Solar Electric Propulsion Demonstration Mission Trajectory Trades

IEPC-2013-456

Presented at the 33rd International Electric Propulsion Conference,
The George Washington University • Washington, D.C. • USA
October 6 – 10, 2013

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Abstract: The SEPTD mission is a stepping stone leading to a reusable electric propulsion stage by demonstrating transfers from LEO to GEO and back to LEO. This set of high ΔV trajectories demonstrates long-term SEP operations and flies the SEPTD space vehicle through the radiation belts, sustained plasma environments, diverse distributed inertia Space Vehicle control environments and repeated Space Vehicle occultations. A large number of trades cases and point designs have been analyzed for requirements development, system sizing, and concept of operations definition. The trades and point designs have been completed using various methods and tools at ranging levels of fidelity. The focus was to find high ΔV solutions that fit within the budget (and hence mass) constraints of the SEPTD Mission requirements. Mass is purposefully constrained to constrain cost. The Baseline Mission begins in LEO, performs a low-thrust transit to GEO, transits back down to an equatorial LEO orbit, and then optionally spirals out from LEO to L1. There are a wide range of mission variants including EP system selection, and the Baseline Mission provides the performance capability for flexibility; including extended missions to NEOs, low lunar orbit, or the moons of Mars.

Nomenclature

ADCS	=	Attitude Determination and Control
ATP	=	Authorization to Proceed
BAA	=	Broad Area Announcement
BPT	=	Busek Primex Thruster
CONOPS	=	Concept of Operations
EM	=	Earth-Moon
EP	=	Electric propulsion
GEO	=	Geostationary Earth Orbit
GTO	=	Geostationary Transfer Orbit
HET	=	Hall Effect Thruster
IEPC	=	International Electric Propulsion Conference
Isp	=	Specific Impulse
km	=	Kilometers

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kW	=	Kilowatts
L1	=	Earth Moon Lagrange Point 1
L2	=	Earth Moon Lagrange Point 2
LEO	=	Low Earth Orbit
LLO	=	Low Lunar Orbit
m	=	Meters
MALTO	=	Mission Analysis Low-Thrust Optimization
MOC	=	Mission Operations center
N	=	Number
NASA	=	National Aeronautics and Space Administration
NEA	=	Near-Earth Asteroid
NEN	=	Near Earth Network
NEXT	=	NASA Evolutionary Xenon Thruster
PPU	=	Power Processing Unit
r	=	Radius
SECKSPOT	=	Solar Electric Control Knob Setting Program by Optimal Trajectories
SEP	=	Solar Electric Propulsion
SEPSPOT	=	NASA Glenn SEP Trajectory Code
SNAP	=	Spacecraft N-Body Analysis Program
TDM	=	Technology Demonstration Mission
TOC	=	Technology Operations Center
TRL	=	Technology Readiness Level
V_O	=	Initial velocity
V_F	=	Final velocity
XFC	=	Xenon Flow Controller
XIPS	=	Xenon Ion Propulsion System
ΔV	=	Delta Velocity Increment
Θ	=	Inclination change

I. Introduction

THE notion of using SEP in near-Earth space has been studied for decades. NASA's current architecture planning, technology roadmaps and goals of ongoing technology development imply near-term utility of high power SEP. Specifically, high power SEP systems are a cornerstone of several Exploration system architectures. Cargo transport to support human exploration beyond LEO is one key application. Potential missions to the moon, near-Earth objects, LaGrange points, and Mars all have variations that leverage high power SEP systems.

Other potential applications of SEP include: resupply, servicing, operational orbit change, debris removal, movement to a decommissioning orbit, and replenishment for assets in geo-centric space. These applications have been discussed at length in technical conferences and the space press.

Validation of high power SEP in operational systems must be accomplished before it can support human missions. Subsystem elements are sufficiently mature to push for this validation now. Demonstration of modular, high power SEP systems provides extensibility and scalability to larger missions. A near-term flight demonstration mission validates SEP for these applications and firmly establishes the 'capability.'

Definition of an appropriate SEP TDM requires an assessment of possible mission trajectories which meet the time and mass constraints specified in the NASA BAA[1] and hence can be achieved within the specified cost boundaries. A wide range of mission trades have been completed in support of the SEP TDM definition study. Variables included starting LEO orbit (altitude and inclination), number of trajectory legs, intermediate and final destinations, and EP system type (ion, Hall) and architecture (number of strings, operating power). A companion paper describes the Baseline Mission Concept.[2]

II. Mission Trade Studies

A large number of trades cases and point designs have been analyzed for requirements development, system sizing, and concept of operations definition, **Table 1**. The trades and point designs have been completed using various methods and tools at ranging levels of fidelity. The focus was to find high ΔV solutions that fit within the budget (and hence mass) constraints of the SEPTD Mission requirements. Mass is purposefully constrained to

Table 1. Mission Scenarios options examined during the study.

Start Orbit	Intermediate Steps and End Orbit	Other Variables
300 km, 28.5°	To GEO; then LEO 0°	2 BPT-4000 strings
400 km, 28.5°	To GEO; then LEO 0°	3 BPT-4000 strings
400 km, 28.5°	To GEO; then LEO 28.5°	4 BPT-4000 strings
400 km, 28.5°	To GEO; then L1	6 BPT-4000 strings
400 km, 28.5°	To GEO; then; LEO 0°; then; L1	3 XIPS-25 strings
400 km, 28.5°	To GEO; then L1; then polar-lunar orbit at 100 km	2 NEXT strings
400 km, 28.5°	To GEO; then L1; then to Mars orbit with Deimos Or to Mars orbit with Phobos	3 NEXT strings 1000 kg start mass
400 km, 28.5°	To GEO; then L1; then to Mars orbit with Phobos	1350 kg start mass
400 km, 28.5°	To GEO; then L1; then flyby Apophis	1638 kg start mass
400 km, 28.5°	To GEO; then L1; then rendezvous with Apophis	2000 kg start mass
400 km, 28.5°	To GEO; then L1; then to 1999 AO10	2100 kg start mass
400 km, 28.5°	To GEO; then L1; then to 2008 HU4	2250 kg start mass
400 km, 28.5°	To GEO; then L1; then to 2000 SG344	2500 kg start mass
400 km, 28.5°	To GEO; then L1; then to 1998 KY26	2750 kg start mass
400 km, 28.5°	To GEO; then L1; then to 2008 EV5	3000 kg start mass
500 km, 28.5°	To GEO; then LEO 0°	
600 km, 28.5°	To GEO; then LEO 28.5°; then to L1	
800 km, 28.5°	To GEO; then LEO 0°	
400 km, 5.2°	To GEO; then LEO 0°	
400 km, 37.8°	To GEO; then LEO 0°	
400 km, 40.0°	To GEO; then LEO 0°	
400 km, 40.0°	To GEO; then LEO 40°	
GTO (185 x 35786 km), 28.5°	To GEO; then LEO 0°	
400 km, Sun-Synch	To GEO; then LEO 0°	
400 km, Sun-Synch	Through super-synchronous at ~60,000 km to GEO; then LEO 0°	
600 km, Sun-Synch	To GEO; then LEO 28.5°; then to L1	

constrain cost. The Baseline Mission is a mission that begins in LEO, performs a low-thrust transit to GEO, transits back down to an equatorial LEO orbit, and then optionally spirals out from LEO to L1. There are a wide range of mission variants, and the Baseline Mission provides the performance capability for flexibility; including extended missions to NEOs, low lunar orbit, or the moons of Mars.

