass to Asteroid** | **1562** | |

| | | |
|---|---|---|
| **Total Power, CBE** | **25074** | |
| **Bus Power, CBE** | 324 | |
| **Bus Power w/Contingency** | 421 | 30% contingency |
| **EP Power with Contingency** | 25988 | 5% contingency |
| **Total Power w/Contingency** | **26409** | |

- **= From Dawn Mass Budget**
- **= Team X Design Model**
- **= Model Specific to this Study**

The SEP model is similar to the chemical and REP models but it is less detailed and of lower fidelity. The lower fidelity is acceptable because the SEP architecture utilizes a smaller launch vehicle than the chemical and REP architectures and can accommodate mass growth relatively easily by utilizing a larger medium class launch vehicle. The instruments, ACS, C&DH, and communications mass estimate were again taken directly from Dawn. The EPS electronics and battery masses are derived from the Dawn EPS system and the solar array was modeled as an ultraflex array with triple junction GaAs cells and a specific power of 120 W/kg. The structure and thermal subsystems masses were taken directly from the REP spacecraft model (Table 2) with no modification. This is a rough approximation considered adequate for this study. The harness mass is scaled from previous Team X results and the harness loss of 1.5% is typical for a spacecraft with a high voltage multi-kW power system. The electric propulsion subsystem mass is calculated using a NEXT thruster subsystem model described in previous work.[15] The xenon tank was sized using a 4.5% tank mass fraction.

In addition to the deterministic propellant allocation for orbit insertion and deep space maneuvers, the model included 7% xenon propellant margin to account for worst case flow rate, missed thrust periods, restarts, and residuals. The model also included 5 kg of xenon for orbital operations and 33 kg of chemical propellant for attitude control and other operations. All of these values are consistent with values used for mission planning for the Dawn spacecraft.

The low thrust trajectory optimizer SEPTOP was used to examine a single target at an array power of 25 kW. This option is shown in Figure 10 below.

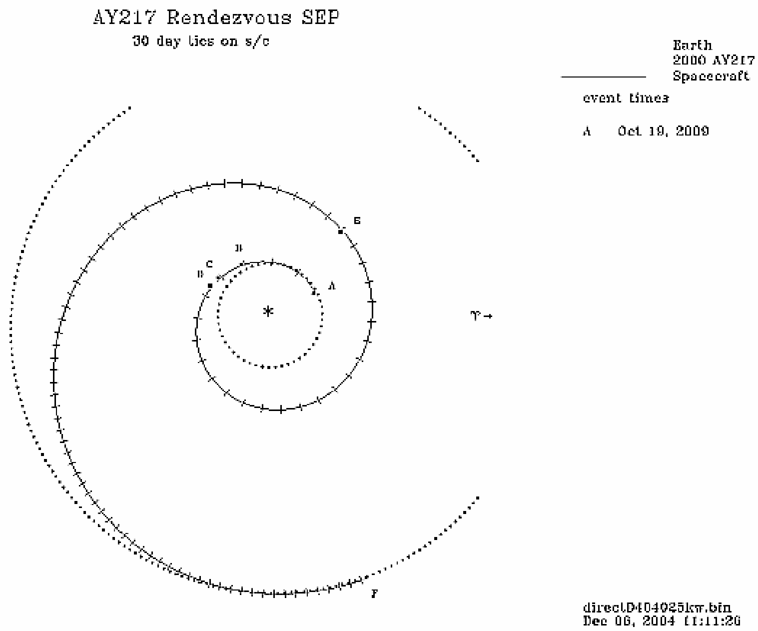



**Figure 10. Solar Electric Propulsion Trojan Orbiter Trajectory**

The spacecraft would be launched on a Delta 4040 launch vehicle to a $C_3$ of 7.5 $km^2/s^2$ and would use NEXT ion engines powered by a 26 kW solar array to travel to asteroid AY217. The total xenon throughput would be very high, over 1100 kg, and five NEXT engines would be required to meet throughput requirements and provide redundancy. The array would generate 900 W of power when the spacecraft reaches the orbit of Jupiter and the size of the array would be driven by the need to operate the thruster at the great distances from the sun. The total power produced at the destination is roughly equivalent to the total power generated by the RPS used in the REP cases. This would enables the use of SEP to accomplish orbit insertion at a distance of 5 AU. The solar array is relatively large, producing 2.5 times more power than the array on the Dawn spacecraft, and the configuration must also accommodate five 40 cm diameter ion thrusters. It is therefore concluded that

- All-SEP Architecture for a Trojan asteroid orbiter is marginally feasible, requiring very high power and xenon throughput.

In addition to requiring a large solar array and multiple thrusters, the SEP architecture is not easily extensible to Centaur objects and other destinations located farther away from the sun. Future work should consider the use of an advanced low power Hall thruster in this application. The combination of higher thrust and lower minimum power requirements may improve system performance for this mission application.

### *SEP-Chemical Architectures*

An alternative to the SEP-only architecture is a combined SEP-chemical architecture in which SEP is used for the heliocentric transit, but chemical propulsion is used for orbit insertion and maneuvering near the asteroid. By replacing SEP with chemical propulsion when far away from the sun, this architecture would in principal allow use of a smaller solar array and EP system. However, the architecture does require a large chemical maneuver, approximately 5 km/s for SEP trajectories with an Earth gravity assist, and a small RPS is needed to provide power for the science instruments. The trajectory optimization program SEPTOP was used to conduct a brief survey of

SEP-chemical architectures with solar array sizes between 6 and 15 kW (1 AU). The survey examined non-optimal combinations of SEP and chemical propulsion because SEPTOP cannot directly do a joint optimization of both the chemical and SEP portions of the trajectory.

No feasible SEP-chemical mission options were found between 6 and 15 kW. However, since a viable all-SEP solution exists at 25 kW, is seems possible that there may be a feasible SEP-chemical option between 15 kW and 25 kW. The trajectory survey was conducted quickly, and further work is needed to draw definitive conclusions on the viability of a SEP-chemical architecture for a Trojan orbiter mission.

*2. Trojan Orbiter Mission Cost*

Overall mission cost estimates were generated for three of the Trojan orbiter options described above: an REP option, a SEP-only option, and a chemical propulsion option. The primary cost estimating tool for these options was the NASA/Air Force Cost Model (NAFCOM) and all estimates are presented in FY04$M. All options are costed assuming a September 2009 launch date and the spacecraft subsystems are assumed to be a current representative TRL levels. The payload was assumed to be identical to the Dawn spacecraft payload and the launch vehicle costs were calculated in a manner consistent with guidelines provided in the 2003 New Frontiers announcement of opportunity. The RPS cost estimate included $11M for NEPA nuclear compliance, nuclear launch safety approval, emergency preparedness, risk communication and spacecraft accommodation/integration as well as $15M per flight unit. 25% cost contingency was included in phase B/C/D costs and 15% contingency was included in phase E costs, but no reserves were included in the cost of the RPS or the launch vehicle. Based on these assumptions, the calculated cost for the Trojan Orbiter mission are shown in Figure 11.

