(Preprint) AAS 15-598

IDENTIFYING ACCESSIBLE NEAR-EARTH OBJECTS FOR CREWED MISSIONS WITH SOLAR ELECTRIC PROPULSION

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This paper discusses the expansion of the Near-Earth Object Human Space Flight Accessible Targets Study (NHATS)[1] with Solar Electric Propulsion (SEP). The research investigates the existence of new launch seasons that would have been impossible to achieve using only chemical propulsion. Furthermore, this paper shows that SEP can be used to significantly reduce the launch mass and in some cases the flight time of potential missions as compared to the current, purely chemical trajectories identified by the NHATS project.

INTRODUCTION

A human visit to asteroids is a challenging endeavor for many reasons. A crucial, and complicated, first step in designing such a mission is the identification of good mission targets. Previous chemical trajectory studies have been performed to facilitate this identification.[1,2,3,4,5] To "*identify any known NEOs, particularly Near-Earth Asteroids (NEAs), that might be accessible by future human space flight missions*",[1] the NHATS project was set up under the NASA Headquarters Planetary Science Division of the Science Mission Directorate in cooperation with the Advanced Exploration Systems Division of the Human Exploration and Operations Mission Directorate. In order to determine whether an asteroid is a viable candidate, several NHATS constraints for trajectories to those asteroids must be satisfied. Those trajectories are identified using Lambert solutions, intentionally ignoring deep space maneuvers, gravity assists or continuous thrust options.[1] With the total round-trip mission duration limited to 450 days, resulting in relatively short mission legs compared to traditional robotic interplanetary missions, there is no spatiotemporal room to improve the trajectory solutions with gravity assists. Therefore, they will be ignored for this research. This paper studies the benefits of implementing solar electric propulsion (SEP); it is hypothesized that the efficiency of SEP will provide solutions that require less mission duration, less launch mass, or new opportunities when compared to chemical-only solutions.

CHEMICAL NHATS BACKGROUND

The trajectories presented here use the same framework as implemented in the original chemical-only NHATS, with the exception that SEP is used in addition to chemical propulsion. Therefore, this section will briefly introduce the constraints NHATS trajectories must comply with.

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First of all, the total mission ΔV must be smaller than or equal to 12 km/s; in addition, the launch C_3 must be smaller than or equal to 60 km^2/s^2. This total ΔV includes the Earth departure maneuver from a 400 km altitude circular parking orbit, the maneuver to match the NEA's velocity at NEA arrival, the maneuver to depart the NEA and, if necessary, a maneuver to control the atmospheric re-entry speed during Earth return, which must be less than or equal to 12 km/s at an altitude of 125 km. The total mission duration is limited to a maximum of 450 days while the minimum stay time at the NEA is 8 days. While the chemical NHATS research has trajectories between 2015 and 2040, this research focuses on the 2025-2030 time frame. This time frame fits in well with the Asteroid Redirect Mission, which is expected to launch in 2020[6] and demonstrate the use of a high-power SEP system. For more details and a discussion on the assumptions and constraints of the chemical NHATS solutions, see Reference 1.

METHOD

Low-thrust trajectory optimization can be done using a multitude of different methods and tools such as SEPTOP/VARITOP,[7] Sims-Flanagan/MALTO,[8] Mystic,[9] etc, each with their own level of complexity and accuracy.[10] Being a preliminary, proof-of-concept study, the speed with which the developed software can scan the search space is considered more important than the level of fidelity of the method. Therefore, the usage of a fast, low-fidelity low-thrust optimization procedure is preferred for this study. The chosen optimization method is a two point direct shooting method with discrete bounded control similar to the Sims-Flanagan method,[8] as implemented in the Boulder Optimization of Low-Thrust Trajectories (BOLTT) tool.[11,12,13] The results have been validated using the Evolutionary Mission Trajectory Generator (EMTG).[14]

Trajectory Representation

The Sims-Flanagan method discretizes the thrust profile using multiple impulsive maneuvers as an approximation of a continuous thrust profile. The trajectory is divided into different legs, which are bounded by control nodes that allow for a constrained discontinuous state. Such a control node can have any physical meaning such as a rendezvous or a flyby of a celestial body, a probe being released in deep space, etc. In this study, the control nodes represent the spacecraft's encounters

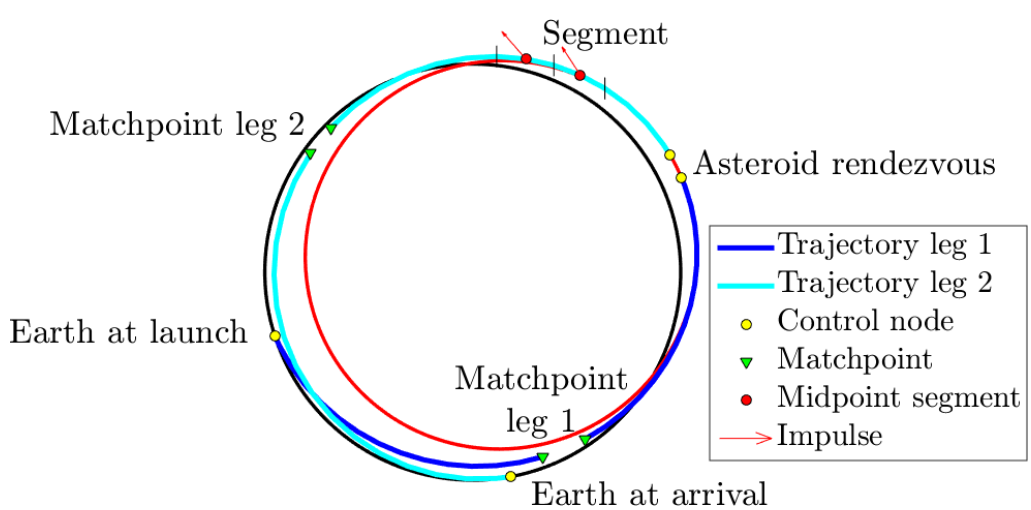


Figure 1: Structure of the Sims-Flanagan formulation on a generic trajectory (adapted from Ref 8)

with the Earth and the asteroid of interest. The control nodes bounding the first leg represent the launch from Earth and the asteroid rendezvous. The control nodes bounding the second leg represent the asteroid departure and the arrival at Earth. This has been visualized in Figure 1.

Each of the legs is discretized into segments. The thrust on a segment is represented by an impulsive maneuver at the midpoint of that segment, of which two have been depicted in Figure 1. These impulsive maneuvers influence the forward and backward propagation of the leg starting respectively at the initial control node and the arrival control node of the leg. These two propagations meet each other in the middle of the leg at a so-called match-point. The heliocentric coordinates and velocities, the hyperbolic excess velocities and the spacecraft mass at the control nodes are used as the starting point for these propagations. These initial conditions are then propagated using the control parameters on each segment, which are the magnitude and direction of the maneuver along with the specific impulse. An RK7(8)13M integration method[15] propagates the trajectories between the impulsive maneuvers; for this low-fidelity analysis the force model is simply a two-body model, but the force model may be easily enhanced for higher fidelity studies. These forward and backward propagations have to be consistent in heliocentric coordinates, velocities and mass at the match-points to ensure a continuous trajectory at the match-points.

Optimization Variables

In this study, the Earth launch and arrival dates are kept constant for the optimization of a single trajectory. The asteroid rendezvous and departure dates are allowed to change, provided that the minimum stay time of 8 days is met. These dates can then be translated into the control nodes heliocentric coordinates and velocities using JPL's HORIZONS system[16] for the Earth ephemeris and the most recent JPL orbit solution for each asteroid's ephemeris. The available power and the dry mass of the spacecraft are also kept constant. This dry mass is the payload of the mission, assumed to be 50 metric tons. As such, the mass at Earth return, which can be found using Equation (1), is fixed:

$$M_{\text{Earth return}} = M_{\text{PL}} + M_{\text{SEP}} = M_{\text{PL}} + P_0 \cdot k_{P_0} \tag{1}$$

where M_{SEP} is the mass of the SEP system, including all mass related to the power system to operate the SEP system, as well as the propulsion system itself. The SEP system mass to power ratio k_{P_0} is assumed to be 30 kg per kW[17] and P_0 is the the available power for the SEP subsystem at a heliocentric distance of 1 AU.