A. Tools and Assumptions

The mission design trades used a wide range of performance assessment methods and tools. Rapid low-fidelity equations were used for rapid assessments. The reference equation is simple as the basic Edelbaum approximation shown in equation 1.

$$\Delta V = \sqrt{{V_O}^2 + {V_F}^2 - 2V_O V_F \cos\left(\frac{\pi}{2} * \theta\right)} \tag{1}$$

Equation 1 provides an estimate of the minimum ΔV to perform a geocentric orbit transfer. The resulting ΔV can provide the mass fraction that is combined with flow rate calculations and duty cycle estimates for trip-time approximations.

For additional fidelity the mission trades were performed using multiple low-thrust analysis tools including SEPSPOT, the NASA Glenn Variant of the Solar Electric Control Knob Setting Program by Optimal Trajectories (SECKSPOT)[3], The Mission Analysis Low-Thrust Optimization (MALTO)[4] program, Spacecraft N-Body Analysis Program (SNAP)[5], and Copernicus[6,7]. The range of tools is appropriate for geocentric optimization, interplanetary transits, low-thrust spirals, and N-body analysis. In general, full N-body calculation were not performed for trajectory optimization, however; N-body dynamics were includes to determine suitable final trajectories at the Lagrange Points. To highlight the differences between tools, **Fig. 1** illustrates the variation in ΔV calculations from three different methods for a low-thrust spiral out from LEO. The results of **Fig. 1** should not be too surprising that SEPSOT and Edelbaum have similar performance estimates since they have the same fundamental analytic formulation. The SNAP results are the n-body propagation performance using thrusting in the velocity direction. There are two differences to note between the tools. The first difference is the trip time, equation 1 does not provide a trip time while SEPSPOT and SNAP do; both tools, for the cases run, assume an immediate full power start-up when in sunlight and no thruster while in shadowing. The 2[nd] different is that SNAP is not an optimization only a simulation. SEPSPOT will often do less than ΔV optimum transfers in order to minimize time. Therefore, the SEPSOT results for minimum time transfer are conservative with respect to mission performance capability. For example, during inclination change, the transfer from LEO-to-GEO will go super-synchronous, and during the transfer from GEO to LEO at 0^o inclination, SEPSPOT will increase inclination above zero and back down in order to get a reduced shadow period.

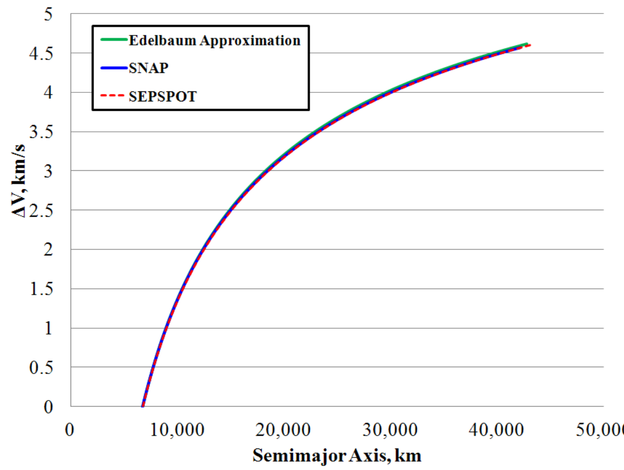

Figure 1. LEO to GEO performance estimate with various tools.

The analysis results assume that margin is applied at the Space Vehicle level for power, residual propellant mass, etc. Interplanetary analysis assumes a 90% duty cycle and a $1/r^2$ solar array model, which is relatively conservative. Low thrust transfers to the LaGrange points from GEO assumed the moon is at its minimum inclination. Also, the LaGrange trajectories do not use a full n-body optimization or evaluate the sensitivity to Halo or Lissajous orbit assumptions and insertion. It is assumed if the Space Vehicle can reach L1 distance and inclination without n-body dynamics, the use of gravity from the moon would not result in increased ΔV if coast periods are permitted. It is worth noting that many of the Halo orbit insertions occur at high inclination. Unless noted, LEO is defined as 400 km altitude.

The thrusters evaluated include the BPT-4000, NEXT, and XIPS. The performance of the thrusters is provided in **Table 2**. The power is the power into the thruster. For the interplanetary missions, the NEXT throttle table is based on the NEXT Projection 10 throttle table and the BPT-4000 is based on JPL testing of the thruster[8-11].

Table 2. Thruster performance for geocentric analysis.

	BPT-4000	NEXT	De-Rated NEXT	XIPS
Power, kW	4.5	6.891	4.5	4.5
ISP, s	2010	4188	3795	3600
Efficiency	56%	70.4%	65%	65%

B. Baseline Mission – Overview and Design

The SEPTD Baseline Mission is fully compliant with the key objectives by demonstrating a modular and extensible solar electric propulsion system. The mission design includes launch, commissioning, spiral cruise operations and decommissioning. It nominally launches in January 2018 and flies two LEO-GEO transits over the ~year-long cruise operations period, **Fig. 2**. The power and propulsion systems are at sufficient specific power to demonstrate the movement of large payloads from LEO to higher energy orbits at performance values consistent with future higher power electric propulsion capabilities (Isp, power-to-thrust, power-to-mass).

The SEPTD Space Vehicle is single flight element with no critical events occurring after launch and solar array deployments. It is based on integrated SEP and Bus Modules. The SEP Module includes three Hall thruster strings (3 + 0) which can be operated singly, in pairs or simultaneously with the advanced, high voltage, light-weight, blanket solar arrays providing adequate power for full power operations of all 3 HET string simultaneously. The thrusters operate only while the Space Vehicle is in the sun, **Fig. 3**. Eclipse durations vary from 33 to 43 minutes at orbital altitudes up to ~12,500 km. Eclipse durations fall off very rapidly and go to zero at ~13,000 km so operations at higher altitudes do not have eclipse-induced SEP operations shutdowns.

