ANALYSIS OF CHEMICAL, REP, AND SEP MISSIONS TO THE TROJAN ASTEROIDS

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Recent studies suggest significant benefits from using 1st and 2nd generation Radioisotope Power Systems (RPS) as a power source for electric propulsion (EP) missions to the outer planets. This study focuses on trajectories to the Trojan asteroids. A high level analysis is performed with chemical trajectories to determine potential candidates for REP trajectory optimization. Extensive analysis of direct trajectories using REP is performed on these candidates. Solar Electric Propulsion (SEP) trajectories are also considered for comparison against REP trajectories. A spacecraft mass is derived for the different types of missions, providing insight to how the REP (both 1st and 2nd generation) missions compare with chemical and SEP missions.

INTRODUCTION

The Dawn spacecraft demonstrates the benefits that EP can bring to cost capped scientific missions for solar system exploration. The electrical power for Dawn is provided by solar arrays whose effectiveness drops rapidly with distance from the Sun, limiting their utility for EP missions that go beyond the main asteroid belt. RPS generate electrical power from heat generated by the decay of radioactive materials and provide an alternate source of electrical power for deep space missions. These devices generate much less power than the Dawn solar array, but provide continuous power regardless of distance from the Sun. For the purposes of this study, 1st generation RPS assumes an efficiency of 4 watts per kilogram and 2nd generation RPS assumes 8 watts per kilogram. Several recent studies have examined the utility of RPS powered electric propulsion (REP) for outer planet missions and suggest that both 1st and 2nd generation RPS could provide an effective means of reaching and orbiting bodies beyond the main asteroid belt with reasonable trip times.[1,2] One potentially interesting destination for REP are the Trojan asteroids orbiting at the Jupiter-Sun L4 and L5 points. This paper examines potential benefits of REP by looking at chemical, SEP, and REP missions to these asteroids.

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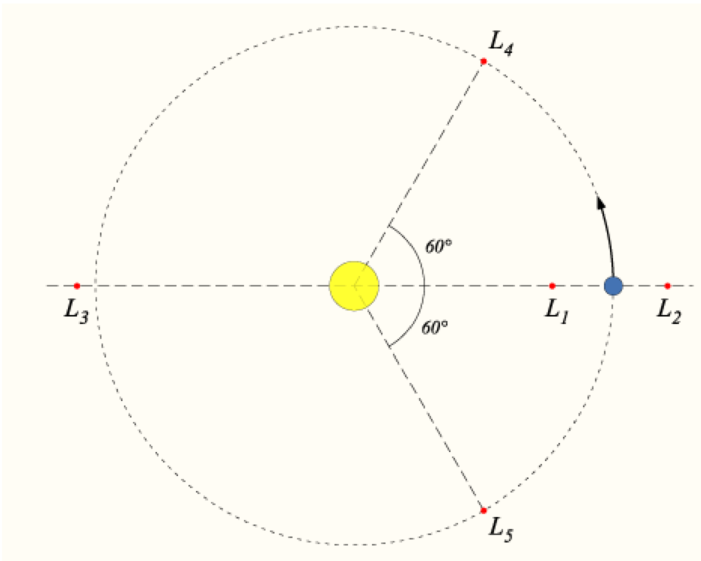
Objective

The purpose of this analysis was to supply optimal trajectories (chemical, low power nuclear electric propulsion, and solar electric propulsion) in support of a larger study aimed towards determining the feasibility of low power REP missions and then to compare them with similar SEP and chemical mission designs. The targets for this analysis are the Trojan asteroids of Jupiter and the trajectories considered were direct, Earth gravity-assist, and Jupiter gravity-assist trajectories. 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 1, are 60 degrees in front of and behind Jupiter in its orbit. They are stable points in the 3-body problem where asteroids have tended to gather in the Sun-Jovian system.



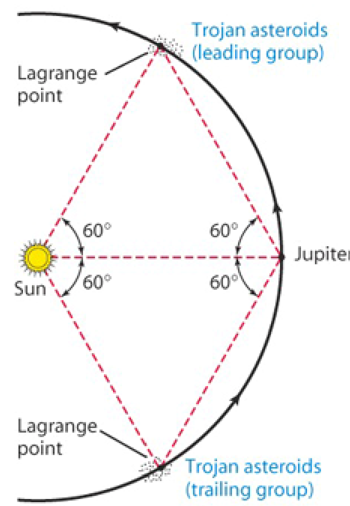


Figure 1 Illustration of the Lagrange Points and the Corresponding Locations of Jupiter's Trojan Asteroids

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 have been documented to populate Jupiter's stable Lagrange points. Figure 2 provides a picture of the main asteroid belt (green) and the obvious lumping of asteroids at Jupiter's L4 and L5 points (white).

CHEMICAL TRAJECTORIES

Although the asteroid rendezvous problem is better suited for EP technology, a chemical trajectory analysis can provide useful insight as well as a good baseline against which to compare an EP mission. A chemical mission to an asteroid usually requires either a complicated trajectory with multiple gravity assists and a long flight time or a very high ΔV trajectory. This is because the asteroid has almost no gravity and, therefore, the target body's gravitational field cannot be used to reduce the ΔV required for rendezvous (i.e. the ΔV for rendezvous will be exactly the same as the arrival V_∞ at the asteroid; a 5 km/s arrival V_∞ will require a 5 km/s rendezvous maneuver). The only way to reduce the ΔV for a chemical rendezvous with an asteroid is to use gravity assists to try to more closely match the asteroid's orbit before performing the rendezvous. These complicated multi-gravity-assist trajectories significantly limit the number of targets available and generally lack backup opportunities for because of the specific phasing that is required. In addition to these issues, a great deal of time is needed to find multi-gravity-assist solutions to just a few asteroids, let alone many of them.

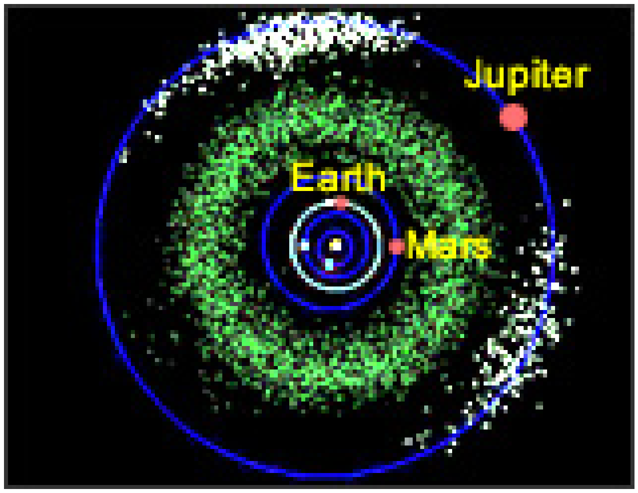


Figure 2 Illustration of Asteroid Belt and the Jupiter Trojan Asteroids

Considering the previous statements and since the main purpose of this study was to assess the benefit of 1st and 2nd generation RPS for an REP mission, the only pure chemical missions considered were direct and Jupiter gravity-assist (JGA) trajectories. The results from the chemical trajectory analysis were then used to single out the most promising targets for low thrust trajectory optimization.

Since the Trojan asteroids are essentially fixed relative to Jupiter, and direct trajectories to Jupiter are generally available in a predictable 13 month cycle, an optimization program can easily be run in batch mode to quickly calculate near optimal trajectories to a large set of Trojan asteroids. The optimization program used for this chemical analysis was MIDAS.[3] The asteroid set that was considered was not the entire list of 1700 asteroids, but instead just the numbered asteroids with an inclination of less than 20 degrees (about 700 asteroids). The data was created using a script that stepped through this list of asteroids and then modified a generic input file. Each of the input files was then run through the optimization code.