SEP operates on much longer time scales than chemical propulsion, requiring long duration thrust arcs that cannot be performed entirely at the most efficient point in the orbit. Furthermore, as SEP operates at much higher specific impulse levels than chemical propulsion, much less propellant mass is required for the same maneuver size. Hence, it would be unfair to compare the trajectories solely on the required ΔV, as is done for the current, chemical trajectories. Therefore, the comparison will be done based on the initial mass in low-Earth orbit (IMLEO) of the trajectory for identical payload masses for both the chemical and SEP trajectories. IMLEO is computed with Equation (2).

$$\begin{aligned}
\text{IMLEO} &= M_{\text{Earth ret}} + M_{\text{SEP prop}} + M_{\text{chem prop, esc}} + M_{\text{kick stage}} \\
&= M_{\text{Earth ret}} + M_{\text{SEP prop}} + (1 + k_{\text{KS}}) \cdot M_{\text{chem prop, esc}} \\
&= \left(M_{\text{Earth ret}} + M_{\text{SEP prop}} \right) \cdot \left((1 + k_{\text{KS}}) \cdot \exp\left(\frac{\Delta V_{\text{esc}}}{I_{\text{sp,1}} \cdot g_0} \right) - k_{\text{KS}} \right)
\end{aligned} \tag{2}$$

This mass consists of the mass at Earth return in Equation (1), the SEP propellant, the chemical propellant required to go from LEO to the required C_3 and the kick stage mass. The chemical propellant required follows the rocket equation relationship between the impulsive velocity change ΔV_{esc}, specific impulse of the chemical system $I_{\text{sp},1} = 450$ s, and g_0, the standard acceleration due to gravity on Earth at sea level. The kick stage mass is modeled as $k_{\text{KS}} = 10\%$ of the chemical propellant mass.

The parameters which can be changed to minimize the launch mass are: the mass at asteroid arrival, the mass at asteroid departure, the departure hyperbolic excess velocity at the Earth's departure, the incoming hyperbolic excess velocity at arrival at Earth and the magnitude and direction of the maneuvers. Mass at asteroid departure must be equal to the final mass at asteroid arrival, though later studies could consider a mass decrement during stay time at the asteroid. The specific impulse remains constant and has been set to 2000 s.

Constraints

In order for a trajectory to be feasible, it must meet the imposed constraints. The constraints on a trajectory for the Sims-Flanagan method depend on the type of control nodes on the legs. However, some constraints are present for all types of control nodes. These will be explained first, followed by an explanation of constraints specific for this study.

Match-point Constraints. In each Sims-Flanagan problem, constraints must be imposed such that the heliocentric coordinates and velocities and the spacecraft mass from the forward and backward propagation at the match-point are equal. These will be called the match-point constraints and are of the form

$$-\epsilon < \alpha_{\text{forward}} - \alpha_{\text{backward}} < \epsilon \tag{3}$$

where α represents the heliocentric coordinates, velocities and the mass at the matchpoint of the forward and backwards propagation and ϵ represent small values, different for each type of constraint.

Thrust Constraints. The magnitude of the maneuver applied at the midpoint of each segment is limited. It must be ensured that the magnitude of this maneuver does not exceed the maximum that the spacecraft can provide during the duration of that specific segment with a certain power level. Note that due to the usage of solar electric propulsion, the available power on each segment depends on the heliocentric distance. The thrust constraints can be found by combining Equations (4) and (5) into Equation (6):

$$P_{\text{jet}} = \eta_{\text{jet}} \cdot P = \frac{1}{2} \cdot T \cdot I_{\text{sp}} \cdot g_0 \tag{4}$$

where P_{jet} is the power of the exhaust jet, P is the available power, η_{jet} is the power conversion efficiency assumed to be 60%[18] and T the thrust

$$T = \frac{\Delta V_i \cdot m_i}{\Delta t} \tag{5}$$

with Δt the duration of the segment, ΔV_i the size of the maneuver and m_i the mass of the system before the maneuver.

$$
\begin{aligned}
P_{\text{jet}} &< \eta_{\text{jet}} \cdot P_0 \cdot \frac{\text{AU}^2}{R_i^2} \\
\Delta V_i &< \frac{2 \cdot \eta_{\text{jet}} \cdot \text{AU}^2 \cdot \Delta t \cdot P_0}{I_{\text{sp}} \cdot g_0 \cdot m_i \cdot R_i^2}
\end{aligned} \tag{6}
$$

where P_0 has already been explained directly after Equation (1) and R_i is the heliocentric distance in AUs.

Study Specific Constraints. Besides the general constraints, there are also some case specific constraints. At launch from Earth, constraints are active to ensure that the C_3 is smaller than 60 km^2/s^2, a constraint imposed by the NHATS framework.[1] Furthermore, the declination of the launch asymptote (DLA) has been limited to be between -28.5° and 28.5°, as a launch from Kennedy Space Center is being assumed for this research. While the DLA constraint is not active in NHATS, it ensures that mass penalties for reaching declinations above the launch site are avoided. The asteroid rendezvous also adds specific constraints; the stay time must be at least 8 days and the mass before and after the rendezvous must be equal. Finally, a constraint on the re-entry velocity of 12 km/s at 125 km altitude has been imposed. This has been translated using the vis-viva equation to a constraint of 4.627 km/s on the incoming hyperbolic excess velocity.

Summary of Design Parameters and Assumptions

In order to obtain realistic results, several design parameters had to be obtained or assumed. Table 1 summarizes design parameters and assumptions used in this work.

Table 1: Important design parameters and assumptions

SEP and method related parameters	
Mass to power ratio SEP system[17]	30 kg/kW
Jet efficiency[18]	60%
SEP system duty cycle[19]	90%
SEP specific impulse	2000 s
Chemical specific impulse	450 s
Mission related parameters	
Maximum total mission duration[1]	450 days
Maximum re-entry velocity[1]	12 km/s
Coast arc before Earth re-entry	2 weeks

Operational Considerations

The solution must be made robust and operationally achievable. Therefore, some operational considerations have been made.

First the thrust level is subject to a higher level of restriction. Normally, one would expect the highest allowable thrust level to be the maximum achievable thrust level on that segment. In this study, the allowable thrust is restricted to 90% of what is actually achievable. Such a duty cycle

margin has been shown to effectively prevent negative consequences of missed thrust.[19] Furthermore, this 90% does not only account for missed thrusts, but also for tasks that may interfere with thrusting periods such as communication.[19]

Secondly, coast arcs have been enforced in two places: a five-day coast arc has been imposed upon leaving Earth to account for the early checkout phase, and a two-week coast arc has been imposed just prior to re-entry to provide time for final corrections to target the entry. The reason from an operational point being to avoid additional tasks to be performed during the critical phases near the planets during the mission. Therefore, two coast arcs have been imposed on the trajectory. A five day coast arc has been imposed upon leaving Earth to account for the early checkout phase and a final coast arc of two weeks prior to arrival at Earth for re-entry operations.

The combination of all these margins makes all of the presented solutions robust, in order to ensure the safe return of the crew.

Optimization Procedure and Initial Filtering

In order to find the optimal solution to the formulated problem, the Sparse Nonlinear OPTimizer (SNOPT)[20] has been used. SNOPT uses the gradient of the constraints with respect to the state variables to find feasible and optimal solutions. SNOPT has the option to calculate these gradients using finite differencing. However, to increase the speed of the optimization procedure, it was opted to find analytical expressions for all these derivatives. These are lengthy formulas and as such, are not included in this paper. These equations were presented in Reference 12. Furthermore, SNOPT requires proper scaling of the problem. This discussion is again not included in this paper, but can be found in Reference 12.

The nature of the low-thrust optimization algorithm greatly differs from the ballistic NHATS trajectory design work; the low-thrust optimization is numerically more intensive. Due to this larger computational load, it was deemed impractical to run all possible combinations of launch date and time-of-flight as is being done with the chemical scenarios. Instead, the chemical trajectories were used to create an estimate for the required power level of the low-thrust trajectory. Based on this estimate, the clearly infeasible trajectories could be filtered out and skipped by BOLTT. The utilized filter has been adapted from Herman et al. (2014).[5]

The filter first estimates the power that may be required by the SEP system to achieve a mission. This estimated power may be computed by combining Equations (4) and (5):[5]

$$P_0 = \frac{\Delta V \cdot m_{\text{avg}} \cdot I_{\text{sp}} \cdot g_0}{2\Delta t \cdot \eta_{\text{jet}} \cdot \varepsilon_T} \tag{7}$$

where ΔV is the required change in velocity after the initial departure burn, for which the chemical trajectories provide a rough estimate. Furthermore, the duty cycle ε_T is 90% and the average mass m_{avg} is the average of the mass after the chemical departure burn, where the SEP part of the mission starts, labeled as $m_{0,\text{SEP}}$ and the mass at Earth return, given by Equation (1). Hence,

$$m_{\text{avg}} = \frac{m_{0,\text{SEP}} + M_{\text{Earth return}}}{2} = \frac{M_{\text{Earth return}}}{2} \cdot \left(1 + \exp\left(\frac{\Delta V}{I_{\text{sp}} \cdot g_0}\right)\right) \tag{8}$$

Inserting Equation (8) into Equation (7) and recognizing that $M_{\text{Earth return}}$ contains a P_0 component, one obtains:

$$P_0 = \frac{\Delta V \cdot m_{\text{PL}} \cdot \left(1 + \exp\left(\frac{\Delta V}{I_{\text{sp}} \cdot g_0}\right)\right) \cdot I_{\text{sp}} \cdot g_0}{4\Delta t \cdot \eta_{\text{jet}} \cdot \varepsilon_T - k_{P_0} \cdot \Delta V \cdot I_{\text{sp}} \cdot g_0 \left(1 + \exp\left(\frac{\Delta V}{I_{\text{sp}} \cdot g_0}\right)\right)} \tag{9}$$

This estimated power is then compared with the power limit for a particular scenario; if the estimated power is far above the limit then the case is skipped. For the launch date - TOF combinations that passed the filter, BOLTT used the chemical trajectories to obtain initial guesses for the Earth departure hyperbolic excess velocity vector, the TOF's of the different legs and the stay time of the chemical trajectories. This approach allows for a faster converging towards a solution and for a quick assessment of the search space. As an example, the functioning of the filter for asteroid 2000 SG344 will be demonstrated. The IMLEO of the chemical trajectories to this asteroid have been depicted in Figure 2.