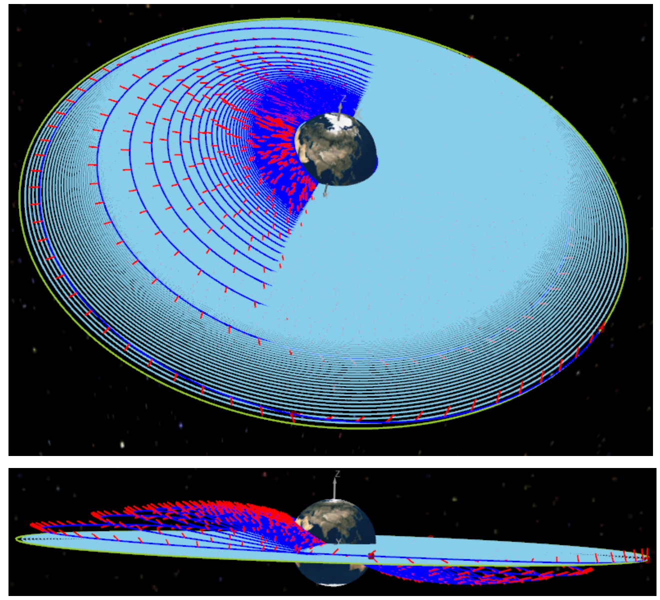

Figure 2. Evolution of SEPTD Mission spiral trajectory from LEO at 28.5° to GEO, then GEO to LEO at 0° – Top: Oblique view, Bottom: Side view; Illustrated with higher thrust-to-power than realistic to reduce the number of spirals and make them visible.

The SEPTD mission, and its SEP Module concept, is a stepping stone leading to a reusable electric propulsion stage by demonstrating transfers from LEO to GEO and back to LEO. This set of high ΔV trajectories demonstrates long-term SEP operations and flies the SEPTD space vehicle through the radiation belts, sustained plasma environments, diverse distributed inertia Space Vehicle control environments and repeated Space Vehicle occultations. At the same time, environmental risks are retired by demonstrating controllability of a SEP system in low-Earth orbit when the Space Vehicle is less tolerant to loss of attitude control over chemical based system; both because large SEP systems can have large gravity gradient effects and the use of chemical attitude control can diminish the effective specific impulse of the EP system. Substantial mission timeline, mass and propellant margins are built into the mission concept enabling flexibility to accommodate possible mission enhancements and uncertainties in mission characteristics.

While the mission trades illustrate the flexibility and capability of a high power electric propulsion system to perform significant ΔV, the Baseline Mission is to stay within geocentric and cis-lunar space. The Baseline Mission is from LEO-to-GEO-LEO at 0^{o} – L1 with a start mass of 2100 kg and relies on 3xBPT-4000 Thrusters. From the Baseline Mission, trades were completed using the NEXT thruster, different starting orbits, and different start masses. **Fig. 4** provides the semimajor axis and inclination as a function of time for the Baseline Mission using 3 x BPT-4000 thrusters.

The Baseline Mission has a final mass of 883 kg and a transit time of 420 days. Increasing the start mass to 2,250 kg increases the final mass to 942 kg and increases the transit time to 440 days. In addition to trading the start

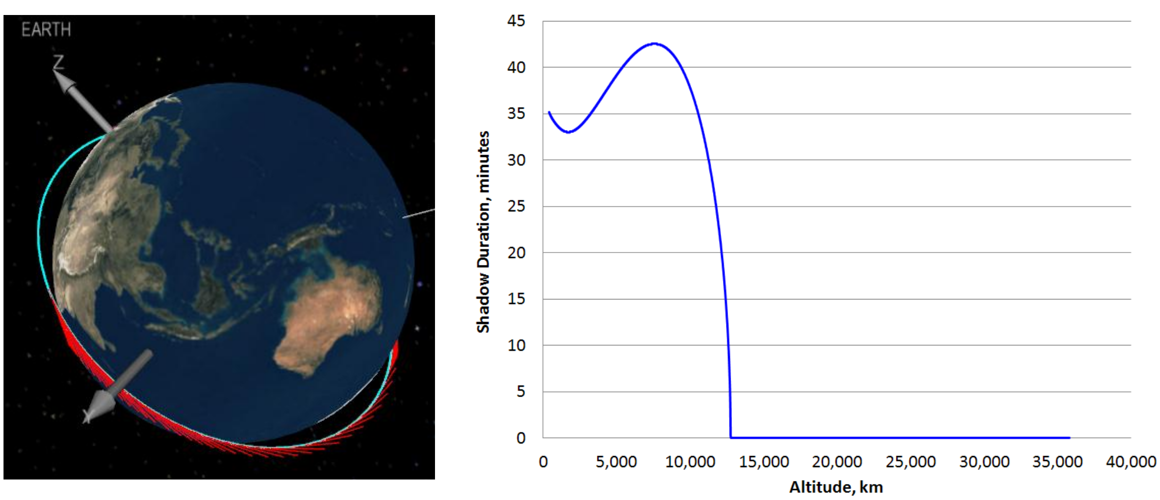

Figure 3. Electric Thruster Operations While Sunlit Only – L) Oblique view of SEPTD low orbit with thrusters operating in the sun and off in shadow; R) Example of eclipse cycle duration as a function of altitude for a low thrust equatorial transfer.

mass, the starting orbits were also traded. The results from various starting orbits are provided in **Fig. 5**. Starting at a higher orbit has only a minimal impact; saving approximately 2 days and 2 kg with an additional 100 km starting altitude. Starting in GTO saves 150 kg and 54 days. Starting at an inclination of 5.2° saves 4 days and 50 kg, while starting at 37.8° has a performance penalty of 46 kg and 26 days. XIPS and NEXT increase performance by 46% and 57% respectively.

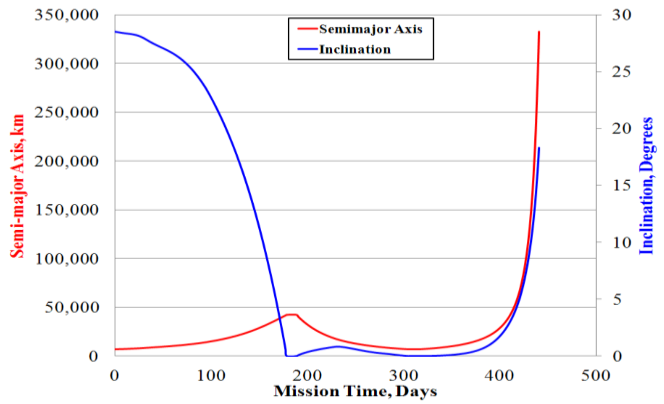

Figure 4. Baseline Mission from LEO at 28.5° to GEO to LEO at 0° to the EM-L1.

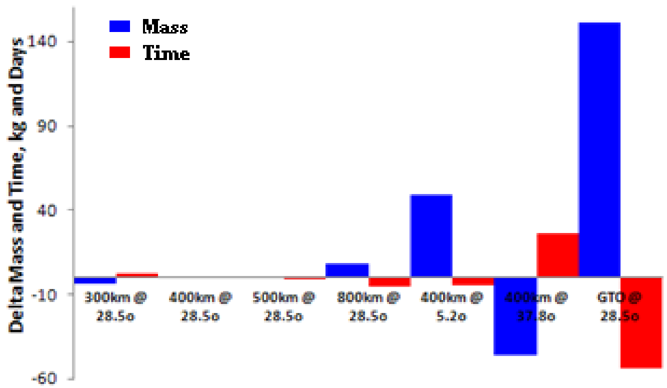

Figure 5. Trades from the baseline BPT-4000 option.