The results for the direct trajectory optimizations are provided in Figure 3 for flight times less than 9 years. Note that while there are 174 trajectories with flight times over 9 years, there were no optimal trajectories with flight times between 6.25 years and 9 years. Although this was not investigated too deeply, we suspect that these optimization runs simply did not fare well because of the use of a script to automate them through the optimization code (i.e. they needed more individual attention). The resulting trajectories generally had launch C_3 values between 55 and 90 km^2/s^2, flight times between 2 and 6 years, and on-board ΔV requirements between 4 and 10 km/s. Each trajectory assumes a launch on an Atlas 551 (although the launch vehicle should not have a major impact on the optimization) and the Isp for the dual mode bi-prop system was assumed to be 325 seconds. Although there is a large variation in time of flight for the trajectories, there does not seem to be a clear trend indicated in the plot. If the delivered mass is plotted versus the asteroid's inclination for each of the trajectories, as in Figure 4, a trend can be seen that seems to indicate the higher inclination targets have less delivered mass. This makes sense since targeting a larger inclination for the final rendezvous orbit would require an expensive plane change.

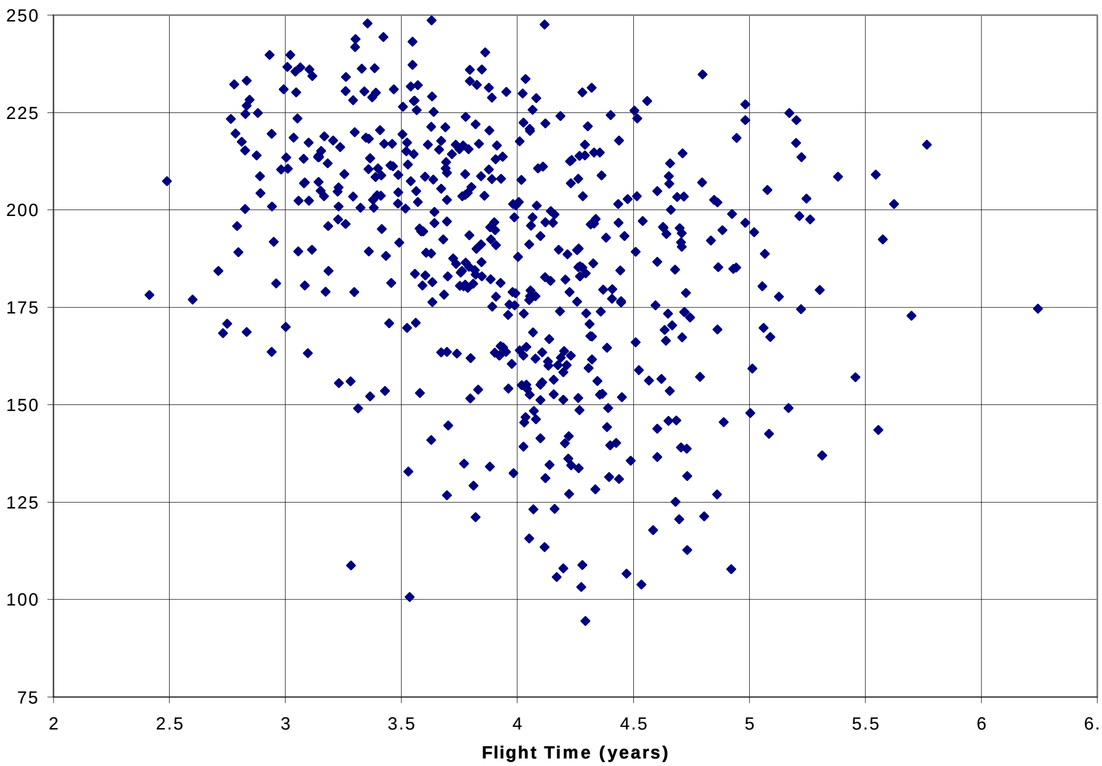


Figure 3 Delivered Mass for Direct Trajectories with Flight Time Less Than 9 Years on an Atlas 551 with a Bi-Prop Isp of 325 s Used for Insertion Maneuver

Although the main reason for calculation all of the direct trajectories was to determine the best asteroid candidates for the electric propulsion (EP) optimizations, it is interesting to compare the electric propulsion trajectories might to some of the better chemical trajectories. With 250 kg being the maximum delivered mass from the direct results, there is clearly not enough mass to create a direct mission to a Trojan asteroid. In the interest of finding a better chemical trajectory for eventual comparison with an EP trajectory, the same batch type run was done for JGA trajectories.

Jupiter is the natural choice for a gravity assist because any trajectory would be repeatable (available about every 13 months). Also, since Jupiter is in essentially the same orbit as the Trojan asteroids, the final heliocentric orbit before rendezvous would be similar to Trojan asteroid orbits and thus have a low arrival V_∞ (so the ΔV for rendezvous will be smaller compared to the direct). Of course, as with most gravity-assist trajectories, the penalty is an increased time of flight. The results from the JGA batch run, seen in Figure 5, show that although the delivered mass is increased to as much as 800 kg, the time of flight has dramatically increased to between 8 and 15 years. Plotted against propellant mass in Figure 6, we can deduce that most trajectories do not generate sufficient delivered mass for an actual mission. The green box in Figure 6 defines an area of feasible missions according to a spacecraft model that will be presented later in the 'mass and power derivations' section. While there are not many asteroids to choose from, the Jupiter gravity assist clearly makes a chemical mission feasible as compared to a direct method.

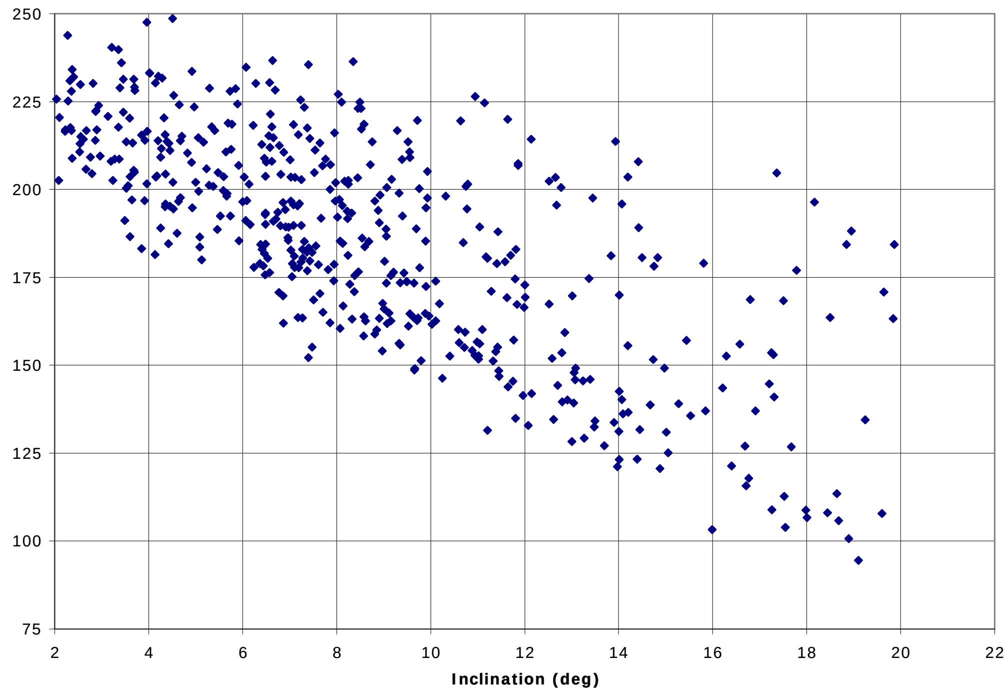


Figure 4 Delivered Mass for Direct Trajectories versus Asteroid Inclination (Atlas 551, Flight Time< 9 years, and Isp of 325 s)

While a lot can be learned from these trajectory runs, it is important to point out the shortcomings of this type of analysis. Since the optimized results came from a batch run rather than each asteroid receiving individual attention, it is possible that some of the results were locally optimal trajectories that could have been improved with extra individual effort. A perfect example of this is the 174 *direct* trajectories with flight times over 9 years. These trajectories did not seem to fare well from the generic input file approach judging from the fact that they were between 3 and 9 years longer in time of flight than the next best trajectory that had a 6.25 year time of flight even though they delivered about the same mass. So individually the chemical trajectories presented here do not necessarily say much about a trajectory to a specific asteroid. However taken as a whole, it is fair to observe trends in the data as well as to use them for comparison against the performance of some EP trajectories.