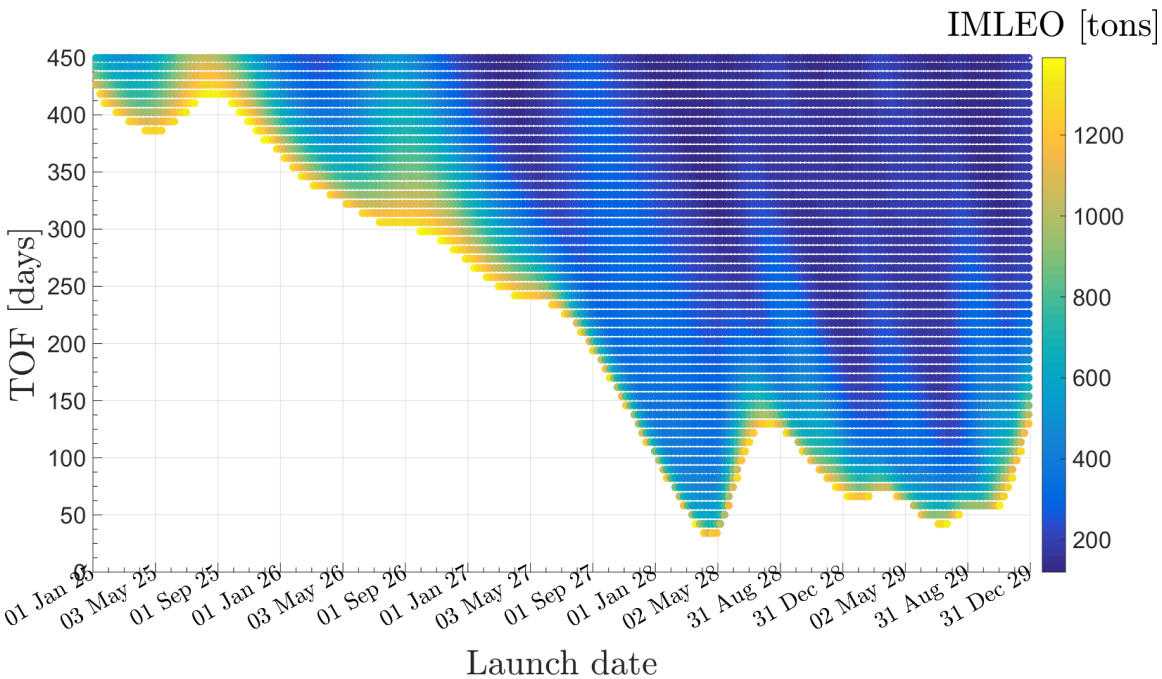


Figure 2: Chemical mission opportunities to 2000 SG344 for a payload mass of 50 tons

Using the information on the chemical trajectories depicted in Figure 2, Figure 3a shows the estimated power for launch date - TOF combinations resulting in an estimated power below 50 kW for 2000 SG344. The estimated feasible points were then passed on to BOLTT. Figure 3b shows the optimized SEP trajectories created using this output of the filter. One observes that the filter is conservative; not everything deemed feasible by the filter is also feasible in reality. However, the filter did not deem anything infeasible which was feasible in reality. This was checked by comparing the output of the BOLTT filter runs with a brute-force application of BOLTT. The BOLTT filter runs identified the same dates to be feasible and resulted in the same IMLEO's as the brute force method, while being about five times faster.

Considerations for Comparison Between Chemical and SEP Trajectories

Asteroid return missions that implement SEP will be compared with chemical versions based on their IMLEO performances. The IMLEO for the SEP trajectories has already been defined in Equation (2). For the chemical trajectories, this equation needs to be slightly modified. The Earth return mass is now identical to the payload mass and the propellant is now only chemical. The chemical propellant has been split up into two different parts though: one part for the escape burn

with the same $I_{\text{sp},1}$ of 450 s as for the SEP trajectories and the other for the other maneuvers with an $I_{\text{sp},2}$ of 350 s. This has been done under the assumption that propellant boil-off renders it impossible to use the same, efficient cryogenic chemical propellant system for maneuvers later on during the mission. The IMLEO for the chemical trajectories can hence be formulated as:

$$\begin{aligned}
\text{IMLEO} &= M_{\text{PL}} + M_{\text{chem prop}} + M_{\text{chem prop, esc}} + M_{\text{kick stage}} \\
&= M_{\text{PL}} + M_{\text{chem prop}} + (1 + k_{\text{KS}}) \cdot M_{\text{chem prop, esc}} \\
&= M_{\text{PL}} \cdot \exp\left(\frac{\Delta V_{\text{tot}} - \Delta V_{\text{esc}}}{I_{\text{sp},2} \cdot g_0}\right) \cdot \left((1 + k_{\text{KS}}) \cdot \exp\left(\frac{\Delta V_{\text{esc}}}{I_{\text{sp},1} \cdot g_0}\right) - k_{\text{KS}}\right) \qquad (10)
\end{aligned}$$

Other than this change in cost function, there is another fundamental difference between SEP and chemical trajectories that impedes the comparison between the two systems: SEP systems can only expel a certain amount of propellant in a certain time frame. This implies that increasing the propellant mass for SEP systems does not always result in more mission opportunities as also the SEP power level needs to increase to expel more propellant in a certain time frame. Hence, the selected SEP power level is a critical design variable to assess the feasibility of a low-thrust trajectory. As such, it largely influences the comparisons made between the feasibility of the chemical and SEP trajectories. Studying the entire trade space of power levels is difficult. Therefore, the comparisons are made for selected discrete values of power levels, functioning as snapshots of the entire trade space.

ASTEROID AND POWER LEVEL SELECTION

The chemical NHATS trajectories have been categorized based on their total ΔV. For this research, one asteroid with a low total ΔV will be investigated along with two asteroids with a medium total ΔV. The selected asteroids have been listed in Table 2 with their respective lowest total ΔV.