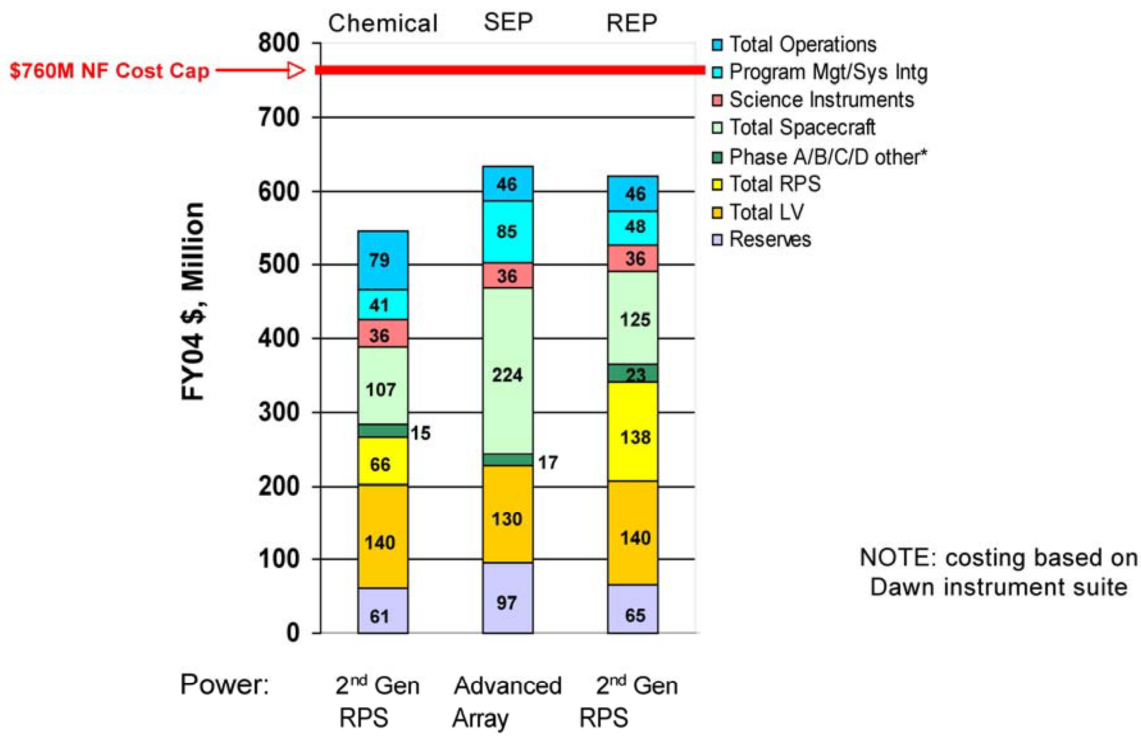



**Figure 11. Trojan Asteroid Orbiter Mission Cost Estimate**

In all cases, the estimated cost exceeded the Discovery mission cost cap but was well within the New Frontiers cost cap adjusted to FY04 dollars. The results showed that chemical missions have the lowest overall cost, but as discussed previously, a chemical system captures on a small percentage of the possible Trojan asteroid destinations. REP missions cost more, but capture a much larger percentage of the possible Trojan asteroid destinations. SEP missions have the greatest cost because of the relatively large solar power system required to do maneuvering at the destination asteroid.

*3. Trojan Orbiter Analysis Summary*

A series of different mission architectures have been analyzed with the goal of putting a small scientific payload into orbit around a Trojan asteroid. Several different architectures using combinations of REP, SEP, and chemical propulsion were considered as a means to meet the mission objectives. Feasible and infeasible options were identified by comparing mass delivery capability to mass requirements derived from subsystem level spacecraft mass models. The results of the analysis are summarized in Table 6.

In Table 6, feasible architectures are defined as architectures that can deliver a Dawn-like instrument payload to orbit around a Trojan asteroid within the New Frontiers cost cap. Technology development costs for the RPS and EP subsystem are not considered in this chart. The instrument payload assumed in this study is a "science floor" payload derived from a Discovery mission that may not provide enough science to justify a New Frontiers class mission.

Based on the analysis presented in this section, the following conclusions have been reached.

- REP with 2nd Generation RPS enables a wide range of targets for a Trojan asteroid orbiter compared to chemical propulsion, and can substantially lower trip time requirements in some cases.
- REP with 1st Generation RPS is marginal for a Trojan asteroid orbiter. With a customized lightweight spacecraft, it is probably feasible to reach a limited number of destinations, but a customized spacecraft may be too expensive for cost-capped mission applications.
- All-SEP Architecture is marginally feasible for a Trojan asteroid orbiter, requiring very high power and xenon throughput.

**Table 5: Trojan Orbiter Mission Feasibility**

| Architecture | Trojan Asteroid Mission Feasibility |
|---|---|
| Chemical Direct | Infeasible |
| Chemical with JGA | Feasible to limited number of targets |
| SEP-Chemical | No feasible solutions found - Further work needed |
| REP, Generation 1 | Marginally feasible to limited number of targets |
| All-SEP | Feasible to unknown number of targets with very large solar array |
| REP, Generation 2 | Feasible to most targets |

Red = infeasible
Yellow = feasible to limited range of targets
Blue = possibly feasible (known issues with spacecraft configuration)
Green = probably feasible

In addition to requiring a large solar array and multiple thrusters, the SEP architecture is not easily extensible to Centaur objects and other destinations located farther away from the sun. The work done in this study assumed the use of the NEXT ion thruster for SEP missions. Future work should consider the use of an advanced low power Hall thruster in this application as the combination of higher thrust and lower minimum power requirements may improve system performance for this mission application.

- Further work is needed on SEP-Chemical architecture

No feasible SEP-chemical mission options were found between 6 and 15 kW. However, since a viable all-SEP solution exists at 25 kW, it is possible that there is a viable SEP-chemical option between 15 kW and 25 kW. The trajectory survey was conducted quickly, and further work is needed to draw definitive conclusions on the viability of a SEP-chemical architecture for a Trojan orbiter mission.

It should be noted that this study did not consider the use of the Star 48 upper stage to improve launch vehicle performance. Use of such a stage may increase mass performance on many of the mission options considered here.

Based on these findings, we have identified two technologies that could be considered enabling for the majority of Trojan asteroid orbiter targets. These technologies are:

- An advanced RPS with a specific power of approximately 6 W/kg or greater.

Since very few Trojan asteroid targets are enabled by the 1$^{st}$ generation RPS (4 W/kg) and virtually all of the Trojan asteroids are enabled by the 2$^{nd}$ generation RPS (8W/kg), it is reasonable to assume that an intermediate value of approximately 6 W/kg will enable most of the Trojan asteroid targets of scientific interest. Further work is needed to verify that this value is sufficient to enable most targets.

- An advanced low power Hall Thruster with xenon throughput capability > 300 kg

The optimum specific impulse for the REP Trojan asteroid orbiter appears to be between 1500 seconds and 2000 seconds, within the Hall thruster's range of operation. However, the total xenon throughput required for these missions is well beyond the state of the art for Hall thrusters operating below 1 kW input power. A low power Hall thruster with a throughput > 300 kg is needed to support REP Trojan asteroid orbiter.

## B. Jupiter Polar Orbiter with Probes Mission

The Jupiter Polar Orbiter with Probes (JPOP) mission is cited in the Decadal Solar System Exploration Survey (DSSES) as the highest priority mission for giant planet research.[16] The mission's objective is to gain a better understanding of Jupiter's strong magnetic and gravity fields and its deep atmosphere. To meet this objective, the spacecraft is to deliver multiple atmospheric entry probes that can penetrate to the 100 bar pressure level and sample a range a latitudes within 30 degrees of the equator. To avoid the highest-flux parts of the Jovian radiation field, a very low perijove, < 1.1 $R_J$, is necessary. The orbiter is expected to remain in orbit for at least one year.

### *1. Transportation Approach*

A launch vehicle that is compatible with the New Frontiers (NF) cost-cap, the Atlas 551, is selected to provide the initial $\Delta V$ to place the spacecraft on a hyperbolic trajectory relative to Earth. This analysis reduced the performance of the Atlas V 551 by 10% in order to be consistent with prior work.[17]

#### *Radioisotope Powered Electric Propulsion*

After separating from the launch vehicle, the REP spacecraft thrusts until the required $\Delta V$ is obtained for a direct transfer to Jupiter. There are three conceivable REP architectures:

(1) All REP—REP for the heliocentric transfer and to spiral in to the final orbit;

(2) REP flyby/chemical—REP for the heliocentric transfer to Jupiter's sphere-of-influence (JSOI) and a (large) chemical stage for the orbit insertion maneuver;

(3) REP rendezvous (RV)/chemical—REP for the heliocentric transfer and RV and a (small) chemical stage for the final orbit insertion maneuver.

The all-REP option would be preferred option because it would be less complex (and therefore less costly) than hybrid architectures. Employing the use of a chemical stage would only be assessed if the payload requirements could not be met with the all-REP option in the transfer times of interest. Using the REP system to rendezvous with Jupiter would substantially decrease the mass of the chemical stage; however, the trade-off would be a substantial increase in the wet mass of the REP stage. This is why, from a mission performance point-of-view, the REP flyby/chemical option offers superior performance to the REP RV/chemical option. Therefore, if a chemical stage is necessary to meet the payload requirements, only the REP flyby/chemical option will be assessed. Additionally, the extra performance benefit of the Star 48V upper stage solid rocket motor is assessed.

### *Solar Electric Propulsion*

The SEP spacecraft utilizes a Venus gravity assist (VGA) to provide increased heliocentric orbital energy en route to Jupiter. A 15-kW (BOL, 1 AU) solar array provides power to two operating advanced state-of-the-art (ASOA) ion thrusters. A generic mass model of the entire SEP module was utilized. This mass model was based on the Titan Orbiter Team X Study.[18] After the propellant load has been determined, the mass of the entire SEP module is estimated with this generic model. The SEP module is jettisoned around 2 AU after the required $\Delta V$ is obtained. The spacecraft then coasts to Jupiter. A SOA chemical stage performs the maneuver to capture the spacecraft into the final parking orbit. An extra 5% is added to this required $\Delta V$ to account for margin and gravity losses.