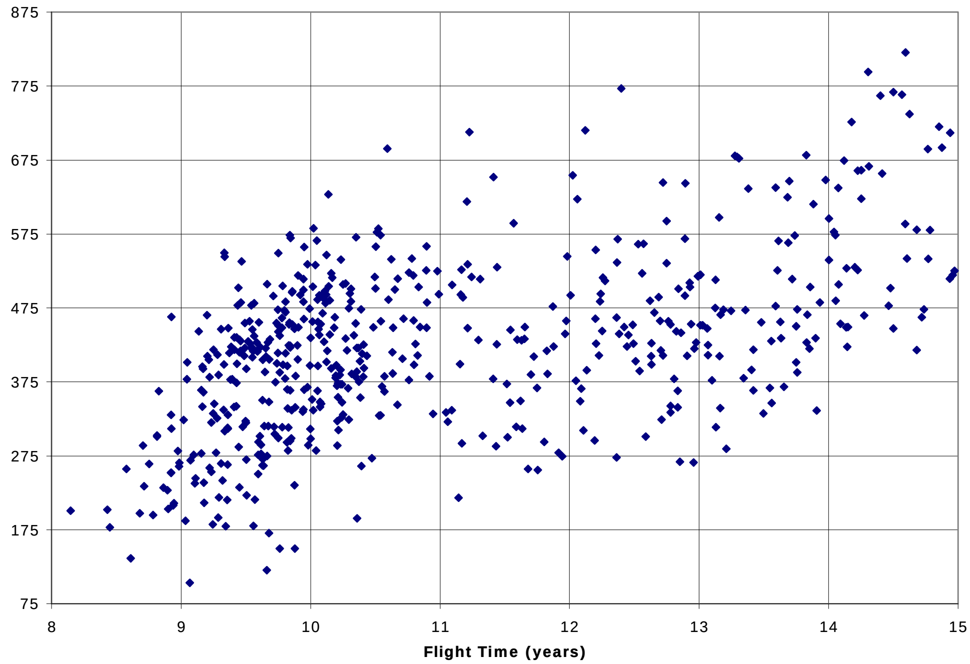


Figure 5 Delivered Mass for JGA Trajectories with Flight Time Less Than 15 Years on an Atlas 551 with a Bi-Prop Isp of 325 s Used for Insertion Maneuver

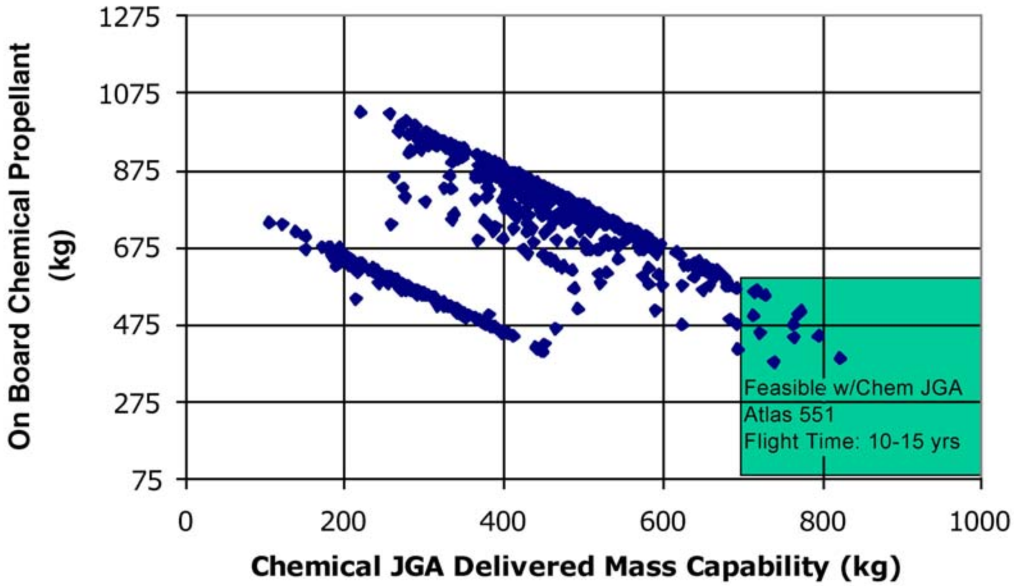


Figure 6 Required Propellant Mass for JGA Trajectories vs. Delivered Mass

REP TRAJECTORIES

REP refers to electric powered low thrust trajectories where the power comes from RTG type sources such as those found on the Cassini spacecraft. This study considered power levels of 750 watts to the electric propulsion system for most scenarios, although a few cases were run at 1000 watts. The electric propulsion trajectories take much longer to optimize and generally require individual attention. As a result, it is not practical to perform a batch run for each of the 700 candidate asteroids. The results from the direct trajectory analysis were used to determine the best candidates for an REP optimization since we decided not use any gravity assists for the REP analysis. This decision was based on initial results suggesting that direct trajectories would deliver enough mass as well as a lack of time available to analyze every scenario. Table 2 lists the asteroids selected for the REP analysis. The first two asteroids (Achilles and Hektor) were selected because they delivered near or more than the average delivered mass from the direct results despite the fact that they had high inclinations (i.e. if the trajectories delivered enough mass to higher inclined asteroids, that would indicate that many of the asteroids with lower inclinations are reasonable targets). The next two (1999 XW218 and 2000 AY217) were selected because of their high mass delivery and low inclination, and the last one (202 EK1) was selected because of its high mass delivery and its *very* low inclination of 0.11°. The trajectories were integrated using the VARITOP[4] optimization program, which is used quite often to optimize low thrust trajectories with constant power.

Table 1 Asteroids Used for REP Analysis Along with Corresponding Direct Trajectory Results

Asteroid Name	SMA (AU)	Inclination (deg)	Launch C_3 (km^2/s^2)	Flight Time (years)	Injected Mass (kg) (Atlas 551)	Delivered Mass (kg) (Atlas 551)
Achilles	5.2	10.3	80.8	4.1	1198	198.1
Hektor	5.2	18.2	87.5	10.4	981	147
1999 XW218	5.3	3.4	79.1	3.4	1258	229
2000 AY217	5.1	2.0	76.3	3.4	1355	244
2002 EK1	5.1	0.11	75.6	2.9	1379	240

Some of the results from the trajectory optimization can be seen in Figure 7 and Figure 8. The trajectories in these figures use the Atlas 551 launch vehicle. Often with electric propulsion trajectories, there are a number of different types of solutions. These types can usually be grouped into categories by the number of revolutions (revs) around the Sun. Typically better performance is achieved with more revolutions because the thrusting can be performed in a more optimal way. Figure 7 provides results for 2 of the asteroids searches for trajectories with 0-1 revs (single-rev) around the Sun. The trajectories to the higher inclined target of asteroid Achilles have poorer performance as would be expected.

As noted previously, trajectories with more revs tend to be more efficient. The results in Figure 8 indicate this since more mass is delivered despite the fact that a smaller launch vehicle (the Atlas 531) is used. In this case, as much as 1170 kg is delivered to asteroid 2000 AY217 with trajectories that have between 1-2 revs compared with 925 kg for 0-1 rev trajectories. As is typical, however, the increased performance comes at a cost of increased flight time.