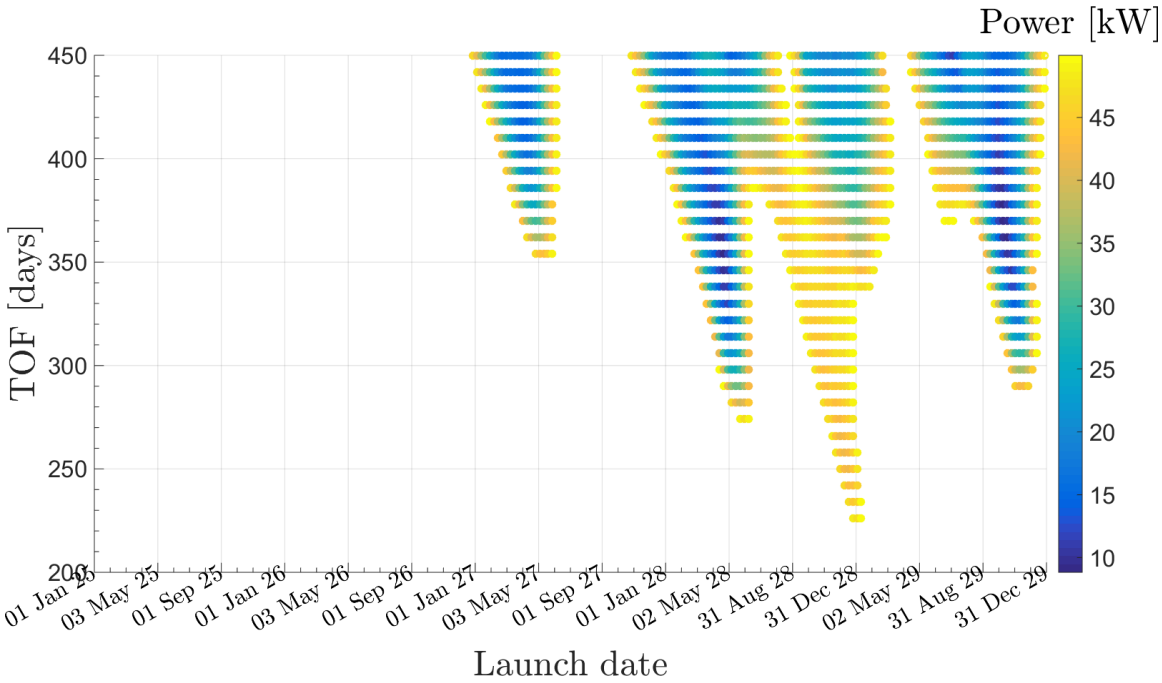
Table 2: Selected asteroids

Asteroid	Minimum ΔV (km/s)
2000 SG344	3.550
2015 BM510	5.921
2004 VJ1	6.069

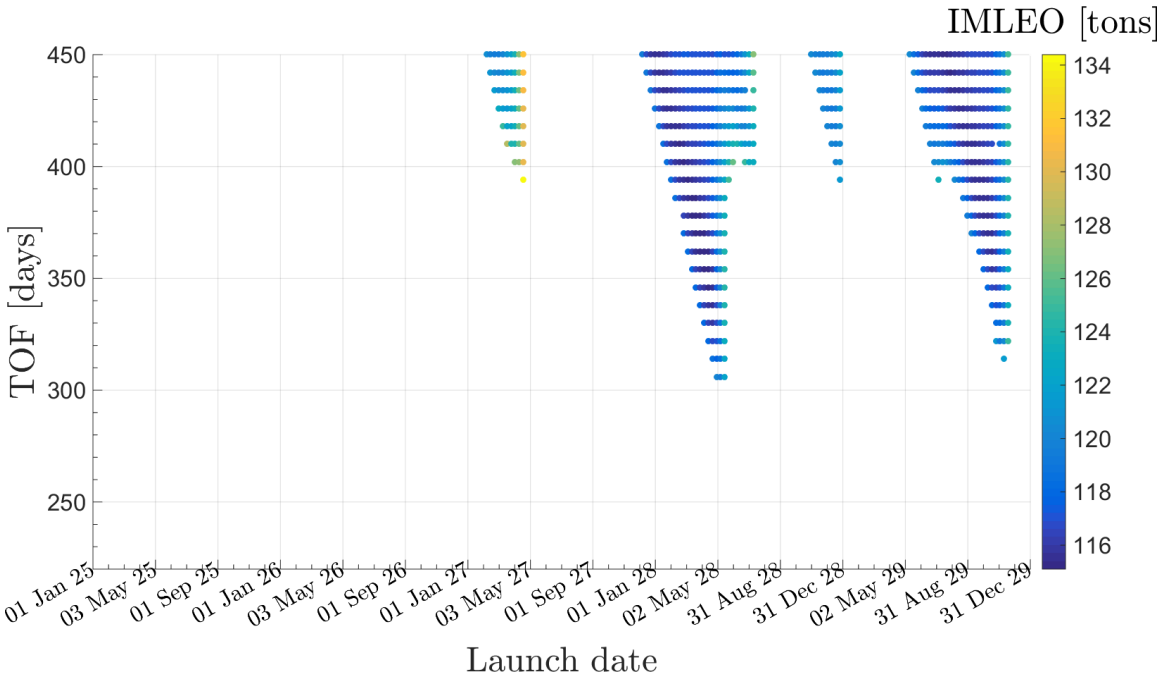
The first power level that will be assessed is 50 kW. This power level is expected to be validated by the Asteroid Redirect Mission.[6] Without making any statements on the feasibility of the use of higher power levels, this research also investigates power levels of 150 and 300 kW. This has been done to show the potential improvements to the trajectories if higher power level SEP systems are available.

RESULTS

The method described above has been applied to multiple scenarios for the selected asteroids. In the next subsections, the results for each selected asteroid will be given and analyzed.



(a) Estimated power - output of filter



(b) Results of filter runs

Figure 3: 2000 SG344 - payload mass of 50 tons

2000 SG344

Based on the results seen for the previously explained filter, it is known that with a power level of 50 kW, several launch seasons exist. The results for the launch seasons between 2025-2030 can

be found in Figure 3b. The minimal IMLEO is 115.1 tons on the 29th of March, 2028 for a TOF of 370 days. The filtered grid required a run time of 2 hrs 45 minutes on an Intel Core i7-3610QM CPU at 2.30 GHz.

A chemical trajectory to this asteroid needs at least 3.550 km/s of which 3.244 km/s is required for the Earth departure burn.[1] Using Equation (10), this results in a launch mass of 120 tons. Most of the 3.550 km/s is used to provide the required C_3. Since the SEP mission design uses the same specific impulse and needs a similar launch C_3, the mass savings using SEP come from the maneuvers that are performed after launch. In this specific scenario, those maneuvers are small; however, the SEP scenario saves at least 4 tons of IMLEO. This saving is far higher for other scenarios. However, for missions where the chemical trajectories require more maneuverability after the launch C_3, it is expected that the mass savings using SEP will increase. This can be seen in Figure 4, where for all launch date - TOF combinations that are possible for the mentioned SEP system, the differences between the chemical and SEP trajectories IMLEO are plotted. One can see that the trajectories requiring more maneuverability after the initial C_3 have higher mass savings. The minimum difference is 4.1 tons on the 22nd of April, 2028 for a TOF of 354 days. The maximum difference is 41.2 tons on the 1st of June, 2028 for a TOF of 402 days.

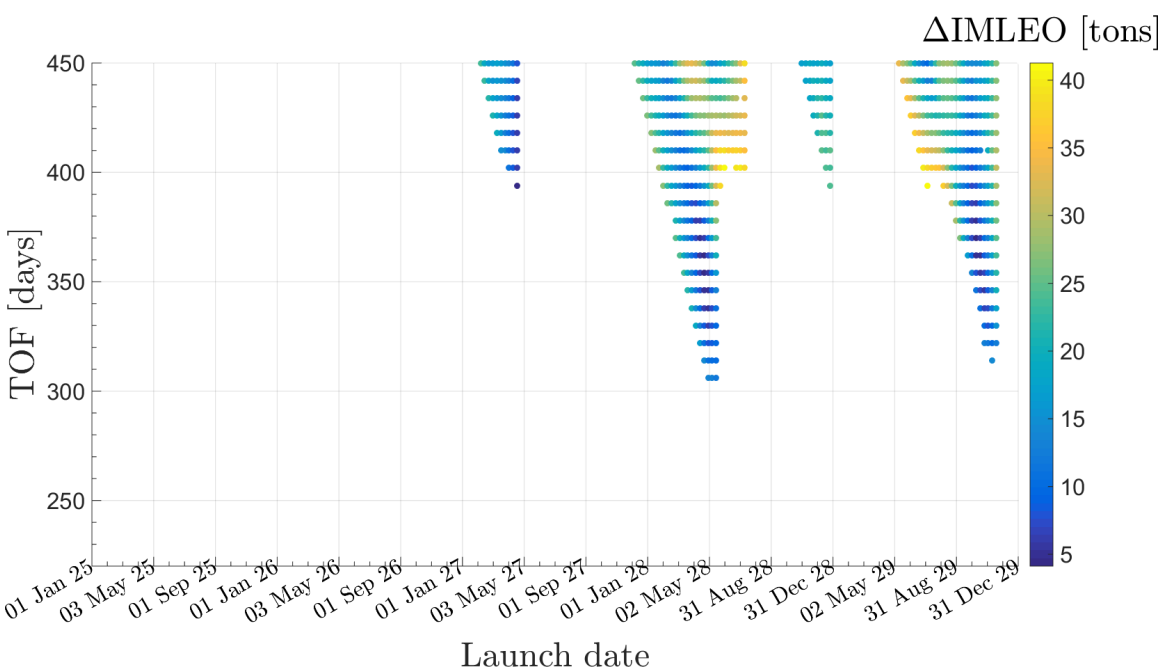
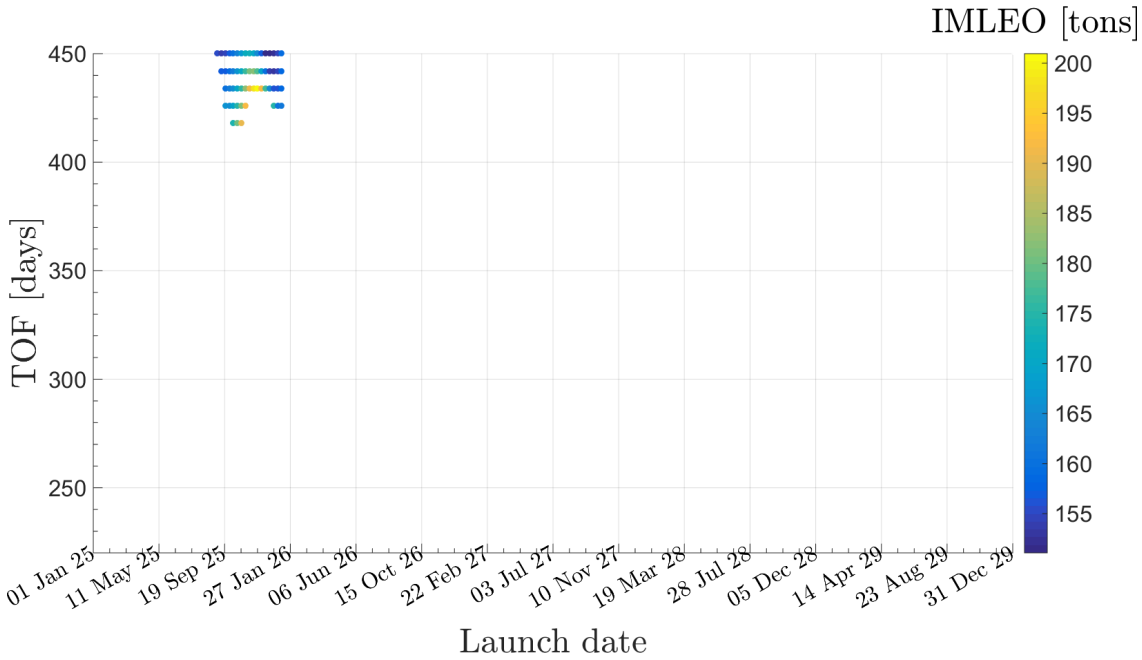