### *State of the Art Chemical*

Because Jupiter is a relatively easy target with respect to the required launch energy (a Hohmann $C_3$ of 77.3 $km^2/s^2$ translates to a flight time of 2.73 years), direct transfers can offer satisfactory performance depending on the required payload. Ballistic trajectories utilizing multiple gravity assists can offer increased mass delivery capability. The use of the Star 48V upper stage solid rocket motor would also likely increase performance for ballistic transfers, particularly for direct transfers; however, this study did not assess this benefit for the ballistic transfers.

## *2. Mission Assumptions*

As sort of a "science-floor", this study assumed a 272 kg payload is required in the final orbit. This payload consists of 55 kg of science instruments and three mini-probes totaling 217 kg. For comparison, the mass of the Galileo probe was ~ 340 kg and was required to penetrate to about 10 – 20 bars of atmospheric pressure.[19,20] The probes on this mission are required to withstand pressure at least five times greater. Clearly, the total probe mass that this study assumes is most likely severely undersized; however, one must keep in mind that the objective of this study was to determine the *potential* applicability of REP for various missions.

To be consistent with prior JPOP mission analysis, a highly eccentric 7,149 km alt x 30-day orbit was targeted.[18] The $C_3$ of this orbit (relative to Jupiter) is – 45.52 $km^2/s^2$. The REP spacecraft must spiral down to a circular orbit with this same energy level; the performance (including the spiral time) is assessed by VARITOP[21]. For a chemical insertion, the instantaneous $\Delta V$ is assumed to occur at perijove.

The specific impulse for the 100-$lb_f$ bi-propellant (NTO/$N_2H_4$) SOA chemical system was assumed to be 325 seconds. The mission analysis work that was done previously showed that a SOA all-chemical transportation option is likely to offer better performance for trip times greater than four years.[18] The performance of representative SOA chemical transfers is shown in Figure 12. This figure shows that the SOA chemical option offers a wide range of mass delivery capability; each additional gravity assist results in a larger payload. Two important caveats go along with Figure 12. The first caveat is that the fidelity of the model has increased since the time that this analysis was performed. The second caveat is that the gravity assist trajectories occur in the 2003 – 2009 (launch) time frame. As a consequence of this, this chart demonstrates *trends only* that are likely to occur in a later time frame. Obviously, there would be some shifting of the data due to better modeling and a more feasible time frame.

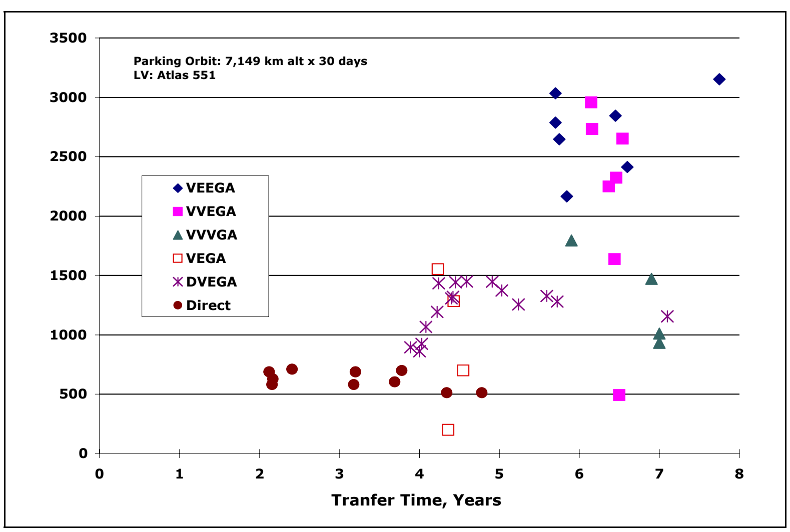



**Figure 12. Representative SOA chemical transfers to Jupiter**

Because of the good performance trends exhibited by the SOA all-chemical transportation option for transfer times greater than four years, a transfer time of less than four years was desired; this transfer time includes the spiral time.

### *3. JPOP Analysis Results*

Figure 13 shows the performance results for a ~ 3.9-year transfer to the final orbit. Clearly, the delivered mass capability of the REP spacecraft is substantially less than the payload requirement for all three generations of the RPS.

Table 7 provides the detailed performance summary results and shows that a ~ 9 month spiral is required and that the optimal specific impulse is roughly 1200 seconds.

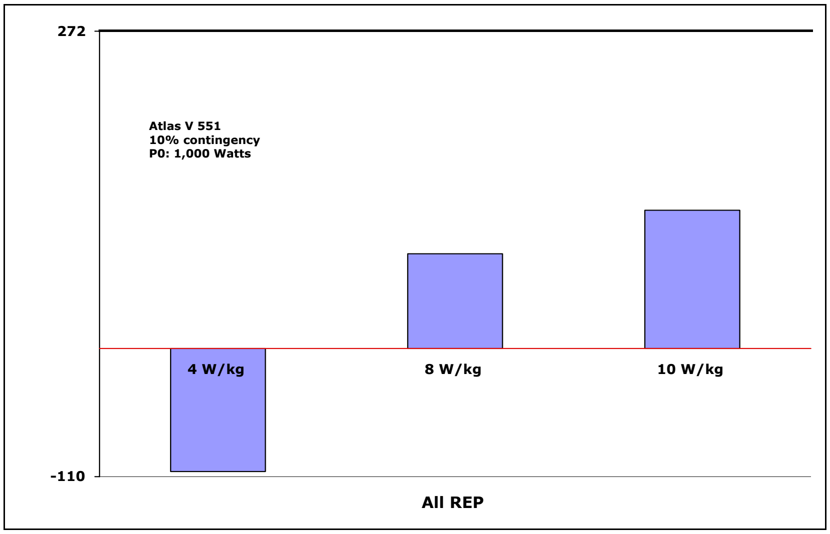


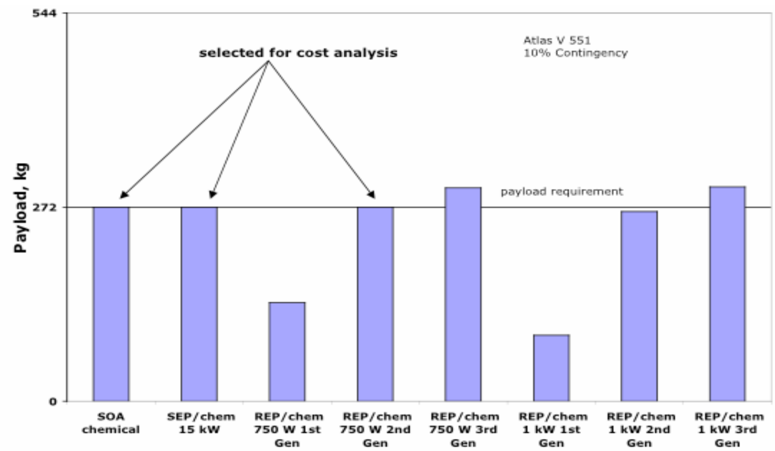

**Figure 13. Delivered mass for all REP transfer to Jupiter using 1st, 2nd, and 3rd generation RPS**

**Table 7. Performance summary results for all REP transfer to Jupiter**

| JPOP Mission Results Summary | All REP | All REP | All REP |
|---|---|---|---|
| Power (Watts) | 1000 | 1000 | 1000 |
| alpha (W/kg) | 4 W/kg | 8 W/kg | 10 W/kg |
| | | | |
| Launch Vehicle | Atlas V 551 | Atlas V 551 | Atlas V 551 |
| Launch Vehicle Contingency | 10% | 10% | 10% |
| | | | |
| C3 ($km^2/s^2$) | 77.8 | 77.8 | 77.8 |
| | | | |
| Earth Departure Date | 14-Sep-2013 | 14-Sep-2013 | 14-Sep-2013 |
| Jupiter Arrival Date | 20-Oct-2016 | 20-Oct-2016 | 20-Oct-2016 |
| Final Oribit Arrival Date | 1-Aug-2017 | 1-Aug-2017 | 1-Aug-2017 |
| | | | |
| m0 (kg) | 1171 | 1171 | 1171 |
| Heliocentric mp (kg) | 578 | 578 | 578 |
| Mass @ Jupiter (C3 = 0 $km^2/s^2$) (kg) | 593 | 593 | 593 |
| | | | |
| Capture Method | REP Spiral | REP Spiral | REP Spiral |
| Target Orbit C3 ($km^2/s^2$) | -45.52 | -45.52 | -45.52 |
| Rp (km) | 2783402 (~ 39 $R_J$) | 2783402 (~ 39 $R_J$) | 2783402 (~ 39 $R_J$) |
| Ra (km) | 2783402 (~ 39 $R_J$) | 2783402 (~ 39 $R_J$) | 2783402 (~ 39 $R_J$) |
| Capture Time (years) | 0.78 | 0.78 | 0.78 |
| Capture mp (kg) | 145 | 145 | 145 |
| | | | |
| Xe Deterministic Total (kg) | 723 | 723 | 723 |
| Xe Contingency | 10% | 10% | 10% |
| Total Xe with Contingency (kg) | 795 | 795 | 795 |
| | | | |
| **Mass after Capture (kg)** | **376** | **376** | **376** |
| | | | |
| Propulsion System Dry Mass (kg) | 83 | 83 | 83 |
| Propulsion System Mass Contingency | 30% | 30% | 30% |
| Total Propulsion System Mass (kg) | 108 | 108 | 108 |
| | | | |
| RPS Mass (kg) | 287.2 | 143.6 | 114.9 |
| RPS Mass Contingency | 30% | 30% | 30% |
| Total RPS Mass w/Contingency (kg) | 373 | 187 | 149 |
| | | | |
| **NDM (kg) w/o power & propulsion** | **-106** | **81** | **118** |
| | | | |
| Heliocentric $\Delta V$ (km/s) | 8.21 | 8.21 | 8.21 |
| Capture $\Delta V$ (km/s) | 3.39 | 3.39 | 3.39 |
| Total $\Delta V$ (km/s) | 11.60 | 11.60 | 11.60 |
| | | | |
| Heliocentric Transfer Time (yrs) | 3.10 | 3.10 | 3.10 |
| **Total Transfer Time (yrs)** | **3.88** | **3.88** | **3.88** |
| Total Propulsion Time (yrs) | 3.88 | 3.88 | 3.88 |
| **Optimal Isp (seconds)** | **1231** | **1231** | **1231** |