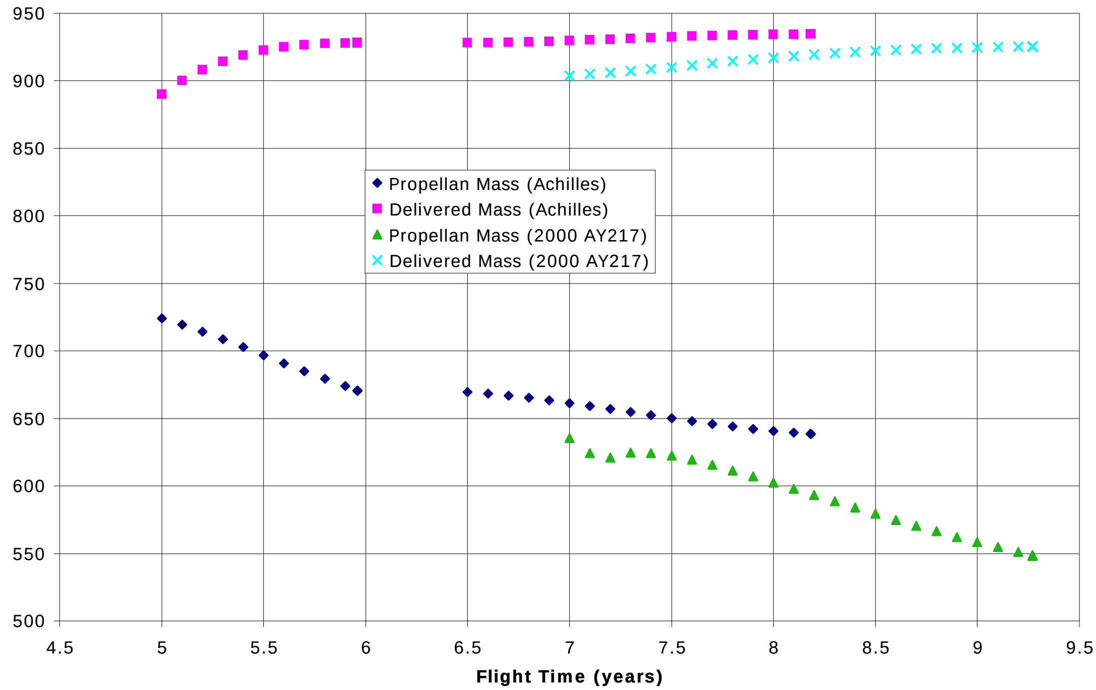


Figure 7 Performance Results from Trajectory Optimization for 0-1 Rev Trajectories with 750 Watts Input to PPUs Using an Atlas 551

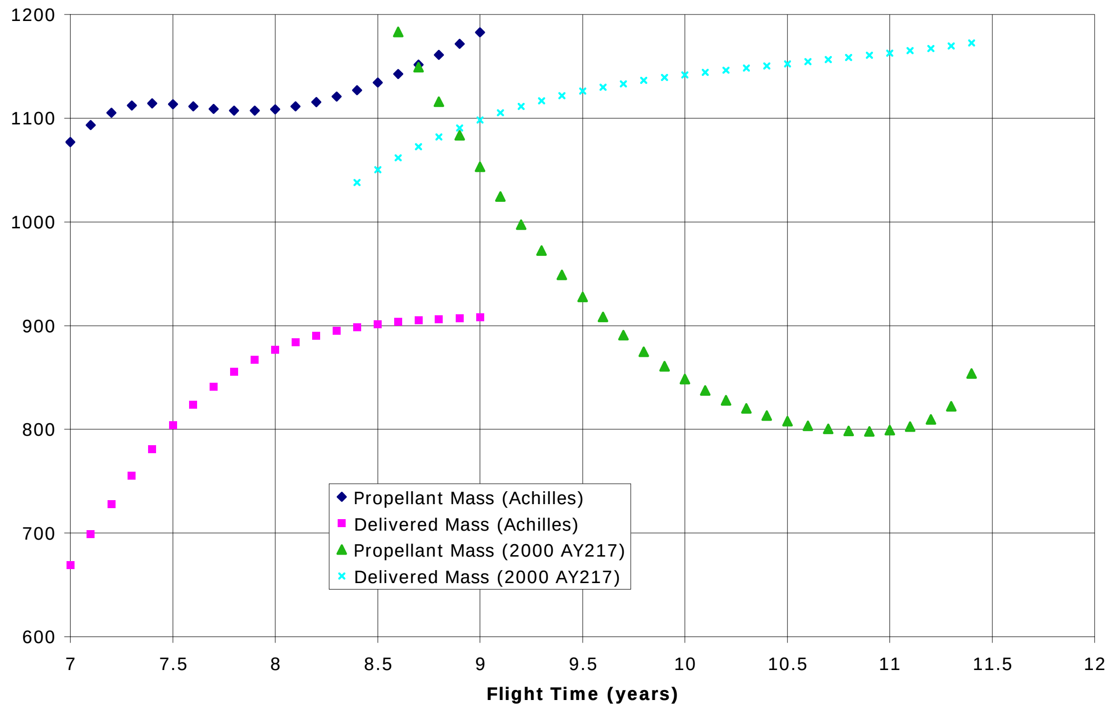


Figure 8 Performance Results from Trajectory Optimization for 1-2 Rev Trajectories with 750 Watts Input to PPUs Using an Atlas 551

A picture of the 1-2 rev trajectory to asteroid 2000 AY217 can be seen in Figure 9. The trajectory shown delivers about 1100 kg to the asteroid with a flight time of 9 years. The trajectory is presented with 30 day time ticks and the significant events (launch, stops and starts to thrusting, and rendezvous) are given on the right hand side of the plot as well as labeled on the trajectory. The trajectory is solid when the thrusters are active and dashed when the spacecraft is coasting. The orbits of Jupiter, the asteroid, and Earth are also provided in this plot.