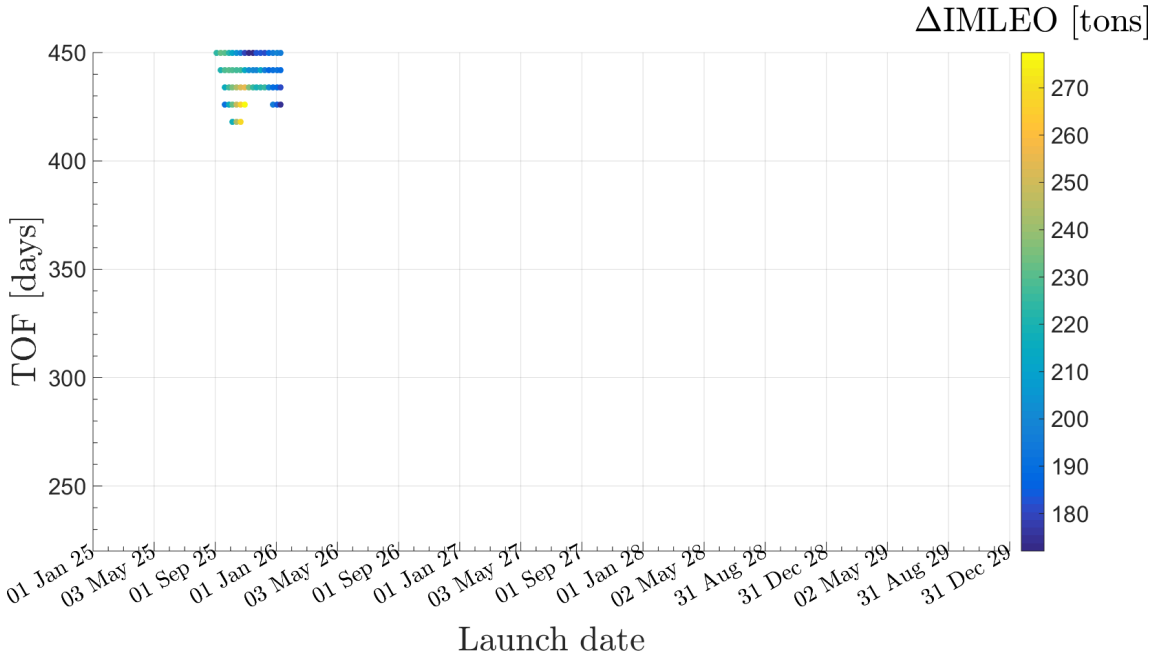
Figure 4: Difference in IMLEO between chemical and SEP trajectories to 2000 SG344 for a payload mass of 50 tons using 50 kW of power

2015 BM510

The estimated power filter has been applied to missions to 2015 BM510. It has been found that one launch season exists for a power level of 150 kW, and is shown in Figure 5a. The minimal IMLEO is 151.1 tons on the 18th of December, 2025 for a TOF of 450 days. The filtered grid required a run time of 1 hour 10 minutes.



(a) SEP mission opportunities

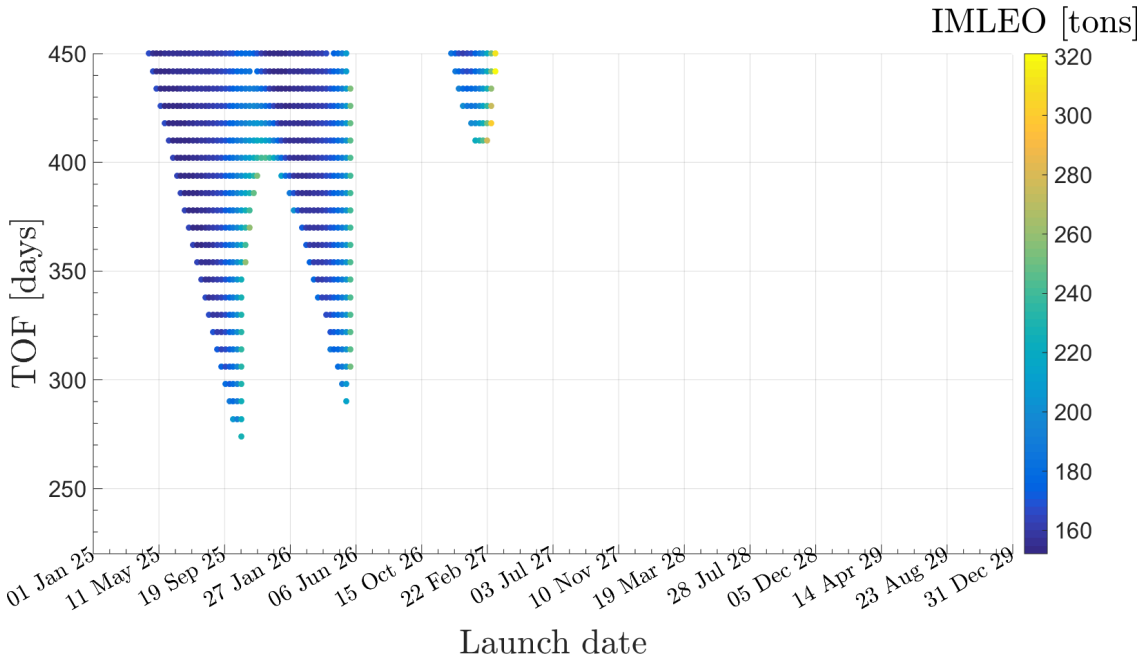


(b) Difference in IMLEO between chemical and SEP trajectories

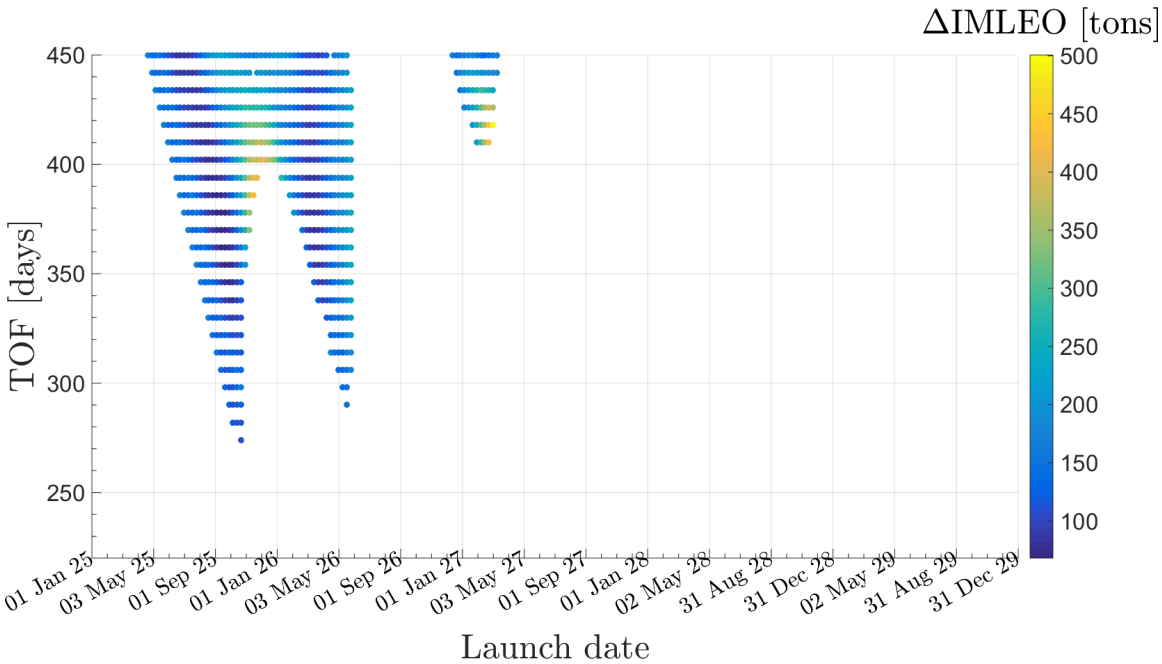
Figure 5: 2015 BM510 - payload mass of 50 tons - 150 kW of power

This launch season is relatively small. Therefore, a scenario with a power level of 300 kW has also been simulated. This results in a much larger season and a second smaller one, as can be seen

in Figure 6a. The minimal IMLEO is 152.1 tons on the 25th of June, 2025 for a TOF of 402 days. The filtered grid required a run time of 1 hour 38 minutes.