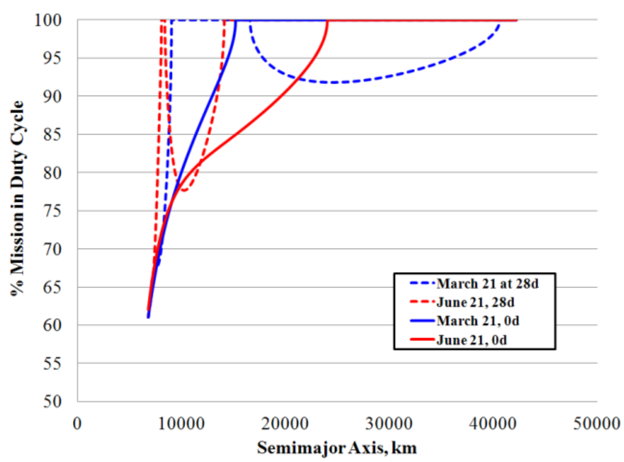

Figure 6. Shadow period vs. circular orbit altitude.

1. Mission Duty Cycle and Occultation

Another consideration for the mission is the expected duty cycle. The results for mission duration assume that the thruster is operating anytime the vehicle is in sunlight. Though beyond the scope of this portion of analysis, this assumes that there is sufficient tracking of the solar array or additional power margin to allow the thrusters to receive full power at all times. Busek has demonstrated pulsed thruster testing that validated the potential to start a Hall thruster almost instantly. The cathode can remain conditioned to mitigate potential restart delays.

When the mission starts in LEO, a large fraction, %, of the orbit is in shadow. As the Space Vehicle transits to a higher altitude, the mission duration in shadow will be reduced. The shadow duration during altitude, the effective duty cycle, is shown in **Fig. 6**. The results in **Fig. 6** show that the shadow periods are reduced when launching at equinox. Also, the results show that the equatorial orbits have no shadow period over the mission shown; at GEO there is a seasonal occultation for an hour.

As the Space Vehicle enters the shadow and stops thrusting, there is an on-off cycle of the system that can be derived to estimate a system cycles requirement. The number of cycles is dependent on the semimajor axis, eccentricity, and inclination of the orbit. The number of cycles is also dependent on the season. Launching near the equinox will have the fewest number of cycles, i.e. March 21. To bound the number of on/off cycles, **Fig. 7** illustrates the number cycles that occur for the baseline 3 BPT-4000 thruster with a start mass of 2100 kg during a transition from LEO-to-GEO or back down from GEO-to-LEO. A higher acceleration will also result in a lower number of cycles. Eclipse cycling does not stress the electric propulsion system since it is designed to experience >10,000 cycles

C. Mission Options

Over the course of the study, several missions have been evaluated and multiple parametric trades have been completed. Most of the trades involve variants of geocentric transfers from LEO to GEO to L1. The trades included looking at thruster options of NEXT, the BPT-4000, and a cursory look at XIPS. **Fig. 8** provides the semimajor axis versus time for the BPT-4000 and NEXT starting from LEO to GEO and then either continuing on to L2, spiraling down to LEO at 0°, and spiraling down to LEO at 28.5°. The results in figure 2 are based on a start mass of 1638 kg using either 2 x BPT-4000 thrusters at 4.5 kW each or 2x NEXT thrusters de-rated to operate at 4.5 kW each. These define the Reduced SEPTD option trajectory and extended mission options.

The first leg of the mission is the same for all three variants, LEO-to-GEO. The delivered mass to GEO is 1402 kg and 1217 kg for the de-rated NEXT and BPT-4000 respectively. The NEXT thruster completes the mission GEO in 364.5 days while the BPT-4000 option requires 216 days. The NEXT thruster also requires 1624 occultation cycles while the BPT-4000 sees 1170; a 40% impact. The LEO to GEO transfer ΔV is approximately 5.8 km/s. The results from the three options from GEO are shown in **Table 3**. The results indicate, as expected, that the ΔV is reduced by a little more than 1 km/s for the equatorial transfer to LEO over performing the plane change back to 28.5°. The BPT-4000 requires more mass than NEXT for the transits, but completes the missions with a shorter mission duration. Also, the number of eclipse cycles is higher with NEXT than the BPT-4000, and there were no additional shadow periods from GEO to the EM-L1. The reduced SEPTD Space Vehicle carries 658.4 kg of xenon. Using NEXT enables all three extended mission options while fuel restrictions limit extended mission options for the HET system to L1 only.

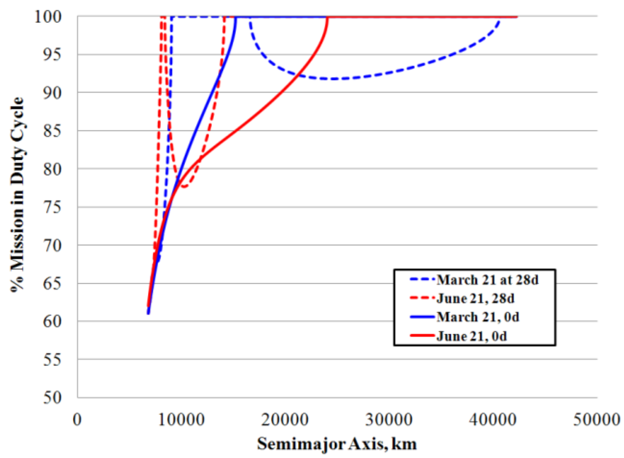

Figure 7. Number of occultations experience during a transit from LEO-to-GEO.

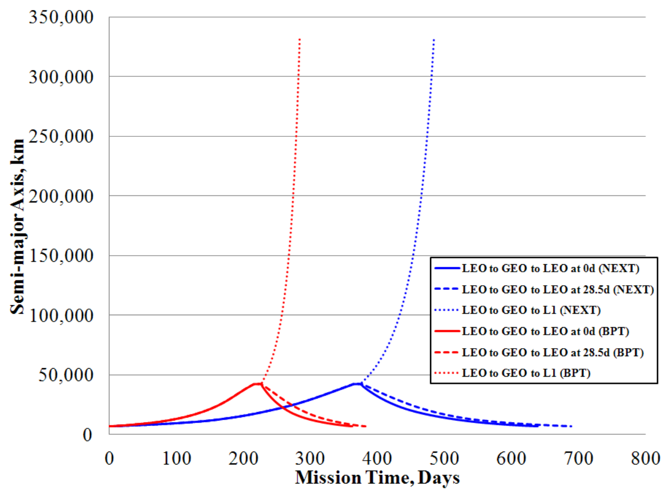

Figure 8. Duration of Reduced Mission (LEO at 28.5° →GEO – Then to LEO at 0° or LEO at 28.5° or L1) Starting At 400 km. The NEXT system requires about double the time of the HET system for each scenario as expected.