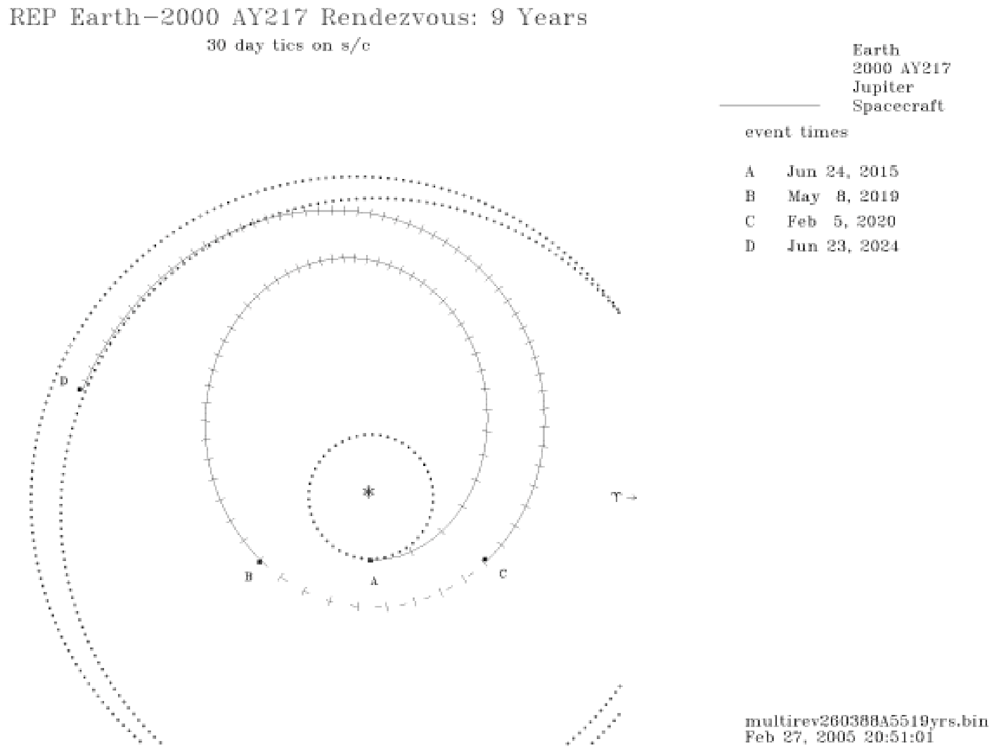


Figure 9 Trajectory to Asteroid 2000 AY217 with 1-2 Revs

Although many of the single-rev trajectories provide enough mass for a spacecraft using 2nd generation RPS technology (which requires about 8 watts per kilogram to just for the power source) they do not perform so well when 1st generation RPS technology is used (about 4 watts per kilogram). An initial reaction might be to go to higher rev trajectories, but as can be seen in Figure 10, the propellant mass is more than 1000 kg for many of the best performing trajectories and, of course, the flight time is much higher. Both can be significant issues for a spacecraft design. To address these issues, we looked at how much mass could be delivered with a single-rev trajectory that had 1000 watts of power and used an Atlas 551 launch vehicle. The single-rev trajectory would provide the benefits of both lower propellant mass and lower flight time and the increased power would provide a much needed boost in delivered dry mass. The results, seen in Figure 11, are somewhat promising as more than 1125 kg can be delivered to the asteroid Achilles for a 6 year flight time. As will be seen later, this was enough mass for a 1st generation REP which is significant since Achilles is one of the higher inclined asteroids.

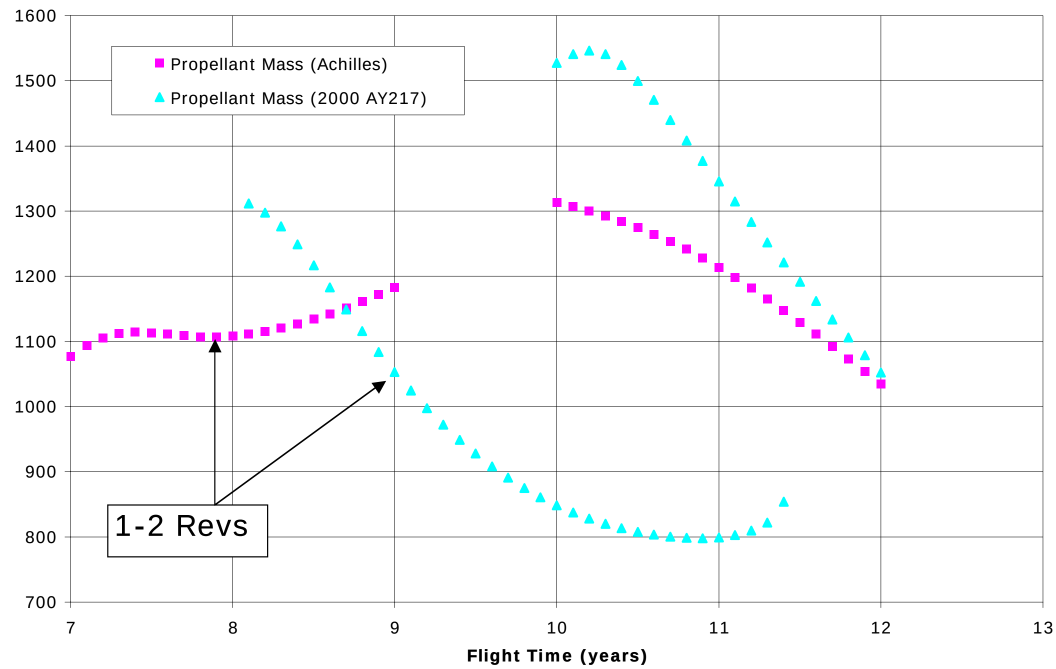


Figure 10 Propellant Mass Used for 1-2 Rev Trajectories (Using an Atlas 551) and 2-3 Rev Trajectories (using an Atlas 531) with 750 Watts Input to PPUs

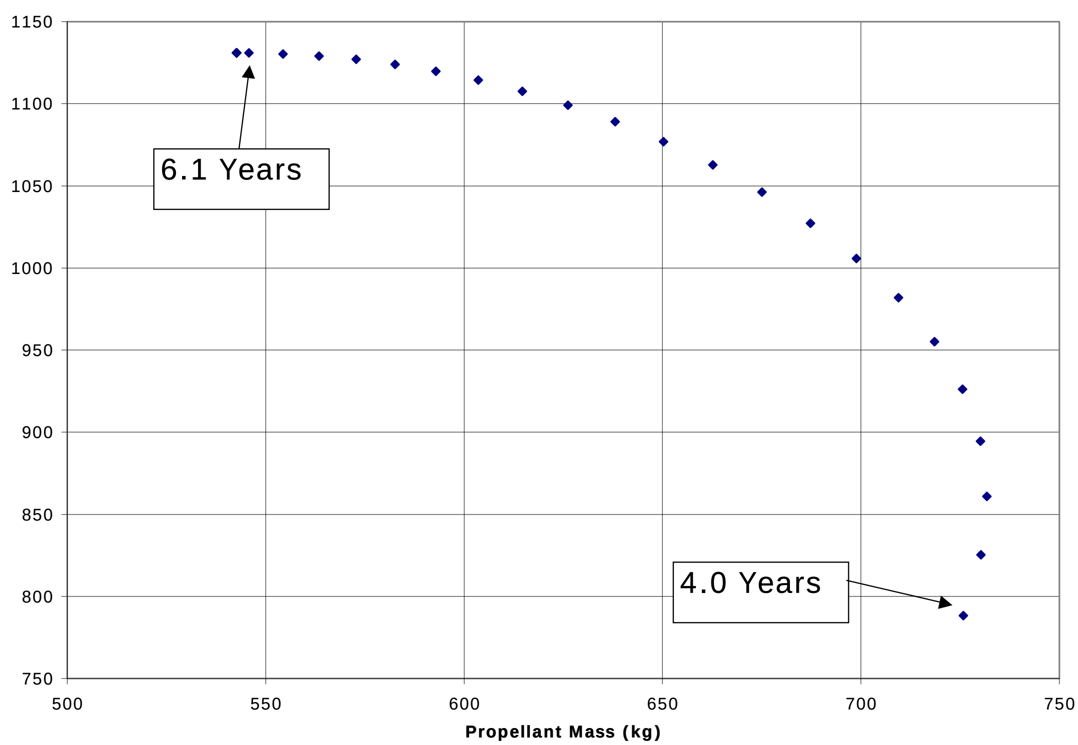


Figure 11 Delivered Mass versus Propellant Mass for Increasing Flight Times of Single-Rev Trajectories to Achilles with 1000 Watts Input to PPUs on an Atlas 551

A slightly different approach was taken to this problem in Figure 12 in order to try to take advantage of the larger mass delivery with multi-rev trajectories. Here, a 1000 watt multiple-rev (1-2 revs) trajectory to Achilles was kept at a constant 9 year time of flight, but the propellant mass was parametrically varied. As the propellant mass is forced to decrease, the delivered mass decreases as well, although it was possible to get significantly more mass than the 1125 kg maximum from Figure 11 for a reasonable increase in the propellant mass.

Other asteroids were investigated to see if they might be promising alternatives to the asteroids mentioned above. A few of the best single-rev trajectories are provided in Table 2. Although trajectories to these asteroids did not perform quite as well as asteroid 2000 AY217, many of them provide enough delivered mass to support a 2nd generation REP mission. Also, note in Table 2 that when the launch vehicle was increased to an Atlas 541, the trajectory would not converge without reducing the maximum possible injected mass by 110 kg (see column titled "Wasted LV Mass").