Because the REP spacecraft could not deliver the required payload in less than four years, a chemical stage was added to spacecraft architecture. After arriving at the JSOI, the SOA chemical stage performs the insertion maneuver. The performance of the REP/Chem relative to the SOA chemical and SEP is shown in Figure 14. The optimal specific impulses for the REP systems were nearly 1600 seconds.

Figure 14 shows that the SOA chemical option delivers the required payload in the shortest time. The second and third generation RPS enables the payload requirement to be met for the REP flyby/chem option. Also, note that the 750-Watt spacecraft slightly outperforms the 1,000-Watt spacecraft. A representative mass estimate for the REP spacecraft utilizing the second generation RPS is shown in Table 8. A representative trajectory is shown in Figure 15.




| | SOA chemical | SEP/chem 15 kW | REP/chem 750 W 1st Gen | REP/chem 750 W 2nd Gen | REP/chem 750 W 3rd Gen | REP/chem 1 kW 1st Gen | REP/chem 1 kW 2nd Gen | REP/chem 1 kW 3rd Gen |
|---|---|---|---|---|---|---|---|---|
| Trip time (yrs) | **2.2** | 3.1 | 3.8 | 3.8 | 3.8 | 3.8 | 3.8 | 3.8 |
| Trajectory: | Direct | VGA | Direct | Direct | Direct | Direct | Direct | Direct |
| $C3$ ($km^2/s^2$): | 85.6 | 38.2 | 66.9 | 66.9 | 66.9 | 66.2 | 66.2 | 66.2 |

**Figure 14. Performance comparison results**

**Table 8. Example REP Flyby/Chem mass estimate (all masses in kg)**

| Item | Mass | Subtotal | Total |
|---|---|---|---|
| **REP Dry** | | | **804.6** |
| Science Instruments | 271.9 | | |
| Telecom & C&DH | 68.9 | | |
| Power | 144.3 | | |
| PMAD & Cabling | 51.7 | | |
| Propulsion | 64.8 | | |
| ACS w/instruments | 10.4 | | |
| Thermal | 15.4 | | |
| Structures | 177.2 | | |
| Propellants | | 171.3 | |
| ACS | 5.4 | | |
| Xe | 155.0 | | |
| Errors, Reserves, Residuals | 10.9 | | |
| **REP Wet** | | | **975.9** |
| **Chemical Dry** | | | **142.0** |
| Thermal | 8.8 | | |
| Structures | 39.7 | | |
| Propulsion | 18.2 | | |
| Tank & Propellant Mgt. | 17.1 | | |
| Components | 58.2 | | |
| Propellants | | 298.9 | |
| NTO/$N_2H_4$ | 252.0 | | |
| ACS | 25.4 | | |
| Helium Pressurant | 1.0 | | |
| Residuals, Margin | 20.5 | | |
| **Chem Wet** | | | **440.9** |
| LV adapter | | | 17.7 |
| Spacecraft Dry | | | 964.3 |
| **Spacecraft Wet** | | | **1434.5** |
| $C_3$ (km2/s2): | | | 69.9 |
| m0 @ $C_3$: | | | 1435 |
| Launch mass margin: | | | 0.5 |

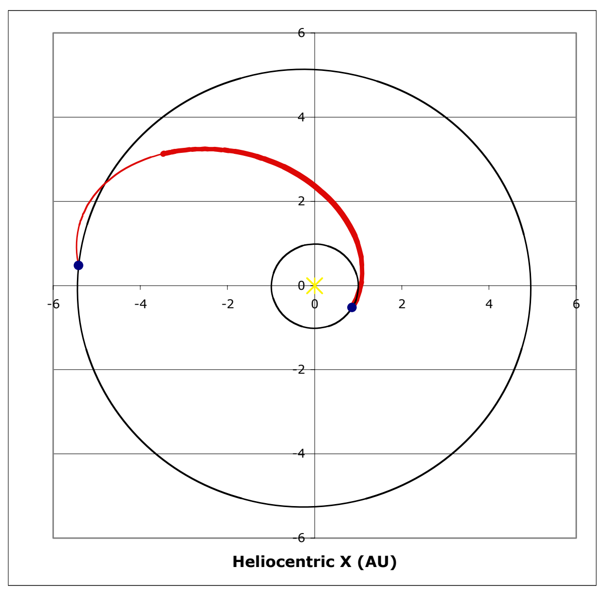



**Figure 15. Representative REP Flyby trajectory (bold curve depicts thrust phase)**

Using the Star 48V enables the first generation RPS to deliver the required payload, and increases the mass delivery capability of the REP spacecraft powered by the second and third generation RPSs. The Star 48V would only be beneficial for launch energies in excess of ~ 35 $km^2/s^2$.[22] As stated earlier, the SOA chemical trajectories would also benefit (substantially) from its use.

A ΔV budget comparison for each transportation option is shown in Table 9 (the Atlas V 551 without the Star 48V). One can see that the REP flyby/chem option resembles more of a ballistic option that an EP option—most of the ΔV is performed by the launch vehicle. The transfer times for the REP flyby/chem option can be substantially reduced ( > 1 year), but what this really means is that the ΔV performed by the launch vehicle is increased and the post-launch ΔV performed by the REP system is reduced. This result further demonstrates the fact that Jupiter is a relatively easy target for SOA chemical.

**Table 9. Delta-V Budget Comparison**

| | | | |
|---|---|---|---|
| Transfer time (yrs): | 2.2 | 3.8 | 3.1 |
| Transportation option: | SOA Chemical | REP flyby/chem | SEP flyby/chem |
| Launch vehicle (km/s): | 6.60 | 6.04 | 4.84 |
| EP or deep space ΔV (km/s): | 0.0 | **1.78** | **6.42** |
| Insertion [w/o g-losses] (km/s): | 0.749 | 0.672 | 1.676 |
| 5% ΔV for g-losses & margin (km/s): | 0.0374 | 0.0336 | 0.0838 |
| Total post-launch ΔV (km/s): | **0.79** | **2.49** | **8.18** |

The potential performance benefit of adding a Star 48V upper stage to the Atlas V 551 was assessed and is shown in Fig. 15.

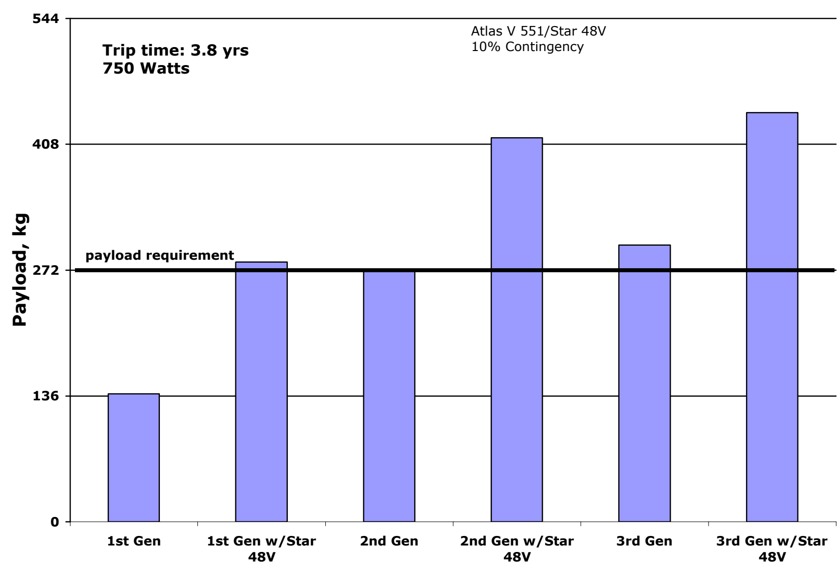



**Figure 16. REP Flyby/Chemical performance results when utilizing the Star 48V.**

### 4. *JPOP Mission Cost*

An absolute cost comparison is shown for the three transportation options identified in Figure 14. The cost analysis results are shown in Figure 17. Only the SOA chemical option fits within the NF cost cap; this option even offers room for cost growth.