2. Sun-Synchronous Starting Orbit

A significant deviation assessed is an option to start in a sun-synchronous orbit. Starting in a Sun-sync orbit allows the Space Vehicle to begin the initial checkout and spiral out in constant or near-constant sunlight. However, the Space Vehicle must start in a very high inclination if the desire is to transfer to an equatorial geocentric orbit. The inclination required versus starting circular altitude is shown if **Fig. 9**. Starting with an altitude of 400 km requires a 97° inclination. The plane change is too large to be practical, although, with the NEXT thruster, it is feasible.

The performance from a starting sun-synchronous orbit to GEO and then down to LEO at 0° with a start mass of 2100 kg is 1040 kg and 1500 kg in 343 days and 656 days for the BPT-4000 and NEXT respectively. The transfer from a 400 km sun-synchronous to GEO is a ΔV of 9.2 km/s for the minimum time transit. The SEPSOT minimum time transit has a transit of interest. Because of the high starting inclination, the minimum time transit increases

Table 3. Performance to the three options from GEO.

	LEO (400km @ 0°)		LEO (400km @ 28°)		EM-L1	
	2 x NEXT	2 x BPT-4000	2 x NEXT	2 x BPT-4000	2 x NEXT	2 x BPT-4000
Start Mass - End mass of Leg 1	1401.9	1216.7	1401.9	1216.7	1401.9	1216.7
ΔV from GEO	4.64	4.63	5.76	5.78	2.15	2.15
Mass at LEO, kg	1237.5	961.9	1200.7	907.4	1323.2	1091.1
Transfer Time, Days	264.4	136.1	314.0	155.5	110.2	57.8
Total Mission Time, Days	628.9	352.1	678.6	371.5	474.8	273.8
Eclipse Cycles	1614	947	1622	669	0	0

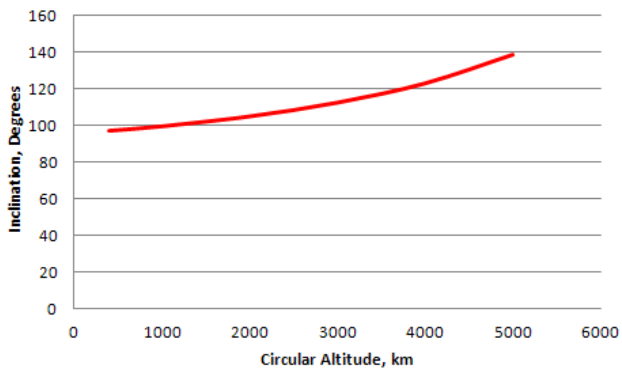

Figure 9. Inclination required for sun-synchronous orbit.

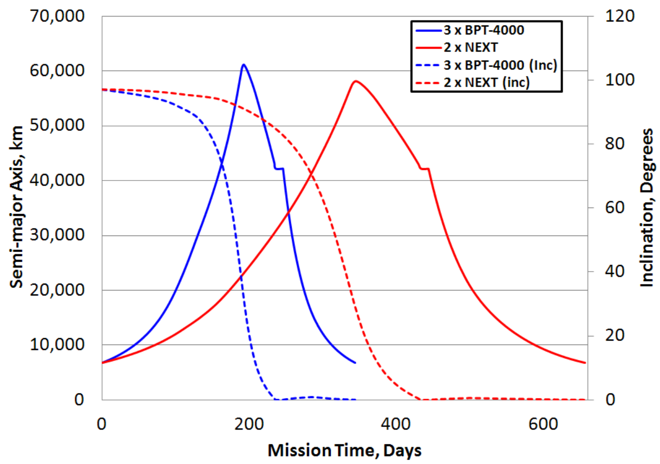

Figure 10. Sun-Synchronous Mission Performance – Mission Duration Starting 400 km Sun-Synch Orbit Out To GEO Via a Super-Synchronous Altitude and Then Down to LEO at 0°. The NEXT system requires about double the time of the HET system as expected.

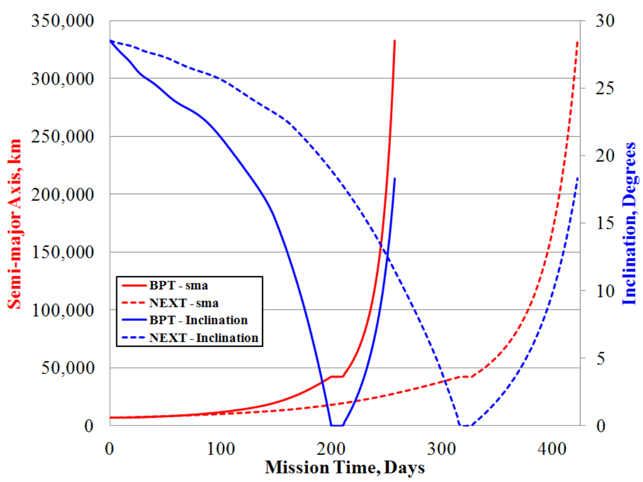

Figure 11. Transit from LEO-to-GEO-to-L1.

eccentricity and raises the orbit to allow for a larger fraction of each orbit to remain in sunlight and provide the fastest transfer possible. **Fig. 10** shows the inclination change and semimajor axis versus time for the transits. A ΔV savings of almost 1 km/s can be saved with a slower more ΔV efficient transfer. Even with the higher performance results, the transit from a sun-sync orbit to GEO is relatively impractical for an operational system though, as described, offers many benefits as a demonstration mission.

3. Missions Beyond Escape

The mission options evaluated through the study also included potential missions to go beyond L1. The trades assumed the Space Vehicle maintained a total propellant limit of 900 kg and the mission timeline must be constrained for total mission duration of two years as a demonstration mission. The escape missions include a transfer to low lunar orbit, Near Earth Objects, and the moons of Mars. For the Hall thruster option, the missions are propellant constrained and for NEXT, the missions are time constrained due to the ΔV and time requirements of the geocentric phase of the mission. The orbit transfer from LEO-to-GEO-to-L1 prior to escape is shown in **Fig. 11**. Approximately half of the mission time is the transit from LEO to GEO and the second half of the mission timeline is for the transit from GEO to L1. The mission requires approximately 8 km/s of ΔV from LEO to L1 with a time optimal solution. With a start mass of 2100 kg, the final mass for the BPT-4000 and NEXT is 1387.2 and 1732.1 g with transit times of 248 and 413 days respectively. **Fig. 11** illustrates a 10-day longer mission profile because of an arbitrary 10-day stay time at L1 to demarcate the mission phases, shown as a plateau in the plot. To meet the propellant and timeline constraints, the BPT-4000 only has 187 kg of propellant remaining and 483days of transit time beyond L1. Given the same constraints, the NEXT thruster has 532 kg of propellant available, but only 318 days for the post L1 transit.