(a) SEP mission opportunities



(b) Difference in IMLEO between chemical and SEP trajectories

Figure 6: 2015 BM510 - payload mass of 50 tons - 300 kW of power

The IMLEO of the chemical trajectories to this asteroid have been depicted in Figure 7. A chemical trajectory to this asteroid needs at least 5.921 km/s of which 3.480 km/s is required for the Earth departure burn.[1] Using Equation (10), this results in a launch mass of 236 tons. This minimum ΔV trajectory has a TOF of 386 days, which is an infeasible area for the 150 kW SEP system. The chemical trajectories have an approximate IMLEO of 300-400 tons in the feasible SEP region, explaining the differences in IMLEO seen in Figure 5b. The minimum difference is 172 tons on the 8th of November, 2025 for a TOF of 450 days. The maximum difference is 277 tons on the 31st of October, 2025 for a TOF of 426 days. For the 300 kW scenario, the minimum ΔV trajectory lays in the feasible region, explaining the differences in Figure 6b. The minimum difference is 68 tons on the 21st of September, 2025 for a TOF of 362 days. The maximum difference is 501 tons on the 3rd of March, 2027 for a TOF of 418 days.

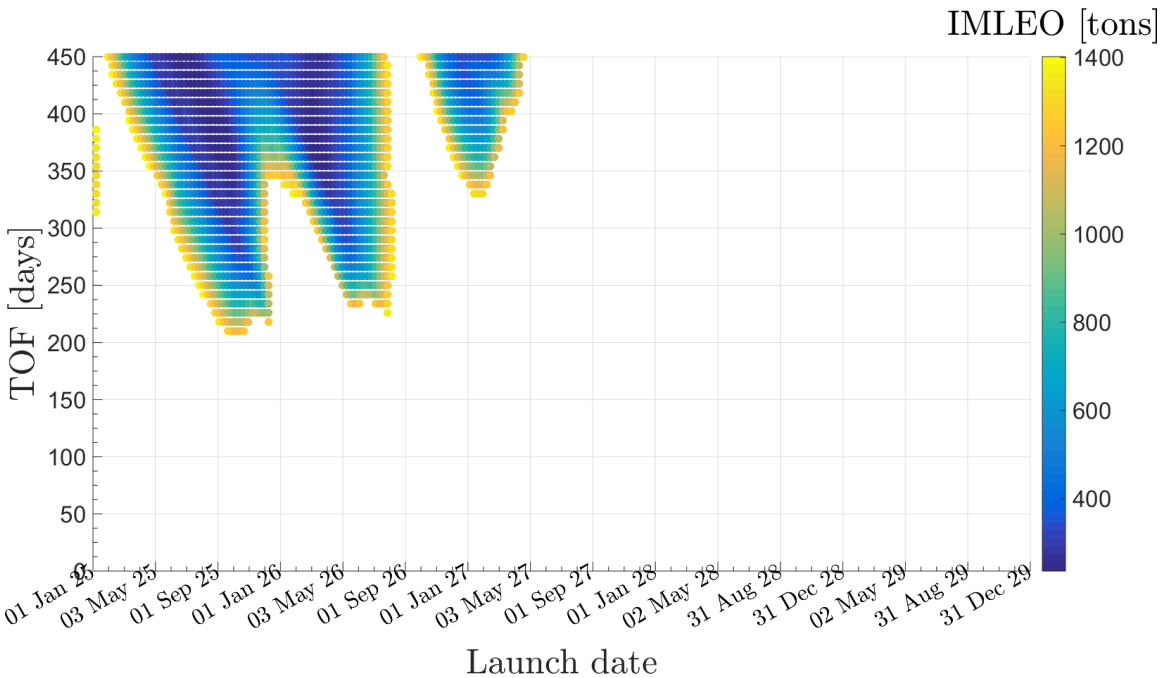
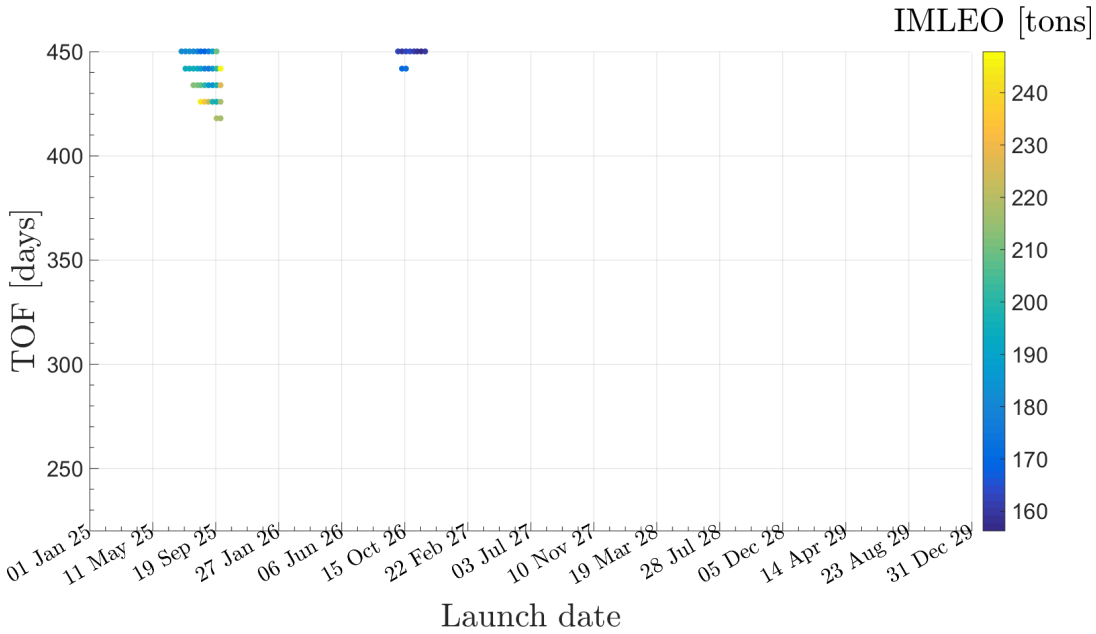


Figure 7: Chemical mission opportunities to 2015 BM510 for a payload mass of 50 tons

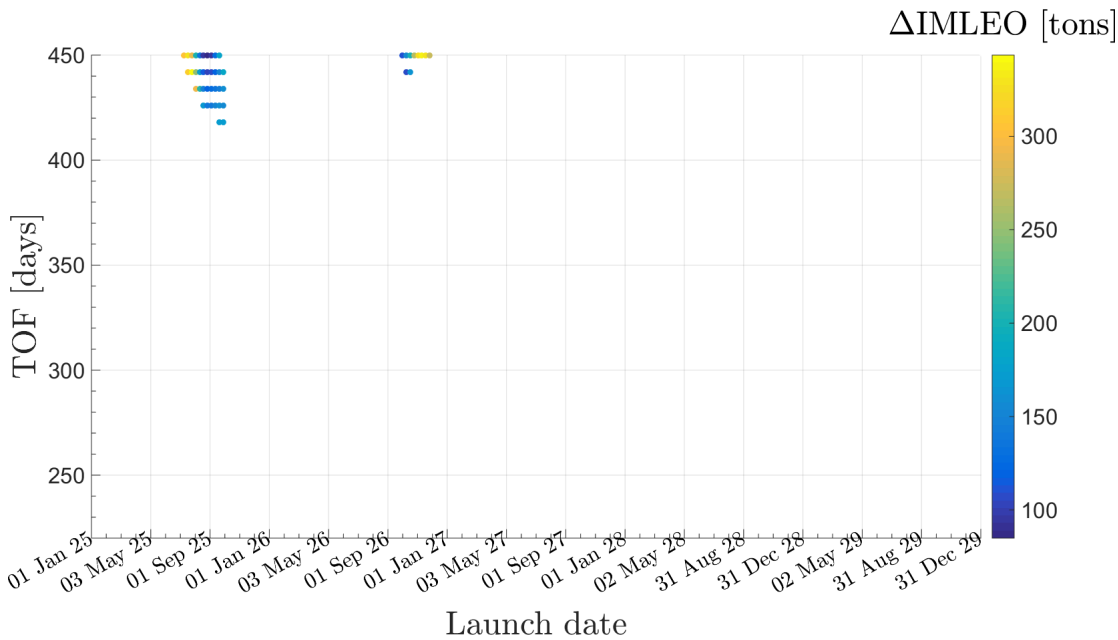
2004 VJ1

The estimated power filter has been applied to missions to 2015 BM510. It has been found that one launch season exists for a power level of 150 kW, and is shown in

Applying the filter with a power level of 150 kW to missions to 2004 VJ1 yields three launch seasons depicted in Figure 8a. The minimal IMLEO is 156.3 tons on the 19th of November, 2026 for a TOF of 450 days. The filtered grid required a run time of 37 minutes.