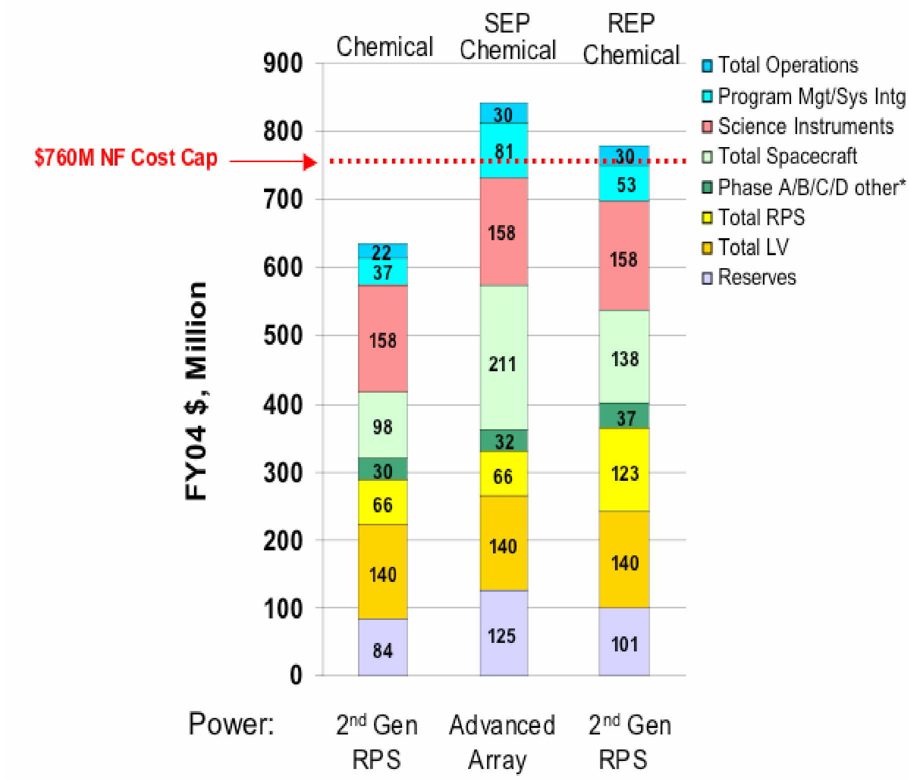



**Figure 17. Absolute cost comparison**

*5. JPOP Analysis Summary*

The performance of spacecraft utilizing REP technology was assessed for a JPOP mission. The results were compared with SEP and SOA chemical. For the total transfer time desired (less than 4-yrs), an all-REP transportation option could not meet the payload requirement of 272 kg. To meet the payload requirement, a chemical stage was added to the REP spacecraft configuration. For this hybrid configuration, the spacecraft powered by a second or third generation RPS delivered the required payload. The 1,000-Watt spacecraft showed no benefit relative to the 750-Watt spacecraft. Adding the Star 48V upper stage enabled the payload requirement to be met with a first generation RPS. However, this hybrid (REP flyby/chem) architecture resembles more of a ballistic option than an EP option when considering how much $\Delta V$ is actually performed by the EP system.

Longer heliocentric transfer times for the all-REP spacecraft would likely result in meeting the payload requirement (for the second or third generation RPS); however, the spiral time would also increase.

The cost analysis that was performed for the three transportation architectures showed that the SOA chemical option is the least costly and the only one that fits within the NF cost cap. Additionally, the SOA chemical option delivered the required payload in the shortest time. Based on these results, the transportation option that shows the best applicability for this mission is the SOA chemical option. This option can also accommodate mass growth (more massive probes) and cost growth.

## C. Comet Surface Sample Return Mission

Electric propulsion applications to comet missions have been studied in several instances in the past.[23,24] In particular, NASA's Next Generation Electric Propulsion NEXT thruster has been looked at extensively for deep space applications, including for a comet sample return mission[25]. Other studies have investigated the use of ion propulsion for various outer planet and Mars missions.[26,27,28] The objective of this analysis is to assess the efficacy of a radioisotope electric propulsion system (REPS) for performing a comet surface sample return (CSSR) mission.

The assessment consists of comparing REPS cost and performance with solar electric propulsion system (SEPS) and SOA chemical propulsion system (SCPS) cost and performance.

### *1. Mission Assumptions*

The mission framework chosen to facilitate technology comparisons assumes a comet surface sample return mission to the comet Tempel 1. Comet Tempel 1 was chosen as the target because is a well known comet of intermediate difficulty. This comet has been studied extensively in the past and was chosen as the destination of NASA's Deep Impact mission. The Earth departure epoch was chosen as 2008, leading to an approximately 8 year mission ending in 2016 for Earth return. The stay-time at the comet was chosen as 60 days, allowing time for comet sample capture and close-up science observations of the comet. See Figure 18 for a summary of mission assumptions.

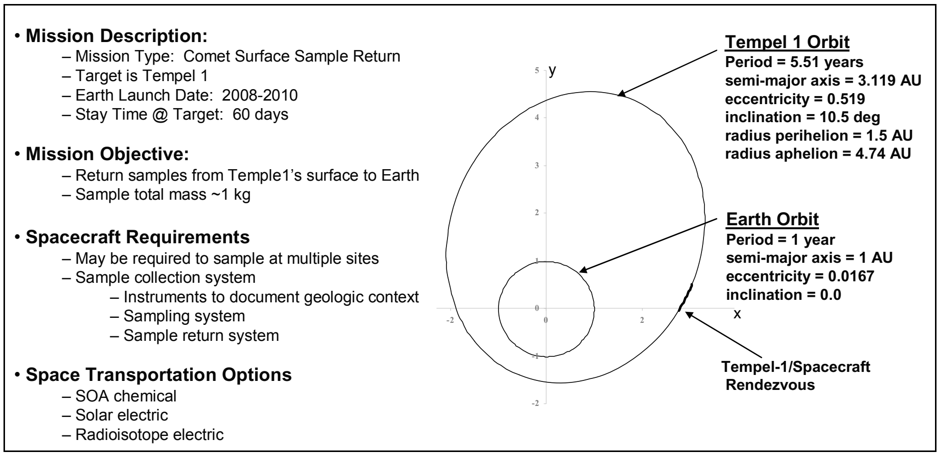



**Figure 18. CSSR Mission Description**

The primary mission objective was to return a sample from Temple 1's surface to Earth using a direct entry capsule. The total sample mass was assumed to be approximately 1 kg. The assumed sample collection system includes instruments to document geologic context, a sampling system, and a sample return system (capsule for sample return at Earth). Each of the propulsion system technologies investigated will be compared by assuming that each system completely performed the above outlined mission from end-to-end.

### *2. Systems Assumptions*

This section delineates the system assumptions that were made for the propulsion technology cases investigated in this study. The technology cases that were investigated include SOA chemical bi-propellant propulsion, NEXT based solar electric propulsion, and advanced radioisotope electric propulsion, as summarized in Figure 19. In all cases, it is assumed that the total science payload mass delivered by each propulsion system is 141 kg. The detailed system assumptions for chemical, SEPS, and REPS are provided in **Error! Reference source not found.**.