Low Lunar Orbit. Missions to low lunar orbit are relatively easy from L1. Unstable orbits around L1 can enter lunar orbit due to perturbations alone. The time constraints require a less than optimal solution. Again, there were no requirements for the L1 orbit; so it was assumed that the transfer from L1 would start with no relative velocity with respect to the Lagrange point. While this is not a valid assumption, this is more conservative than an optimal transfer than passes through L1 with a non-zero relative velocity. For example, the Smart-1 mission passed through EM-L1 from high earth orbit to the moon without stopping at L1. The mission shown is more consistent with a mission from L1 that includes a short stay in an orbit around L1 prior to departing towards the moon and provides an opportunity to demonstrate Halo orbit station keeping.

For the Low lunar orbit mission design, Copernicus was used to for the transit from L1 to a polar lunar orbit with an altitude of 100 km. Copernicus was used for a feasible solution without a coast period using a suboptimal control law. An optimal solution would include coasting from L1 to lunar orbit. The results are shown in **Table 4** and **Fig. 12** illustrates the mission from EM-L1 to a polar LLO.

Table 4. Thruster performance for geocentric analysis.

	3 x BPT-4000	2 x NEXT
Start Mass - End mass of Leg 1	1387.2	1732.1
ΔV from L1 to 100 km LLO, km/s	1.87	1.46
Mass at LLO, kg	1261.7	1608.5
Transfer Time, Days	40	58
Propellant Required, kg	126	124
Eclipse Cycles	Lunar Eclipse	Lunar Eclipse
Total Time including Time to L1, Days	523	376

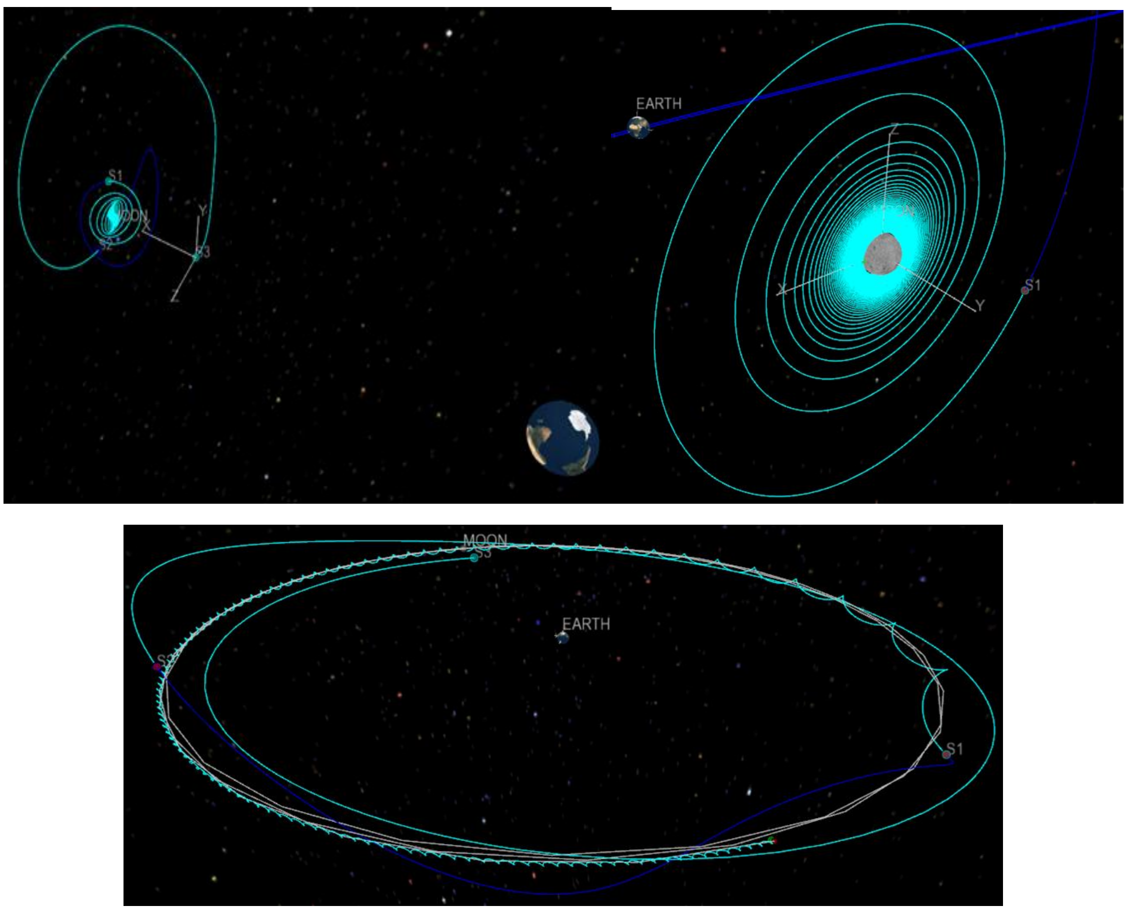

Figure 12. EM-L1 transit to LLO in a two-body rotating frame (top left), in a moon centered frame (top right), and in an Earth centered frame (bottom).

Just as with the geocentric missions, the BPT-4000 thruster can perform the lunar orbit insertion and spiral to 100 km altitude with a reduced mission time, but requires more propellant.

Near Earth Asteroids. Because of the high interest in Near-Earth Asteroids (NEAs), for planetary protection and for manned mission targets, flexibility for a baseline or even extended mission to a NEA with the demonstrator Space Vehicle may be desired. One of the challenges of performing a rendezvous with the SEP vehicle is the limited duration available for the interplanetary transit.

Apophis. A mission to Apophis is evaluated for its interest as a potentially hazardous object. A range of options have been evaluated for both the BPT and NEXT cases. The BPT-4000 options to Apophis are illustrated in **Fig. 13**. The BPT-4000 does not have sufficient propellant to allow an Apophis rendezvous. Increasing the trip time and allowing an L1 departure time at an ideal launch opportunity still does not provide sufficient performance. However, an Apophis flyby only requires 115 kg and is a feasible mission; even with the two year total mission constraint.

The Apophis mission was also evaluated using the NEXT thruster. The NEXT option has significant propellant capability, but very little time to complete the transit. Like the BPT-4000, the NEXT thruster can meet all mission constraints for the Apophis flyby. However, unlike the BPT-4000, if the L1 departure time can be phased and the time of flight can increased by a few months, a rendezvous mission becomes feasible. The flyby and rendezvous options with NEXT are shown in **Fig. 14**.