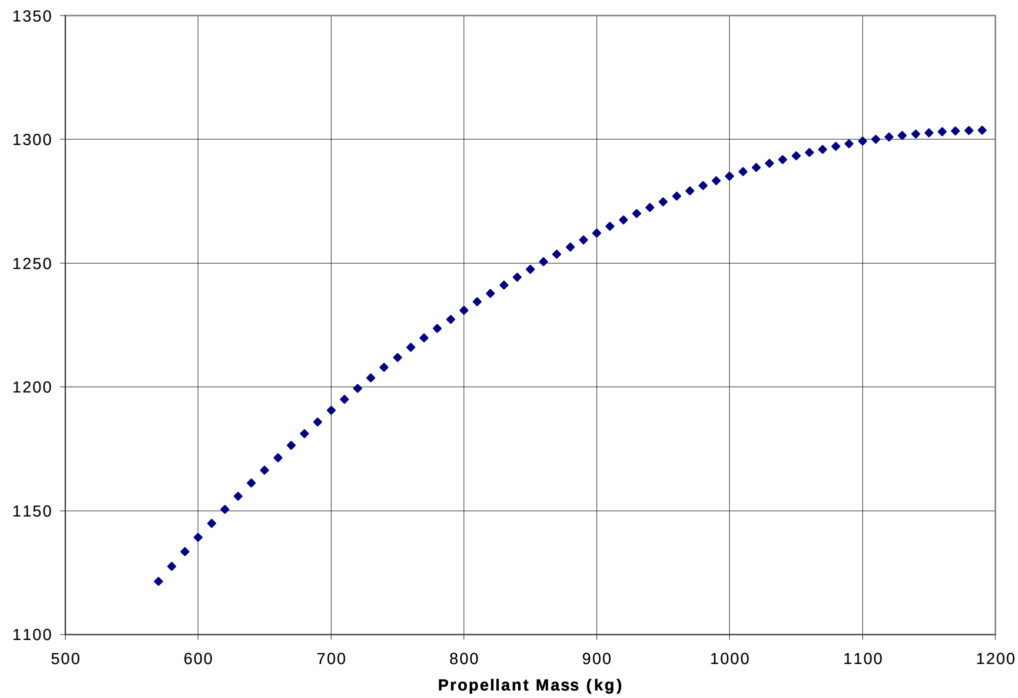


Figure 12 Delivered Mass for 9 Year Flight Time of 1-2 Rev Trajectories Using an Atlas 551 and 1000 Watts input to PPUs as Propellant Mass is Forced to Decrease

Table 2 Single-rev 750 Watt Results for Selected Other Asteroids

Asteroid Name	Launch Vehicle	Injected Mass (kg)	Wasted LV Mass (kg)	Delivered Mass (kg)	Propellant Mass (kg)	C_3 km^2/s^2	Flight Time (yrs)
2002 EK1	Atlas 531	1256	0	799	456	67.7	9
2002 EK1	Atlas 541	1319	110.0	838	481	69.5	10
1999 XW218	Atlas 531	1345	0	808	536	65.2	10

SEP TRAJECTORIES

The purpose of this analysis was determine what kind of SEP (Solar Electric Propulsion) system would be required to support a mission similar in scope to an REP mission. Two SEP options were considered. The first option used a lower power array (6kW with 1 engine or 15 kW with 2 engines) with a solar electric Earth gravity assist (SEEGA) to get to the asteroid and then used a chemical propulsion system to perform the rendezvous. The other option used a higher powered vehicle (25 kW ultraflex array) to transfer directly to the asteroid and then used the SEP system to perform the rendezvous as well. In all SEP cases, the NEXT engine was used in the optimization and the asteroid target was asteroid 2000 AY217. The analysis was performed using the SEPTOP[5] optimization software.

The results from the first option are presented in Table 3. These results are not very promising and the main reason for this is the large chemically performed maneuver required for rendezvous. The only way of dramatically increasing the delivered mass would be to include a Jupiter gravity assist as was done with the chemical trajectories. This would have the effect of decreasing the arrival V_∞ so less propellant is required. Using the chemical trajectories as a reference, the arrival V_∞ (or equivalently, rendezvous ΔV) could be reduced to perhaps as low as 2.5 km/s. This would increase the 15 kW case to a delivered mass of almost 2000 kg. Of course, the flight time would also increase to 10 to 15 years (or more), so this option was not pursued any further.

Table 3 SEP Results for SEEGA with Chemical Insertion Option (Isp=325 s Assumed for Chemical Maneuver)

Launch Vehicle	Power at 1 AU (kW)	C_3 (km^2/s^2)	Injected Mass	Arrival Mass (kg)	Arrival V_∞ (km/s)	Delivered Mass (kg)	Time of Flight (years)
Atlas 521	6	30.6	2496	2163	5.5	385	4.2
Atlas 531	6	36.9	2591	2261	5.5	402	4.2
Atlas 541	6	41.9	2666	2337	5.5	416	4.2
Delta 4040	6	9.6	2136	1790	5.5	318	4.2
Atlas 541	15	10.9	4791	4055	5.4	744	4.2

As mentioned above, the other SEP option considered was to use a large ultraflex solar array (25 kW at 1 AU) that would have enough power even at 5 AU to perform the rendezvous. Since the power is so large in this case, up to 3 NEXT engines were used in the optimization (there was very little benefit in going to 4 engines). The trajectory used was a direct trajectory because it was decided that an Earth gravity assist would impede the growth of the initial orbit and result in a longer flight time without a great mass benefit. This assumption was not confirmed with any specific results.

Since available power is proportional to $1/R^2$, 25 kW at 1 AU would be equivalent to about 1 kW at 5AU. This means that, on average, the power for the SEP rendezvous will be greater than for the REP trajectories and should therefore result in a higher delivered mass. This is in fact the case as can be seen in Figure 13 and Figure 14. Figure 13 gives results for a spacecraft with a 25kW array that is launched on a Delta 4040, a significantly smaller launch vehicle than any used in REP analysis. As might be expected, longer flight times yield better performance. Figure 14 provides a look at how sensitive the performance is to initial power. In this example, launched on an Atlas 401, 1 kW of additional power at 1 AU provides an extra 100 kg delivered to the asteroid (for an 8.5 year flight time). A plot of the 25 kW trajectory in Figure 15 shows that the spacecraft travels around the Sun for 1-2 revs before the rendezvous with the asteroid.

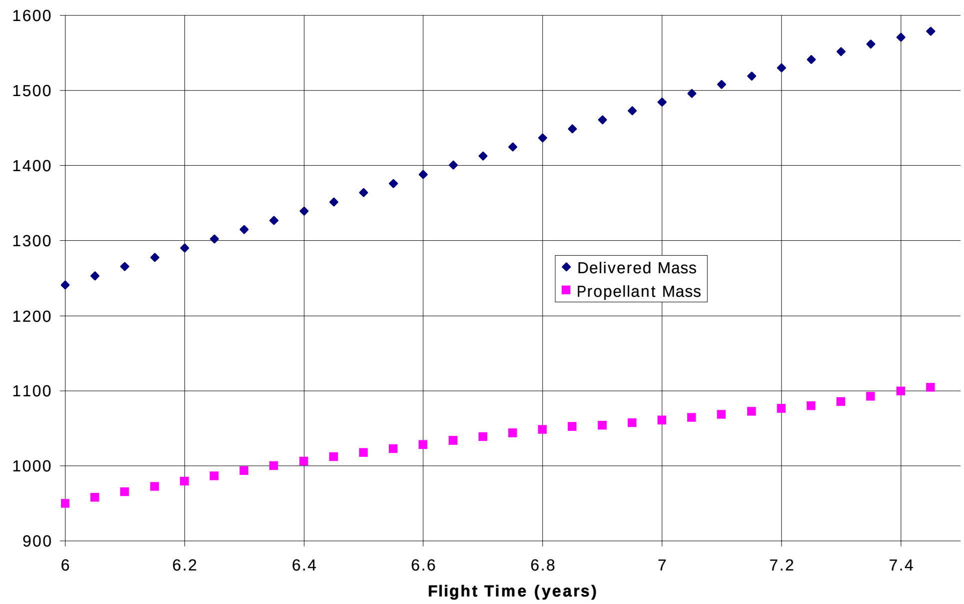


Figure 13 SEP Rendezvous Results as a Function of Flight Time for 25 kW Array Using a Delta 4040