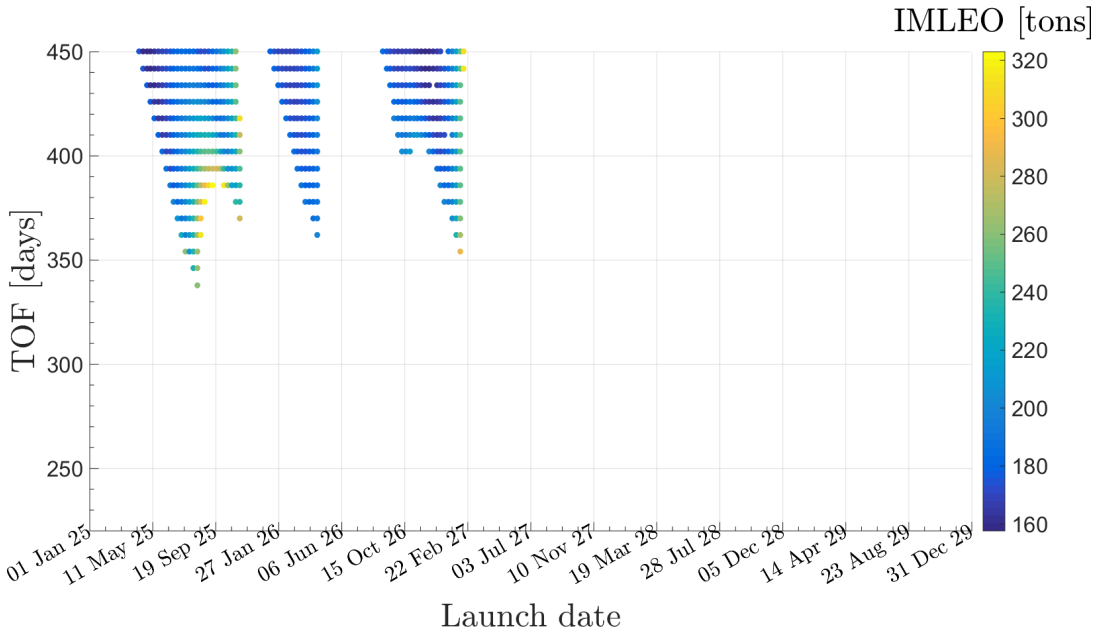
(a) SEP mission opportunities



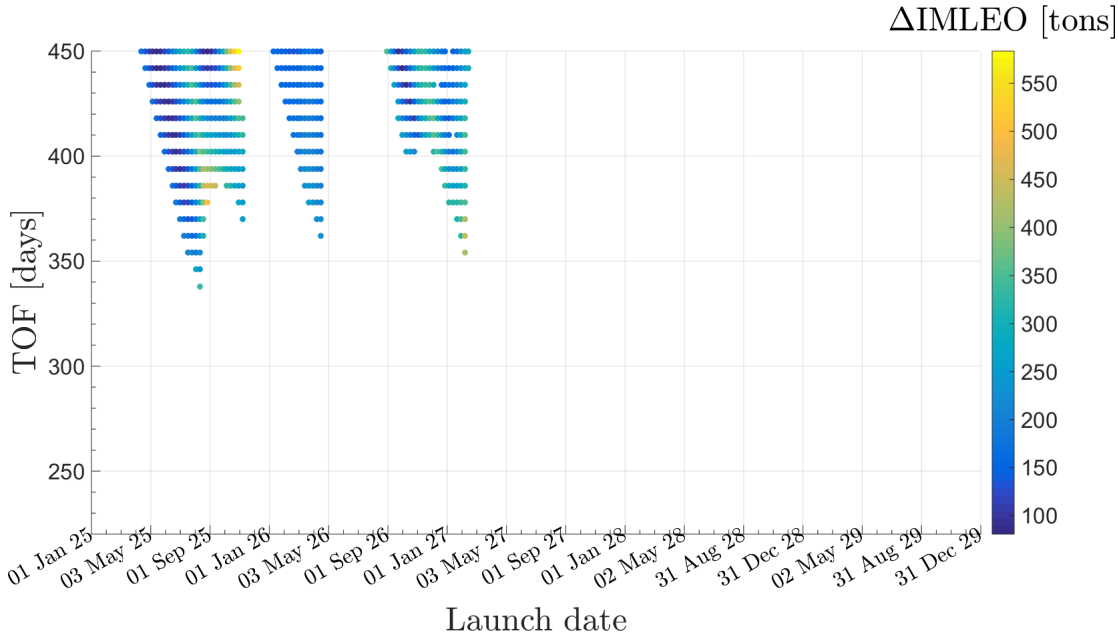
(b) Difference in IMLEO between chemical and SEP trajectories

Figure 8: 2004 VJ1 - payload mass of 50 tons - 150 kW of power

These launch seasons are relatively small. Therefore, a scenario with a power level of 300 kW has also been simulated. This results in two larger seasons and the arising of a new season, as can be seen in Figure 9a. The minimal IMLEO is 157.6 tons on the 30th of April, 2025 for a TOF of 450 days. The filtered grid required a run time of 2 hours 30 minutes.



(a) SEP mission opportunities



(b) Difference in IMLEO between chemical and SEP trajectories

Figure 9: 2004 VJ1 - payload mass of 50 tons - 300 kW of power

The IMLEO of the chemical trajectories to this asteroid have been depicted in Figure 10. A chemical trajectory to this asteroid needs at least 6.069 km/s of which 3.441 km/s is required for the Earth departure burn.[1] Using Equation (10), this results in a launch mass of 247 tons. The minimum difference is 84.9 tons on the 28th of August, 2025 for a TOF of 450 days. The maximum

difference is 343.4 tons on the 27th of July, 2025 for a TOF of 442 days. For the 300 kW scenario, the differences in IMLEO can be seen in Figure 6b. The minimum difference is 80.8 tons on the 10th of October, 2026 for a TOF of 434 days. The maximum difference is 583.5 tons on the 31st of October, 2025 for a TOF of 450 days.

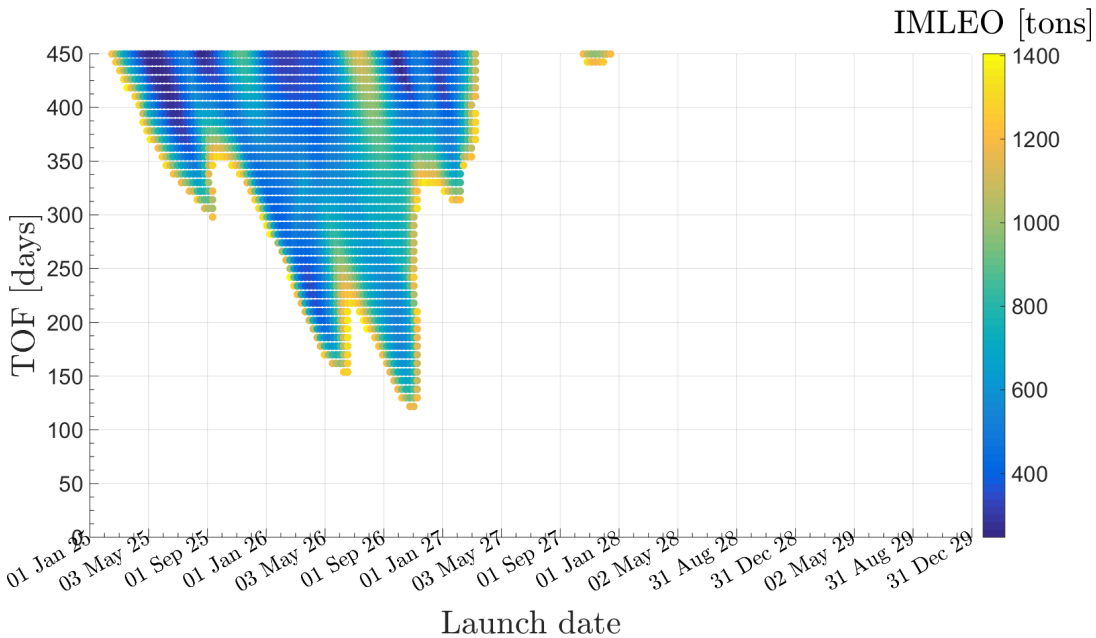


Figure 10: Chemical mission opportunities to 2004VJ1 for a payload mass of 50 tons

Discussion of Results

The large differences in IMLEO, required power, required flight time, etc. make it difficult to make comparisons between the different scenarios and to draw conclusions. To give a quick

Table 3: Minimal IMLEO for the different scenarios

Asteroid	Case	Minimal IMLEO [tons]	Launch date [mm-dd-yyyy]	TOF [days]	Minimal IMLEO 21 day launch period [tons]
2000 SG344	50 kW	115.1	03-29-2028	370	115.2
	chemical	120.0	10-10-2029	370	120.6
2015 BM510	150 kW	151.1	12-18-2025	450	152.1
	300 kW	152.1	06-25-2025	402	152.2
	chemical	236.2	09-05-2025	386	237.9
2004 VJ1	150 kW	156.3	11-19-2026	450	159.7
	300 kW	157.6	04-30-2025	450	159.9
	chemical	247.2	05-16-2025	450	251.3

overview of the different scenarios, the minimal IMLEO and its corresponding launch date and TOF have been listed for the different scenarios in Table 3. Also, the minimal IMLEO for which a 21-day launch period exists has been displayed. It can be seen that for all considered asteroids, the minimal IMLEO is lower using solar electric propulsion.