Near Earth Asteroids. With the large number of Near-Earth asteroids available, and assuming a technology demonstrator mission in 2017-2018, targets can be selected for the proper phasing. A sweep of available targets under consideration for future human mission was completed. The resulting sweep highlight the performance of 1999 AO10. The results to 1999 AO10 for the BPT-4000 and NEXT are provided in **Fig. 15**. The BPT-4000 requires 143 kg of propellant and 435 days from L1 and the NEXT option requires 130 kg of propellant and 318 days from L1 to rendezvous with 1999 AO10. Both thrusters meet all mission constraints for propellant and the total mission less than two years.

Phobos and Deimos. Another high priority target for the human architecture, and also for planetary science is the moons of Mars. To get to the moons of Mars, the Space Vehicle must spiral out from L1 to escape, rendezvous with the Mars vicinity and then spiral down to the moons and rendezvous. The Space Vehicle must spiral down to Deimos at an altitude radius of 23,460 km from the center of Mars and Phobos is at a radius of 9377 km. The

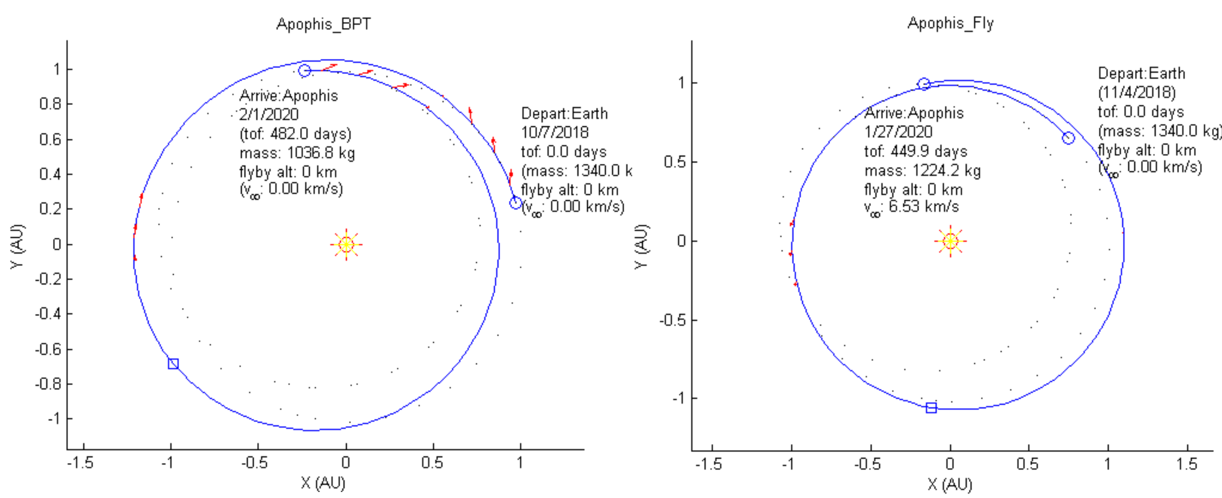

Figure 13. BPT rendezvous (left) and flyby (right) options to Apophis.

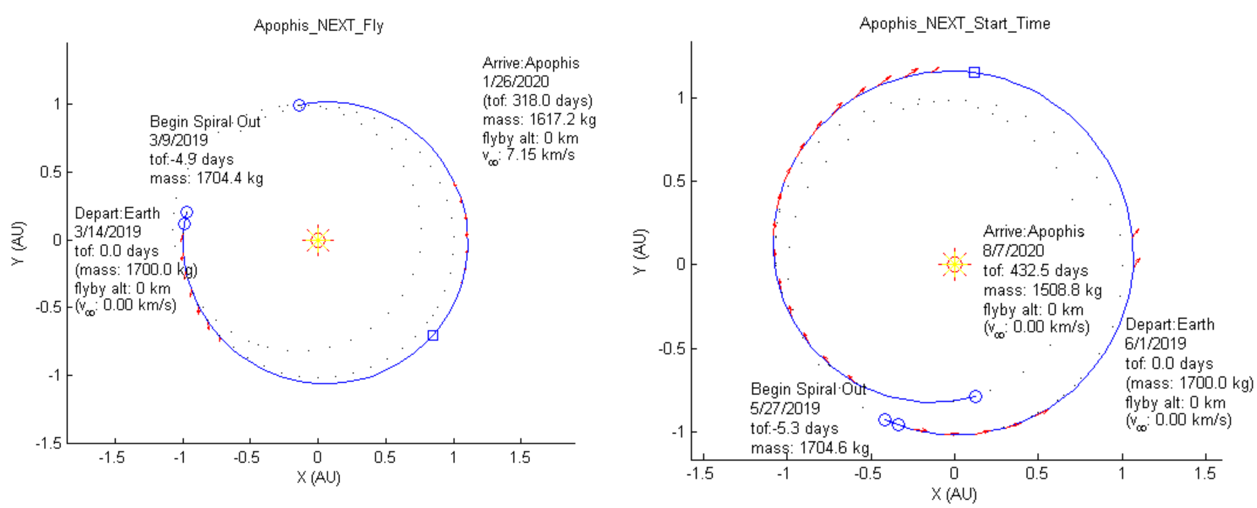

Figure 14. NEXT flyby (left) and rendezvous (right) options to Apophis.

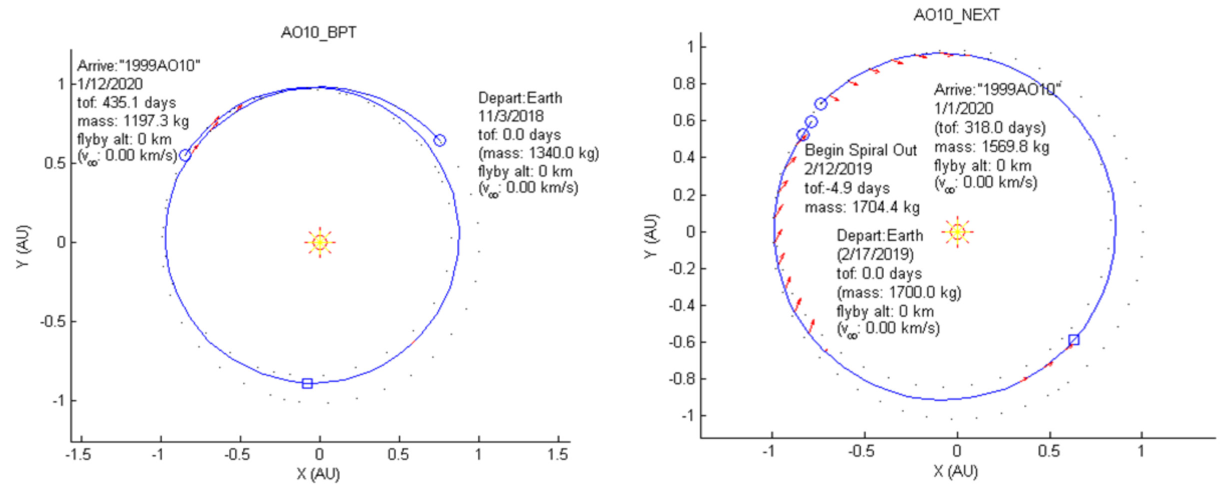

Figure 15. BPT (left) and NEXT (right) rendezvous missions with 1999 AO10.

performance of the BPT-4000 to Deimos and Phobos is shown in **Fig. 16**. The BPT-4000 does not have enough performance to rendezvous with either moon. The BPT-4000 requires 718 days to get to Mars and another 37 days to Deimos or 72 days to Phobos. The BPT-4000 also requires 335 kg of propellant to get to Mars and needs another 42.5 kg to spiral to Deimos and 83 kg to spiral to Phobos. This is well beyond the capability of the BPT-4000.