SCP is based on historical AXAF and Cassini spacecraft data with power based on radioisotope power systems models from GRC. The chemical propulsion system is a bipropellant NTO/$N_2H_4$ system with an $I_{sp}$ of 325 sec. For the CSSR mission, one 445 N (100 lbf) engine was sufficient to perform the mission. Reserves, residuals, and other contingencies were 5%, 3%, and 2% respectively. Spacecraft power was assumed to be three 2$^{nd}$ generation RPS's with a total beginning of life power of 350 W. SOA chemical structures were based on various historical spacecraft and other models such as the thermal control physics based model. Thermal control was assumed to be SOA heatpipe and MLI. ACS was assumed to be monopropellant hydrazine. As will be indicated in an upcoming section of this study, the mass of the chemical propulsion systems required to perform this mission is greater than the capability of any current launch vehicle.

| Spacecraft System Case | Propulsion | Baseline Payload[1] | Spacecraft Systems Model Heritage |
|---|---|---|---|
| **Chemical** | SOA Chemical | 141 kg | **Propulsion**: AXAF and Cassini<br>**Power**: GRC RPS model<br>**Structures**: Various historical S/C<br>Some physics based models |
| **REPS** | Radioisotope Electric | 141 kg | **Propulsion**: GRC model<br>**Power**: GRC RPS model<br>**Structures**: Near & Messenger<br>Some physics based models |
| **SEPS** | Solar Electric | 141 kg | **Propulsion**: NGEP & DS-1<br>**Power**: Advanced Solar Array<br>**Structures**: Various historical S/C<br>Some physics based models |

(1) Payload consists of 3 cameras, sampling system, and sample return system

**Figure 19. Technology Cases Investigated in CSSR Study**

| Spacecraft System Case | Launch Vehicle | Propulsion | Power | Other |
|---|---|---|---|---|
| **Chemical** | Infeasible | SOA NTO/$N_2H_4$ Liquid Apogee Engine<br><br>Isp = 325 sec<br><br># engines = 1 | 3 RPS's<br>350 W<br>(2nd Gen) | **Tank** mass fraction =<br>**ACS** = SOA hydrazine<br>**Thermal** = radiators, heatpipe and MLI<br>**Propellant Contingencies**<br>- reserves = 5%<br>- residuals = 3%<br>- other = 2% |
| **REPS** | Atlas 401 | GRC radioisotope electric propulsion model:<br><br>1 propellant management system<br><br>2 PPU's with 1 for redundancy.<br><br>Advanced Xe ion thrusters and 1 spare, with throughput limit = 300 kg Xe | 750 W<br>Multiple RPS's<br>(1st, 2nd, 3rd Gen) | **Xe tank** mass fraction = 4.5% of total propellant.<br>**ACS** = SOA hydrazine<br>**Thermal** = radiators, heatpipe, and MLI<br>**Total Propellant Contingencies** = 7% of deterministic Xe |
| **SEPS** | Delta 4040 | NEXT base SEP,<br><br>1 propellant management system<br><br>2 operational PPU's, 1 spare<br><br>2 operational NEXT thrusters, 1 spare, throughput limit = 300 kg Xe | 14.3 kWe EOL<br>GaAs Multi-junction Array,<br>alpha=150 W/kg | **Xe tank** mass fraction = 4.5% of total propellant.<br>**ACS** = SOA hydrazine<br>**Thermal Control** = radiators, heatpipe and MLI<br>**Propellant Contingencies** = 8.6% of deterministic Xe |

**Figure 20. CSSR System Assumption Summary**

REPS spacecraft propulsion was based on a GRC Xe based propulsion model that included mass estimates for propellant management, thrusters, and power processing units. It was assumed that there were 2 PPU with one for redundancy. The number of Xe thrusters was chosen to adequately provide the total mission throughput, with maximum throughput of an advanced Xe thruster assumed to be 300 kg of Xe. One spare thruster was assumed to provide sufficient mission reliability. For power, a GRC RPS model was assumed. NEAR and Messenger structures models were also assumed. Thermal control for the REPS was based on physics models. The Xe tank fraction was set at 4.5% for the total propellant load. Thermal control was assumed to be SOA heatpipe and MLI. The total Xe propellant contingency was assumed to be 7% of the total deterministic Xe propellant.

SEPS models were based on historical data from DS-1 and estimated systems mass data from the NASA's Next Generation Electric Propulsion program. The propulsion system was assumed to have 2 operational thrusters and 1 spare thruster, with maximum throughput of 300 kg of Xe. Likewise, there was assumed to be 2 operational PPU's and 1 spare PPU. The propellant management system is assumed to be NEXT based. The SEPS power system was assumed to be based on an advanced GaAs multi-junction solar array model[29] with an array alpha of 150 W/kg[30]. Structures and thermal systems were based on various historical and physics based models. The Xe propellant tank mass fraction was assumed to be 4.5% of the total propellant load, including all contingencies. The propellant contingencies were assumed to be a total of 8.6% of the deterministic propellant load. As in previous propulsion system models in this study, the thermal control is assumed to be SOA heatpipe and MLI. ACS is assumed to be SOA monopropellant hydrazine based.

### *3. CSSR Analyses Results*

In this section, analyses results are presented for the REPS, SEPS, and SCP technologies. For each technology investigated, a parametric study was performed to determine the minimum launch vehicle that could perform the mission. Further, trajectories were optimized for each technology that provided the reference payload of 141 kg to the appropriate destinations with the minimum total mass requirements and within the minimum total transfer time.

#### *Chemical Propulsion*

The SCPS mission analyses results are summarized in Figure 21. This chemical propulsion system transportation scenario turns out to be infeasible for any current launch vehicle choice. The optimal case found to-date for this mission/technology combination would require the launch vehicle to inject the spacecraft to an optimal $C_3$ of 48 km$^2$/sec$^2$. A deep space maneuver of 1650 m/s would subsequently be required during the 1st leg of the mission. After this maneuver, a Venus gravity assist and a relatively small deep space maneuver of 74 m/s would set the stage for a comet capture maneuver of 3200 m/s. The sample retrieval system would then collect the sample from the comet surface. After the sample is collected, 50 kg of sample collection mass is assumed to be left at the comet. At the 60 day stay time limit, a 2000 m/s comet departure maneuver is performed to begin the Earth return journey. A

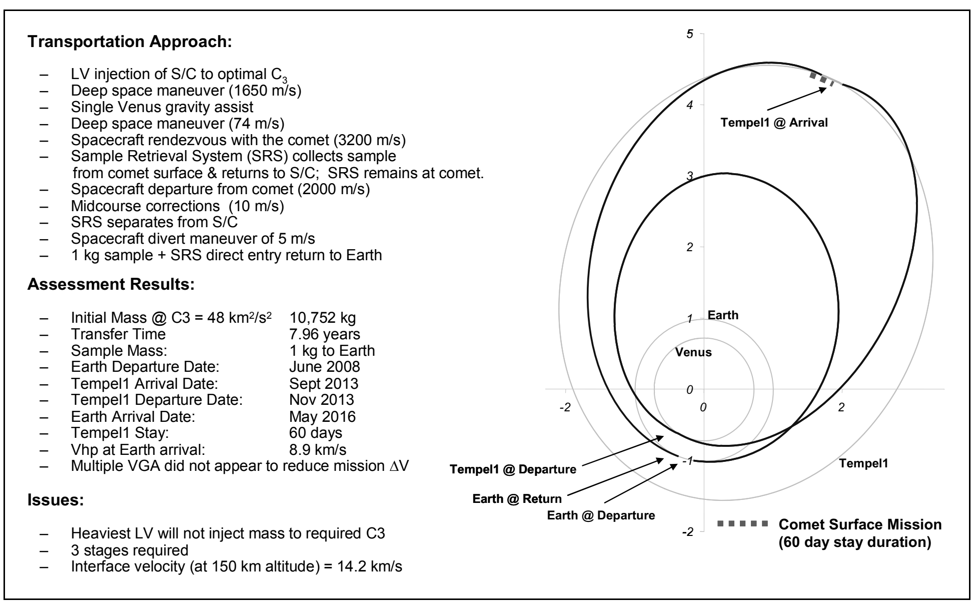



**Figure 21. Chemical Propulsion Mission Analysis Summary**

small midcourse correction of 10 m/s is assumed for the return phase, with the sample return system separating from the primary spacecraft some days before Earth arrival. The primary spacecraft performs a divert maneuver that safely forces an Earth flyby of the primary spacecraft. The 1 kg sample is concurrently forced to enter the Earth's atmosphere via a direct entry capsule based system. Total transfer time for the chemical propulsion based mission was 7.96 years.

There are two primary issues associated with the chemical propulsion based mission. First the spacecraft required to perform the total mission delta-v of nearly 7000 m/s optimized to 3 stages with a total mass of over 10,000 kg. This mass was larger than any existing launch vehicle could inject to the required $C_3$ of 48 km$^2$/sec$^2$. Second, it should be noted that the hyperbolic excess velocity of the spacecraft at earth return was approximately 8.9

km/s. This excess speed translates to an Earth entry interface speed of over 14 km/s. This entry interface velocity is significantly more than the NASA Stardust mission entry interface velocity.