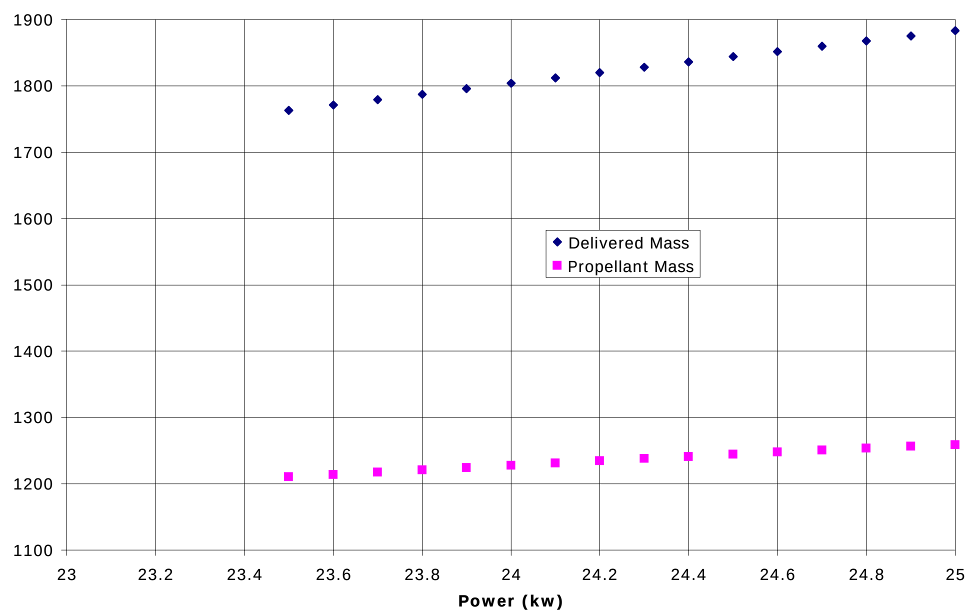


Figure 14 SEP Rendezvous Results for an Atlas 401 with a Fixed Flight Time Plotted as a Function of Power at 1 AU

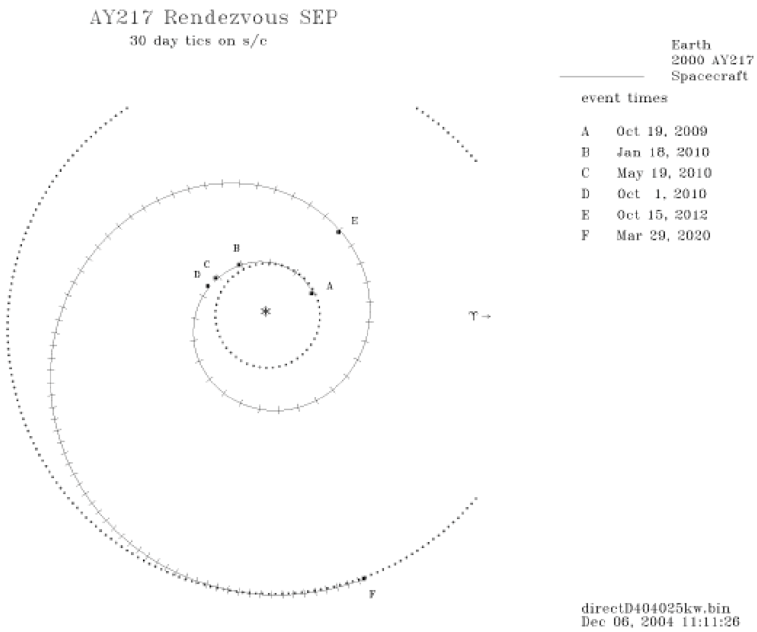


Figure 15 25 kW SEP Rendezvous with Asteroid 2000 AY217

MASS AND POWER DERIVATIONS

Spacecraft power and mass were derived for each type of mission (Chemical, REP, and SEP) to facilitate a more in depth comparison between the three methods. The instruments, attitude and control system (ACS), command and data handling (C&DH) and communication subsystem masses and power estimates were taken directly from Dawn's master equipment list (MEL) and were fixed at the values shown in Table 4.[6] Power is provided by the RPS technology in all but the SEP. The RPS is oversized to allow for 1.15% power degradation/yr over the life of the mission. The remaining subsystems are derived or scaled from subsystem parametric models developed by JPL's advanced projects design team ("Team X").[7] The Team X 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. In addition to the deterministic propellant allocation for orbit insertion and deep space maneuvers, the model adds 33 kg of propellant for targeting and orbital operations (taken from Dawn) as well as 2% over ΔV margin and 3% residuals (consistent with the Team X propulsion model).

Table 4 provides the derived spacecraft mass and power list for a chemical mission to one of the Trojan asteroids. A total power of 324 watts is required for the chemical spacecraft, and the delivered mass for this trajectory is 707 kg. The power is provided by 2nd generation RPS technology which is consistent with assumptions used in the REP spacecraft model. This provides for a fair "apples to apples" comparison of architectures. The trajectory used to calculate this spacecraft mass corresponds to one of the trajectories in the green box of Figure 6. It has an 11.2 year flight time to the asteroid numbered 24313. It launches with a C_3 of about 78 km^2/s^2 on an Atlas 551, has a deep space maneuver of about 1.02 km/s, and an insertion maneuver of about 0.84 km/s.

Table 4 Mass and Power Breakdown for Chemical Mission to a Trojan Asteroid

Mass Breakdown			Power Breakdown		
Subsystem	**Mass (kg)**	**Comments**	**Subsystem**	**Power (kW)**	**Comments**
Structures	127	Team X	Structures	0	
Thermal	23	Team X	Thermal	38	Team X
Chemical Propulsion	91	Team X	Chemical Propulsion	5	Assumption
ACS	40	Dawn	ACS	30	Dawn
Comm	30	Dawn	Comm	12	Dawn
Power	71	8 w/kg	Power	30	Team X
CDH	25	Dawn	CDH	70	Dawn
Harness	29	Team X	Harness	4	Estimate: 1.5%
Instruments	42	Dawn	Instruments	60	Dawn
LV Adapter	16	1.5% Wet Mass	LV Adapter	0	
CBE	**493**		**CBE**	261	
Contingency	148	30% CBE	Bus Power Contingency	78	30%
Chem Prop	630	Dawn/Team X			
Xenon Prop	0				
Total Wet Mass	**1271**		**Total Power**	**324**	
Delivered Mass	**707**				

Table 5 provides the derived spacecraft mass and power list for 2nd generation REP mission to asteroid 2000 AY217. The trajectory chosen for this table has a 7.0 year flight time and launches with a C_3 of about 71 km^2/s^2 on an Atlas 551. The Isp for the trajectory optimized to 1521 seconds, corresponding to an efficiency of about 0.43. The delivered mass for this case calculates to be 821 kg, more than enough considering that this trajectory can deliver up to 900 kg with the propellant load given in Table 1. In fact, a mission to this asteroid can also fit on an Atlas 541.[8] Although the delivered mass in this case is higher than the chemical propulsion spacecraft, the spacecraft will have a 1 kW capability when it arrives at the asteroid. This will certainly be an advantage for science.