The NEXT thruster has significantly better performance to the moons of Mars. However, the trip times are still beyond the total mission duration of two years. The system does not have enough power to complete a transit, and the spiral time is significantly longer for NEXT than the BPT-4000. The results of the NEXT thruster to Deimos and Phobos are provided in **Fig. 17**. The NEXT only requires 233 kg of propellant to get to Mars and another 33 kg and 61 kg to spiral down to Deimos and Phobos respectively. The transit to Mars requires 667 days with another 105 days and 194 days to spiral down to the moons. The propellant requirement is within the NEXT capability, but the total mission time is 3.25 and 3.5 years to Deimos and Phobos respectively.

III. Conclusion

A wide range of mission trades have been completed in support of the SEP TDM definition study. The Baseline Mission is from LEO-to-GEO-to-Equatorial LEO, with an optional transit to L1. The mission is feasible using Hall thrusters or the NEXT ion propulsion system. Transfer times are shorter using the Hall option, while performance is higher using the gridded ion option. The trades demonstrated that multiple methods and tools provide consistent performance either through approximations, optimization, or simulation with provided control laws. The results indicate that the Space Vehicle will encounter on the order of 1600 shadow cycles during a transit from LEO to GEO or from GEO to LEO; while there are very few occultations at higher altitude. The trades also show that the

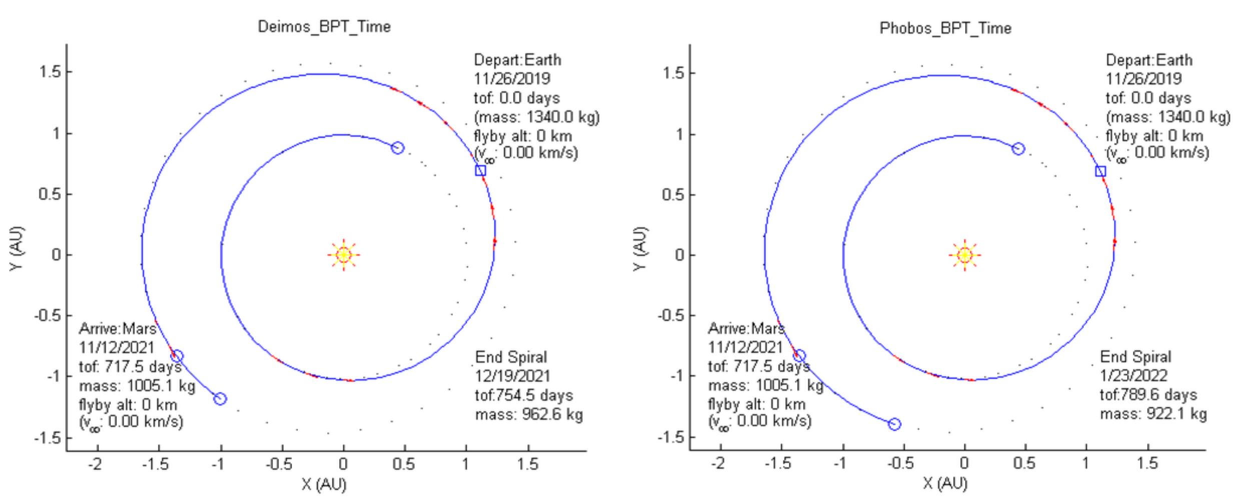

Figure 16 BPT-4000 performance to Deimos (left) and Phobos (right).

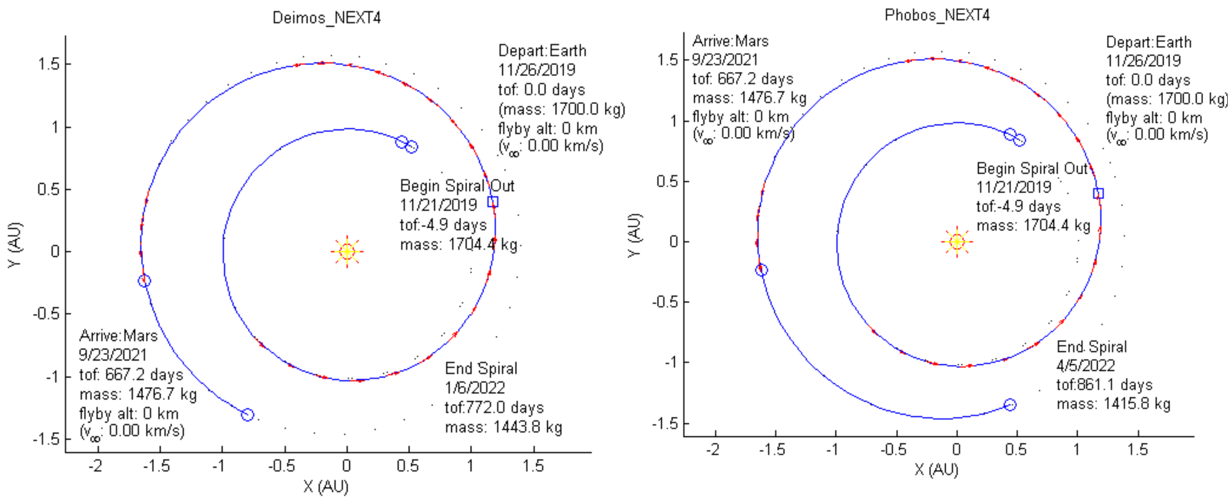

Figure 17. NEXT performance to Deimos (left) and Phobos (right).

frequency and duration of occultation is dependent on altitude, inclination, and has a seasonal dependence. The Baseline SEPTD Space Vehicle has significant flexibility with potential to perform multiple geocentric transits or demonstrate the geocentric operation before continuing on to the Moon, the moons of Mars, or Near Earth Asteroids. The technology demonstration Space Vehicle can demonstrate all phases of the human exploration "flexible path" while also demonstrating performance applicable to future science missions.

Acknowledgments

The Ball Aerospace SEP TDM Study Team would like to thank NASA Glenn Research Center for their support of this work under contract number NNC11CA36C. The authors thank the numerous members of the Ball Aerospace SEP TDM Study Team including the many companies who contributed information during the study.

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