To avoid drawing conclusions based on single point comparisons, Table 4 has been constructed. In this table, different launch configurations are considered. Each specific launch configuration corresponds to a maximum IMLEO, imposing an upper limit on the IMLEO for the trajectories. Based on this upper limit, sub-launch seasons are filtered out of the launch seasons for each configuration. The table then lists the sub-launch season sizes and their minimal TOF's. For these calculations, the following launchers were considered. The SLS 1xRL, short for the SLS/iCPS 1xRL10B2 which uses one RL10B2 engine, can lift 70.0 tons into LEO.[21] The SLS 4xRL, short for the SLS/LUS 4xRL10C2 which uses four RL10 C2 engines, can lift 93.1 tons into LEO.[21]

Table 4: Launcher analysis

Launchers required	Asteroid	Case	Launch season [days]	Minimal TOF [days]
2 SLS 1xRL (140 tons)	2000 SG344	50 kW	568	306
		chemical	488	298
	2015 BM510	N.A.	N.A.	N.A.
	2004 VJ1	N.A.	N.A.	N.A.
2 SLS 4xRL (186.2 tons)	2000 SG344	50 kW	568	306
		chemical	994	146
	2015 BM510	150 kW	136	418
		300 kW	448	290
		chemical	N.A.	N.A.
	2004 VJ1	150 kW	136	434
		300 kW	408	378
		chemical	N.A.	N.A.
3 SLS 4xRL (279.3 tons)	2000 SG344	50 kW	568	306
		chemical	1232	106
	2015 BM510	150 kW	136	418
		300 kW	496	274
		chemical	200	306
	2004 VJ1	150 kW	152	418
		300 kW	488	338
		chemical	120	402

CONCLUSION

This study has demonstrated that SEP can be used to significantly improve crewed missions to asteroids. For each considered asteroid, the IMLEO was lower compared to the chemical trajectories, resulting in fewer required launches. Furthermore, it was noted that for two out of the three observed asteroids, 2015 BM510 and 2004 VJ1, the minimal TOF was lower for the 300 kW cases than for any feasible chemical missions. Altogether, this shows SEP can both reduce the launch mass cost of human missions to asteroids, as well as shorten the length of these missions. While only three asteroids were considered for this study, the results presented here suggest that many other targets in the asteroid population would enjoy similar performance improvements through the use of SEP.

ACKNOWLEDGMENTS

The authors thank Dr. George Born * for his review of this study. This work was partially funded by NASA Grant NNX14AM35H.

REFERENCES

[1] B. Barbee, P. Abell, D. Adamo, C. Alberding, D. Mazanek, L. Johnson, D. Yeomans, P. Chodas, A. Chamberlin, and V. Friedensen, "The Near-Earth Object Human Space Flight Accessible Targets Study: An Ongoing Effort to Identify Near-Earth Asteroid Destinations for Human Explorers," *2013 IAA Planetary Defense Conference*, Flagstaff, AZ, April 15-19 2013.

[2] A. Zimmer and E. Messerschmid, "Going Beyond: Target Selection and Mission Analysis of Human Exploration Missions to Near-Earth Asteroids," *Acta Astronautica*, Vol. 69, December 2011, pp. 1096–1109.

[3] A. Zimmer, *Mission Analysis and Conceptual Spacecraft Design for Human Exploration of Near-Earth Asteroids*. PhD thesis, University of Stuttgart, 2012.

[4] A. Zimmer, F. Kroll, and E. Messerschmid, "Going Beyond: Mission and System Analysis of Human Exploration Missions to Near-Earth Asteroids," *61st International Astronautical Congress*, Prague, Czech Republic, 2010, IAC-10.A5.4.4.

[5] J. Herman, A. Zimmer, J. Reijneveld, K. Dunlop, Y. Takahashi, S. Tardivel, and D. Scheeres, "Human Exploration of Near Earth Asteroids: Mission Analysis for a Chemical & Electric Propulsion Mission," *Acta Astronautica*, 2014, 10.1016/j.actaastro.2014.07.034.

[6] J. Englander, M. Vavrina, B. Naasz, R. Merill, and M. Qu, "Mars, Phobos, and Deimos Sample Return Enabled by ARRM Alternative Trade Study Spacecraft," *AIAA/AAS Astrodynamics Specialist Conference*, San Diego, CA, August 4-7 2014.

[7] C. Sauer, "VARITOP/SEPTOP/SAIL User's Manual," tech. rep., Jet Propulsion Laboratory, 2003.

[8] J. Sims, P. Finlayson, E. Rinderle, and T. Vavrina, M.A.and Kowalkowski, "Implementation of a Low-Thrust Trajectory Optimization Algorithm for Preliminary Design," *Collection of Technical Papers - AIAA/AAS Astrodynamics Specialist Conference*, Vol. 3, Keystone, CO, August 21-24, 2006, pp. 1872–1881.

[9] G. Whiffen, "Mystic: Implementation of the Static Dynamic Optimal Control Algorithm for High-Fidelity, Low-Thrust Trajectory Design," *Collection of Technical Papers - AIAA/AAS Astrodynamics Specialist Conference and Exhibit*, Keystone, Colorado, 21-24 August 2006.

[10] T. Polsgrove, L. Kos, R. Hopkins, and T. Crane, "Comparison of Performance Predictions for New Low-Thrust Trajectory Tools," *Collection of Technical Papers - AIAA/AAS Astrodynamics Specialist Conference and Exhibit*, Keystone, Colorado, 21-24 August 2006.

[11] S. De Smet, J. Parker, J. Herman, and R. Noomen, "Preliminary Design of a Crewed Mars Flyby Mission using Solar Electric Propulsion," *24th AAS/AIAA Space Flight Mechanics meeting - AAS 14-366*, Santa Fe, NM, January 26-30, 2014.

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[12] S. De Smet, "Preliminary Design of a Crewed Mars Flyby Solar Electric Propulsion Mission," Master's thesis, Delft University of Technology, December 2014. http://repository.tudelft.nl/view/ir/uuid%3A289554a2-77bf-4692-9e47-39ff75015a19/.

[13] S. De Smet, J. Parker, J. Herman, and R. Noomen, "Mission Design for a Crewed Earth-Venus-Mars-Flyby Mission using Solar Electric Propulsion," *32nd annual AAS Guidance and Control Conference - AAS 15-091*, Breckenridge, CO, January 30 - February 4, 2015.

[14] J. Englander, D. Ellison, and B. Conway, "Global Optimization of Low-Thrust, Multiple-Flyby Trajectories At Medium and Medium-High Fidelity," *24th AIAA/AAS Space Flight Meeting, Paper AAS 14-366*, San Diego, CA, January 26-30 2014.

[15] P. Prince and J. Dormand, "High order embedded Runge-Kutta formulae," *Journal of Computational and Applied Mathematics*, Vol. 7, No. 1, 1981, pp. 67 – 75, http://dx.doi.org/10.1016/0771-050X(81)90010-3.

[16] J. Giorgini, D. Yeomans, A. Chamberlin, P. Chodas, R. Jacobson, M. Keesey, J. Lieske, S. Ostro, E. Standish, and R. Wimberly, "JPL's On-Line Solar System Data Service," *Bulletin of the American Astronomical Society*, Vol. 28, No. 3, 1996, p. 1158.

[17] D. Landau and N. Strange, "Human exploration of near-Earth asteroids via solar electric propulsion," *In AAS/AIAA Space Flight Mechanics Meeting Proceedings, New Orleans, Lousiana*, February 2011.

[18] D. T. Jacobson, D. H. Manzella, and J. W. John, "The Performance and Wear Characterization of a High-Power High-Isp NASA Hall Thruster," *Collection of Technical Papers - 41st AIAA/ASME/SAE/ASEE Joint propulsion conference and exhibit*, July 2005.

[19] D. Oh, J. Snydert, D. Goebel, R. Hofer, D. Landau, and T. Randolph, "Solar Electric Propulsion for Discovery Class Missions," *33rd International Electric Propulsion Conference*, October 2013.

[20] P. Gill, "Users Guide for SNOPT Version 7: Software for Large-Scale Nonlinear Programming," tech. rep., University of California, San Diego, 2008.

[21] B. Donahue and S. Sigmon, "The Space Launch System Capabilities with a New Large Upper Stage," *AIAA Space 2013 Conference and Exposition*, San Diego, CA, September 10-12 2013.