Table 5 Mass and Power Breakdown for 750 Watt 2nd Generation REP Mission to a Trojan Asteroid

Mass Breakdown			Power Breakdown		
Subsystem	**Mass (kg)**	**Comments**	**Subsystem**	**Power (kW)**	**Comments**
Structures	117	Team X	Structures	0	
Thermal	25	Team X	Thermal	42	Team X
Electric Propulsion	83	4.5% Tank Fact.	Electric Propulsion	750	5% Conting.
ACS	40	Dawn	ACS	30	Dawn
RCS	14	Dawn	RCS	2	Dawn
Comm	30	Dawn	Comm	12	Dawn
EPS	136	8 w/kg	EPS	30	Team X
CDH	25	Dawn	CDH	70	Dawn
Harness	28	Team X	Harness	15	Guess: 1.5%
Instruments	42	Dawn	Instruments	30	Dawn
LV Adapter	19	1.5% Wet Mass	LV Adapter	0	
CBE	**568**		**CBE**	966	
Contingency	170	30% CBE	Bus Power Contingency	65	30%
Chem Prop	33	Dawn	EP Contingency	38	5%
Xenon Prop	684	8% Margin			
Total Wet Mass	**1456**		**Total Power**	**1068**	
Delivered Mass	**821**				

As mentioned earlier, the 1000 watt REP trajectory search to Achilles produced results that could accommodate even the 1st generation RPS technology. The mass for this spacecraft, given in Table 6, is 1087 kg upon arrival at the asteroid. The trajectory to asteroid Achilles chosen for this table has a 6.0 year flight time and launches with a C_3 of about 68 km^2/s^2 on an Atlas 551. The Isp for the trajectory optimized to 1783 seconds, corresponding to an efficiency of about 0.45. The delivered mass for this 1300 watt spacecraft is about 1087 kg.

Table 6 Mass and Power Breakdown for 1000 Watt 1st Generation REP Mission to a Trojan Asteroid

Mass Breakdown			Power Breakdown		
Subsystem	**Mass (kg)**	**Comments**	**Subsystem**	**Power (kW)**	**Comments**
Structures	117	Team X	Structures	0	
Thermal	25	Team X	Thermal	42	Team X
Electric Propulsion	79	4.5% Tank Fact.	Electric Propulsion	1000	5% Conting.
ACS	40	Dawn	ACS	30	Dawn
RCS	14	Dawn	RCS	2	Dawn
Comm	30	Dawn	Comm	12	Dawn
EPS	378	4 w/kg	EPS	30	Team X
CDH	25	Dawn	CDH	70	Dawn
Harness	28	Team X	Harness	15	Guess: 1.5%
Instruments	42	Dawn	Instruments	30	Dawn
LV Adapter	21	1.5% Wet Mass	LV Adapter	0	
CBE	**777**		**CBE**	1216	
Contingency	233	30% CBE	Bus Power Contingency	65	30%
Chem Prop	33	Dawn	EP Contingency	50	5%
Xenon Prop	598	8% Margin			
Total Wet Mass	**1641**		**Total Power**	**1331**	
Delivered Mass	**1087**				

Finally, the mass and power for the SEP spacecraft are provided in Table 7. The power mode used in this table is for full thrust cruise. The trajectory used for this mass derivation is a 7.4 year flight time to asteroid 2000 AY217. The trajectory has a C_3 of 0.5 km^2/s^2. Once this power mode was derived, we realized that the spacecraft would need 420 watts of power to operate compared to the 250 watts reserved in the trajectories presented above. The result of this was to force the trajectory optimization program to a 28.5 kW (at 1 AU) ultraflex array (although, with margin, the spacecraft array actually grows to 30 kW). In this design the spacecraft is accelerated by 3 NEXT engines with 2 engines being carried as spares. Despite the fact that trajectory is one of the best performing trajectories, there is little launch vehicle margin on the design (wet mass is 2696 and trajectory injects 2702). This is not as big an issue as it might seem since the trajectory uses a Delta 4040 launch vehicle, and there are plenty of larger launch vehicles that could be used to deliver some more mass. One issue that might actually force this trajectory onto a much larger launch vehicle (or to a better performing, higher rev trajectory) is the large throughput of Xenon. Currently this trajectory requires 1100 kg of Xenon to go through 3 NEXT engines. Any significant growth in spacecraft mass, could also force this trajectory to a higher power level. A major difference between this spacecraft model and the previous three is that there is no need for RPS technology on this spacecraft since the solar array should provide roughly a kilowatt of power once the spacecraft arrives at the asteroid. That said, building a deep-space spacecraft with a 30 kW ultraflex array could still be a technological challenge.

Table 7 Mass and Power Breakdown for 30 kW SEP Mission to a Trojan Asteroid

Mass Breakdown				**Power Breakdown**		
Subsystem	**Mass (kg)**	**Comments**		**Subsystem**	**Power (kW)**	**Comments**
Structures	222	Scaled from REP model		Structures	0	
Thermal	100	Team X		Thermal	150	Team X
Electric Propulsion	274	5 NEXT thrusters		Electric Propulsion	28100	5% Conting.
ACS	40	Dawn		ACS	30	Dawn
RCS	14	Dawn		RCS	2	Dawn
Comm	30	Dawn		Comm	12	Dawn
EPS	320	Ultraflex: 120 w/kg		EPS	15	Team X
CDH	25			CDH	70	Dawn
Harness	45	Team X		Harness	15	Estimate: 1.5%
Instruments	42	Dawn		Instruments	30	Dawn
LV Adapter	35	1.5% Wet Mass		LV Adapter	0	
CBE	**1146**			**CBE**	25074	
Contingency	344	30% CBE		Bus Power Contingency	97	30%
Chem Prop	33	Dawn		EP Contingency	1405	5%
Xenon Prop	1172	8% Margin				
Total Wet Mass	**2696**			**Total Power**	**29926**	
Delivered Mass	**1605**					

CONCLUSIONS

The major results from this study are summarized in Table 8. We were able to find a chemical mission that works for our spacecraft model, but there were only a few choices to choose from and over 700 asteroids were included in the original search. While it's possible that there were some trajectories that were could have made it into the green box of Figure 6 with some individual attention, it's still clear that there are not many options for chemical trajectories. The flight time is very long, and the spacecraft will only have 324 watts once it arrives at the asteroid.

Table 8 Summary of Spacecrafts Derived for Each Mission Type

Mission Type	Launch Vehicle	Max Del. Mass (kg)	Prop Mass* (kg)	S/C Mass (kg)	Dry Mass Margin (kg)	S/C Power (w)	Flight Time (yrs)	Isp (s)	RPS Tech
Chem.	Atlas 551	713	564	707	6	324	11.2	325	2nd
REP (750 w)	Atlas 551	904	636	821	83	1068	7.0	1521	2nd
REP (1 kW)	Atlas 551	1130	554	1087	47	1331	6.0	1783	1st
SEP (30 kW)	Delta 4040	1611	1091	1605	6	29926	7.4	N/A	N/A

***This is deterministic propellant mass required to deliver "Max Del. Mass" to the destination.**

The SEP trajectory does allow for a mission to an asteroid without using RPS technology, but the solar array on the spacecraft is huge. Considering that the trajectory in Table 8 only has 6 kg of margin, there is significant concern that a small increase in the power reserved for the spacecraft subsystems, could drive the required total array power up even more, or that the launch vehicle will need to get much larger. In the case of a larger launch vehicle, the propellant mass will also grow considerably from it's current value of almost 1100 kg which could have a considerable impact on the propellant tanks used for the spacecraft. Probably the biggest concern for the SEP mission, though, is building a 30 kW ultraflex array for a deep space mission.

Finally, the REP cases, while using on a very large Atlas 551, do have considerable dry mass margin. They also have the benefit of low flight times. If more mass was needed, a trajectory with a different number of revs or simply more flight time could be used. They also have the benefit of a very large power source for the spacecraft upon arrival at the asteroid. The extra mass could possibly be used for more instruments or at least instruments that utilized more power. Although the 750 watt trajectory is 2nd generation RPS technology, the 1000 watt case is not and has plenty of power at the asteroid with only a 6 year time of flight. The asteroid used for the 1000 watt rendezvous was Achilles which has a very high 10.3° inclination implying that there are likely to be a number of other asteroids that will also fit with the 1st generation technology.

From the work presented here, it appears that RPS technology can certainly be very useful in EP missions to the Trojan asteroids, and possibly mission enabling. The technology would almost certainly be necessary for missions to asteroids even further out than 5 AU (such as Centaurs) for both spacecraft power and propulsion (assuming that deep space reactors do not become available any time soon). It may also prove to be a useful technology for EP missions to the outer planets.

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