## *Solar Electric Propulsion*

The SEPS mission analyses results are summarized in Figure 21. For the SEPS transportation scenario, the launch vehicle[31], a Delta 4040, injects the spacecraft to an optimal $C_3$ = 18.6 km$^2$/sec$^2$ at an initial mass of 1640 kg. The SEPS system and its 141 kg of science payload then slowly spiral out to rendezvous with comet Tempel 1. This outbound spiral requires over 1.5 revolutions around the sun. After rendezvous, the spacecraft remains at Tempel 1 for 60 days, during which the sample retrieval system gathers the sample and deposits it in the return capsule. 50 kg of retrieval system remains at the comet, and the SEPS vehicle begins Earth return maneuvers. There is a long powered return with some weeks of coast before Earth encounter. The direct entry sample return capsule along with its 1 kg of comet sample is released with a $V_{hp}$ of over 9.8 km/sec and the primary SEPS vehicle performs a divert maneuver to miss the Earth, going into orbit about the sun. The capsule performs a direct entry at Earth interface, returning the comet sample to Earth. The total transfer time for the SEPS mission is approximately 8.1 years with departure in May 2008 and a return to Earth in June 2016. One issue related to this scenario is the rather large Earth entry interface condition of approximately 15 km/sec. Further, the data provided in Figure 23 indicates the SEPS performance over a range of total solar array power. The baseline trajectory previously mentioned is indicated on this figure at a power level of 14.3 kW. It can be seen that this array power level provides some margin in mass delivery.

**Transportation Approach:**

- LV injects S/C to optimal $C_3$
- Spiral away from Earth orbit and match comet orbit
- SEPS rendezvous with Tempel 1
- Sample Retrieval System (SRS) collects sample from comet surface & returns to S/C; SRS remains at comet
- SEPS spacecraft returns to Earth for flyby
- Sample Return System separates from S/C
- Sample direct entry return to Earth
- SEPS performs Earth divert maneuver

**Assessment Results:**

| | |
|---|---|
| – Launch Vehicle | Delta 4040 |
| – IMC3=18.58 km$^2$/s$^2$ : | 1640 kg |
| – Array Power Level | 14.3 kWe |
| – Transfer Time | 8.05 years |
| – Payload: | 1 kg to Earth |
| – Earth Departure: | May 20, 2008 |
| – Tempel1 Arrival: | Mar 23, 2011 |
| – Tempel1 Departure: | May 22, 2011 |
| – Earth Arrival: | June 7, 2016 |
| – Tempel1 Stay: | 60 days |
| – Vhp at Earth arrival: | 9.82 km/s |

**Issues:**

- Sample return Earth interface entry speed of 14.9 km/s

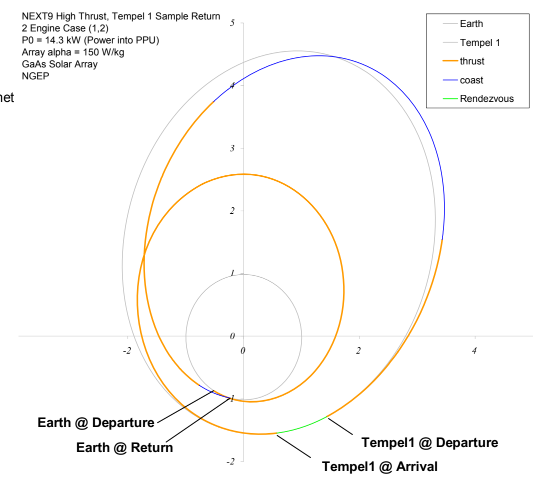



**Figure 22. SEPS Mission Analysis Results Summary**

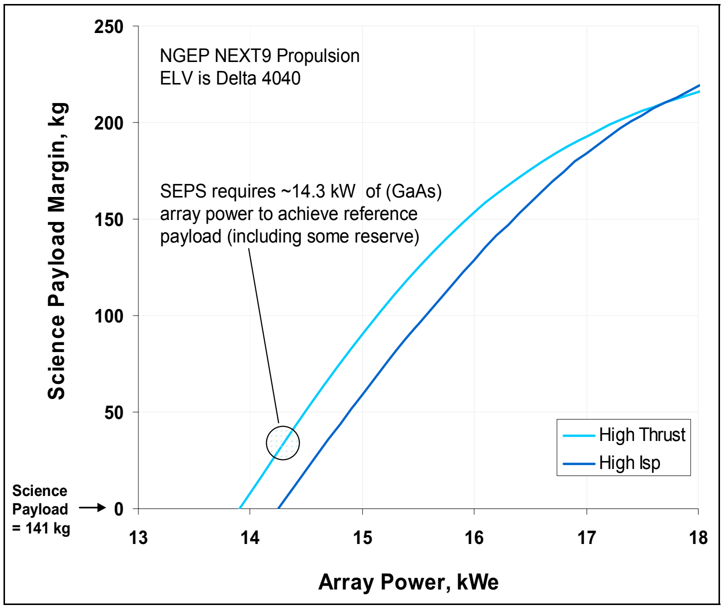



**Figure 23. SEPS Mission Analysis Power Scan Results**

*Radioisotope Electric Propulsion*

The REPS mission analyses results are summarized in Figure 24. This REPS transportation scenario is very similar to the SEPS case. The launch vehicle injects the spacecraft to an optimal $C_3$ = 55 km$^2$/sec$^2$ at an initial mass of 944 kg. The second generation REPS with its 141 kg of science payload then slowly spirals out to rendezvous with comet Tempel 1. This outbound spiral requires over 2 revolutions around the sun due to the low 750 W power of the REPS spacecraft coupled with the chosen Atlas 401 launch vehicle. After rendezvous, the spacecraft remains at Tempel 1 for 60 days, during which the sample retrieval system gathers the sample and deposits it in the return capsule. 50 kg of retrieval system remains at the comet, and the REPS vehicle begins Earth return maneuvers. Very similar to the SEPS trajectory, there is a long powered return with some weeks of coast before Earth encounter. The direct entry sample return capsule is released with a $V_{hp}$ of approximately 10.7 km/sec and the primary REPS vehicle performs a divert maneuver to miss the Earth, going into orbit about the sun. The capsule performs a direct entry at Earth interface, returning the comet sample to Earth. The total transfer time for the SEPS mission is approximately 12 years with departure in June 2010 and a return to Earth on June 2022. The primary issue of this scenario is the rather large Earth entry interface speed of over 15 km/sec.

For the Atlas 401 launch vehicle, Figure 26 shows payload as a function of Earth launch $C_3$. The baseline case that was analyzed in detail is shown in the diagram at $C_3$=55 km$^2$/sec$^2$. At this value of $C_3$, the payload is somewhat above the 141 kg reference payload, yielding some a reasonable payload margin for growth contingency. Note that for this case, the total BOL spacecraft power is 850 W, with 750 W into the PPU. The extra 100 W is for spacecraft housekeeping. For the baseline case described above, typical detailed mass statement is presented in Figure 25. It can be seen that the baseline REPS spacecraft has a dry mass of approximately 650 kg and a propellant load of approximately 250 kg, yielding a payload of 141 kg with a few percent payload margin.

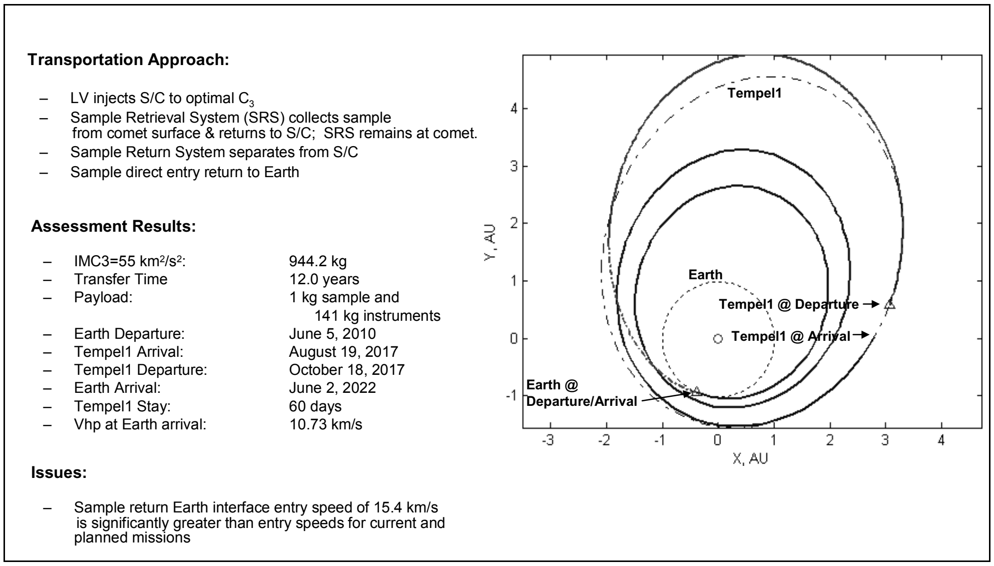



**Figure 24. REPS Mission Analysis Results Summary**

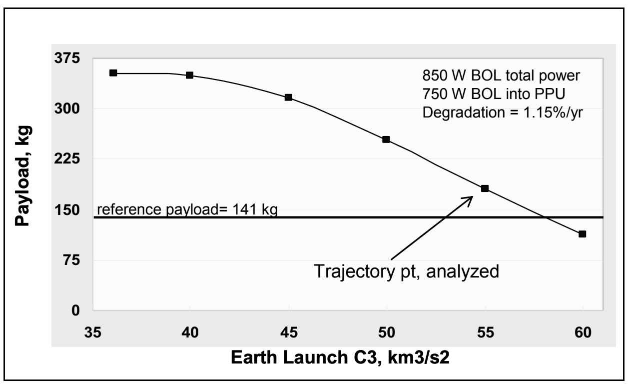



**Figure 26. Payload Delivery Results for 2nd Generation REPS Mission**

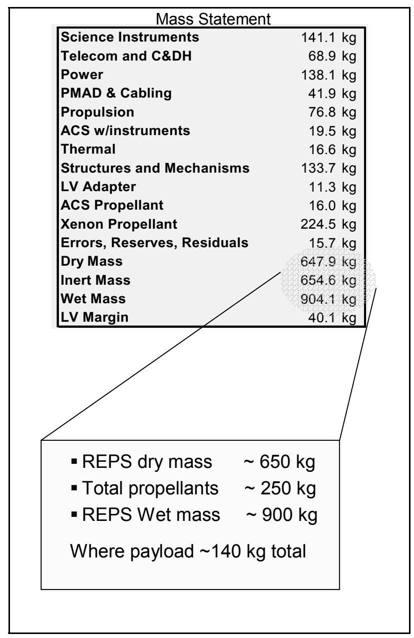



**Figure 25. Typical Mass Statement for 2nd Generation REPS Vehicle**

### 4. *CSSR Mission Cost*

Costs for the technology cases investigated in this study are provided in Figure 27. The cost for the chemical based mission was not given because the mission was deemed infeasible from a launch vehicle standpoint. The 14.3 kW SEPS based mission cost a total of just under ~617 million dollars ($M). The second generation REPS based mission with a BOL propulsion power of 750 W cost approximately 16% more than the SEPS mission or 720 $M. The larger REPS cost as compared to the SEPS cost can be attributed to several factors: 1) REPS requires a larger launch vehicle than SEPS; 2) the REPS mission used RPS power at a costs of 138 $M as compared to approximately

22 \$M for SEPS power (16% of the REPS power cost); and 3) larger REPS mission operations cost due to a 4 year longer total transfer time for REPS as compared to SEPS.

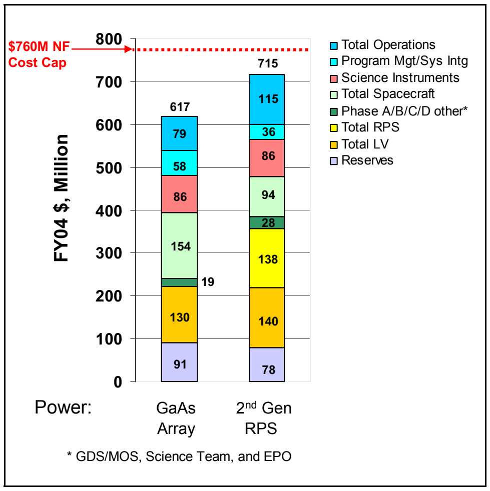



**Figure 27. CSSR Cost Comparison of REPS with SEPS**

### *5. CSSR Analysis Summary*

A summary of the results and a comparison of the technologies were made based on several figures of merit (FOM). The FOMs considered in this assessment include propulsion performance and total mission cost. The chemical propulsion based mission was found to require a three stage spacecraft with total mass of over 10 metric tons. No existing launch vehicle can place this mass to the required $C_3$. Thus, there was no viable mission scenario found, and the chemical based mission was deemed to be infeasible.

A comparison of SEPS and REPS was made and the summary is provided in Figure 28. For both the SEPS mission and the REPS mission, the total science payload delivered was 141 kg. The SEPS vehicle delivered the reference payload on an 8 year direct trajectory with a total stack mass of approximately ~1600 kg. There were three REPS cases investigated in this analysis. Those three cases correspond to three levels of RPS technology: first generation RPS is assumed to have an RPS alpha of 4 W/kg; second generation RPS is assumed to have an RPS alpha of 8 W/kg; and third generation RPS is assumed to have an RPS alpha of 10 W/kg. The first generation REPS based vehicle can perform the mission, and has a stack mass of approximately 1400 kg and a transfer time of approximately 12 years, compared to the SEPS transfer time of about 8 years. The second and third generation REPS based vehicles are significantly lighter than the SEPS vehicle and the first generation REPS vehicle, but the third generation does not reduce the mass by a relatively large amount with respect to the second generation vehicle. This rather minor mass reduction coupled with a relatively difficult and likely expensive effort to achieve third generation RPS technology, suggested that the analysis focus on the second generation REPS.

No SCPS mission cases were found viable during this study. The launch vehicle requirements to make a chemical mission viable were outside the scope of currently available expendable launch vehicle lift capability. REPS can perform the CSSR mission for 1st, 2nd, and 3rd generation RPS class REPS cases. It was found that 2nd and 3rd generation REPS technology resulted in stack masses that were lighter than the SEPS stack. SEPS based spacecraft can perform the mission with faster roundtrip transfer time and a smaller launch vehicle than REPS. Both SEPS and REPS could perform the mission within costs that were under the New Frontiers cost cap. This study identified SEPS as the less costly approach to perform the CSSR mission.

Further work could characterize the REPS performance over a wider range of comets. Additionally, investigation of a larger launch vehicle, such as the Atlas 501, should be made for REPS application to the CSSR mission. REPS total transfer times for this CSSR mission may be dramatically shortened with a larger launch vehicle.

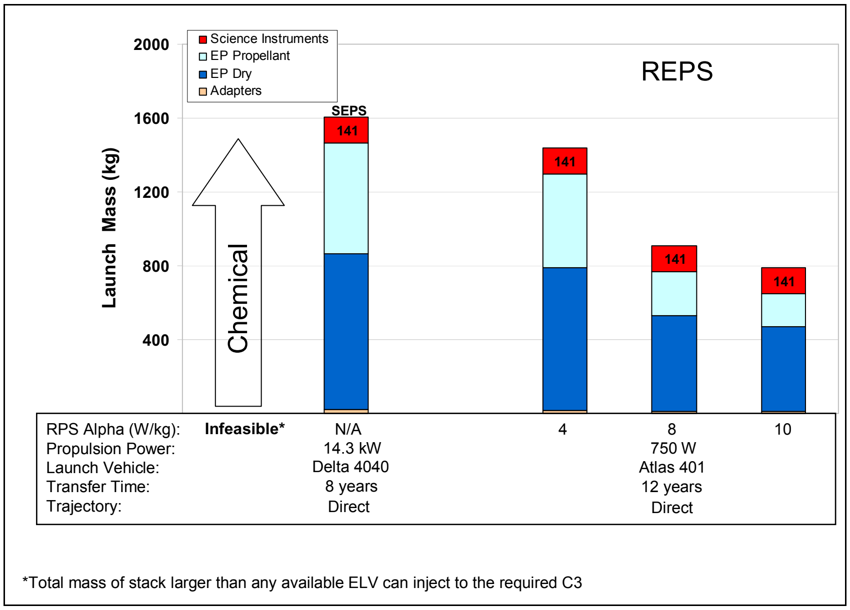



**Figure 28. Performance Comparison of REPS with SEPS and Chemical Propulsion**

## IV. Conclusion

REP offers a significant trip time reduction and increased target capture for the Trojan Asteroid Orbiter mission when utilizing a second generation radioisotope power system (RPS) and an advanced SOA Hall thruster. Marginal benefits were realized with the use of first generation RPS. REP was found not to be a viable option for the JPOP mission primarily due to the increased propellant mass required for capturing into a final parking orbit. For the CSSR mission, REP may be a viable option, although more analyses need to be performed.

Overall, REP appears to be best applied to small body missions beyond the main asteroid belt. Results of prior GRC analyses and this study consistently demonstrated that REP with 2$^{nd}$ generation RPS is beneficial for Trojan asteroid missions, and that large bodies are not likely favorable missions. Additionally, aggressive development of 2$^{nd}$ generation RPS could make REP a candidate for future inclusion in the ISPP technology investment portfolio.

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Reference herein to any specific commercial product, process, or service by trade name, trademark, manufacturer, or otherwise, does not constitute or imply its endorsement by the United States Government, or the Jet Propulsion Laboratory, California Institute of Technology.

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