Low Thrust Trajectory Analysis (A Survey of Missions using VASIMR® for Flexible Space Exploration - Part 2)

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ABBREVIATION

AARC – Ad Astra Rocket Company

AU – Astronomical Unit

CM - Command Module

CTV - Crew Transfer Vehicle

DRA - Design Reference Architecture

ESA - European Space Agency

ESOI - Earth Sphere of Influence

GEO - Geostationary Earth Orbit

GNC - Guidance, Navigation and Control

GSO - Geo-Synchronous Orbit

GTO - Geostationary Transfer Orbit

GUI - Graphic User Interface

IM - Initial Mass

KSC - Kennedy Space Center

LEO - Low Earth Orbit

MOI - Mars Orbit Insertion

NASA - National Aeronautics and Space Administration

NEA - Near Earth Asteroid

NEP - Nuclear Electric Propulsion

NTR - Nuclear Thermal Rocket

OTV - Orbital Transfer Vehicle

RSA - Russian Space Agency

SEP - Solar Electric Propulsion

SM - Service Module

SMA - Semi-major Axis

SOI - Sphere of Influence

TMI - Trans Mars Injection

VASIMR® - Variable Specific Impulse Magnetoplasma Rocket

VF-200 - VASIMR® flight engine at 200 kW

VX-200 - VASIMR® lab experiment at 200 kW

NOMENCLATURE

A - acceleration [m/sec^2]

a - semimajor axis [km] (SMA)

e - eccentricity

F - Thrust [N]

G - gravitational constant $(6.67\ 10^{11}$ $m^3/kg/sec^2)$

g - gravitational acceleration ($9.98\ m/sec^2$)

H - angular momentum [m^2/sec]

h - altitude [km]

I_{sp} - Specific Impulse [sec]

i - inclination [deg]

M - Mass [kg]

P - Power [kW]

R - Radius [km]

T - orbital period [sec]

t - time [sec]

V - Velocity [km/sec]

w - weight parameter

α - Specific Mass [kg/kW]

ΔV - Delta V [km/sec]

η – Efficiency or effectiveness

ν - true anomaly [deg]

Ω - longitude of ascending node [deg]

ω - angular velocity [deg/day] or argument of periapsis [deg]

Subscripts:

a - apogee

E - Earth

f - final

h - angular momentum

NR - Nuclear Reactor

p - perigee

PL - payload

pr - propellant

PT - Propellant Tank

SA - Solar Arrays

th - thruster

0 - initial

θ - azimuthal

Section 1. Flexible Mission Strategies

Space exploration can greatly benefit from the high-power electric propulsion capabilities provided by the Variable Specific Impulse Magnetoplasma Rocket (VASIMR®).[1][2] When combined with chemical rocket technologies in a flexible architecture, the VASIMR® allows new and dramatically improved mission scenarios to be considered. Employing existing state-of-the-art solar cell technology, VASIMR® is able to achieve dramatic propellant mass savings to move payloads in Earth orbit and preposition payloads for assembly near the moon, the edge of Earth's gravitational sphere of influence, and beyond. Robotic prepositioning of assets at key locations in space allows cost and risk to be reduced for later transits between staging locations. The possibility of multi-megawatt power levels using nuclear sources also allows VASIMR® technology to significantly reduce the travel time and improve abort options for human interplanetary missions between staging locations near the Moon and Mars. Power levels are considered from currently available solar technologies to those requiring future development, including nuclear-powered systems.

In Section 2 of this report, we describe the various strengths, limitations, and assumptions of mission software tools used for this study. In Section 3, we give the parameters and assumptions typically used for these studies, unless stated otherwise in the specific study. In Section 4 we describe capabilities for the transfer mission from Geostationary Transfer Orbit to Geostationary Earth Orbit (GTO-GEO mission) using a VASIMR® with realistic mass and performance values based on results from the VX-200, a VASIMR® laboratory experiment operating at 200 kW, and the VASIMR® flight design, VF-200, operating at 200 kW. In Section 5, we consider orbit transfers from LEO to Near Earth Asteroids (NEA) for Solar Electric Propulsion (SEP) robotic as well as human roundtrip missions. In Section 6, we describe Solar-powered robotic cargo missions from a Low Earth Orbit (LEO) to Mars for support of human Mars missions. In Section 7, we analyze a Nuclear Electric Propulsion (NEP) robotic mission to Jupiter, showing the advantages of Variable Specific Impulse mission compared with a Constant Specific Impulse case. In Section 8, we describe precursor interstellar NEP missions, demonstrating the advantages of high specific impulse in order to get to 1000 AU within about 30 years. In Section 9, we describe a mission from GTO to Earth-Lunar L1 Lagrange point. In Section 10, we give a brief summary and suggest scenarios that warrant more detailed study along with basic technology requirements and future needs for these missions.

Section 2. Mission Study Analysis Tools

Ad Astra Rocket Company (AARC) employs several software tools for simulating the missions with VASIMR® engines. When considering electric propulsion systems, the fundamental equations of motion must be examined to avoid implicit assumptions commonly used for chemical propulsion systems. For example, the power and propellant mass flow are somewhat independent of one another for VASIMR®, and the specific impulse can be changed during a maneuver. For these mission studies, relatively simple tools are used first to identify missions suitable for progressively higher-fidelity analysis. After a useful mission scenario is identified, more detailed surveys are performed with AdAstra3DTraj code.[3] Where warranted, still higher-fidelity analysis can then be performed using Copernicus.[4] In this section, we give a brief description of these mission analysis tools.

2.1 Copernicus

Copernicus[4] is a generalized spacecraft trajectory design and optimization system developed at the University of Texas at Austin. This software is released to NASA centers and affiliates. It is supplied with a complex GUI (Graphic User Interface), and includes variable I_{sp} (Specific Impulse) capability. Copernicus is an n-body tool with a high degree of flexibility. The user can model a number of different missions, with selectable gravitational bodies, objective functions, optimization variables, constraint options, and levels of fidelity. Additionally, it can optimize a mission for both constant and variable specific impulse trajectories. Copernicus employs multiple shooting and direct integration methods for targeting and state propagation.

2.2 AdAstra3DTraj

The Fortran code AdAstra3DTraj[3] was written by Ad Astra Rocket Company for a direct 3D trajectory simulation. It employs multiple gravitational bodies and customized navigation strategies, including variable specific impulse. It also allows for simple parametric scans and limited optimization, to help select the most viable scenarios.

Section 3. Typical Parameter Assumptions

The following parameters and assumptions are typically used throughout this paper unless specifically stated otherwise. The mass budget can be presented as $M_0 = M_{PL} + M_{pr} + M_{PT} + M_P + M_T$, where

M_{PL} is payload mass,
M_{pr} is propellant mass,
M_{PT} is propellant tank mass (assumed to be $0.1\ M_{pr}$),
$M_T = \alpha_T P$ is the mass of VASIMR® thrusters,
M_P is the mass of power system (SEP or NEP).

In SEP case, the power system is presented by the solar arrays, which has the following mass: $M_P = M_{SA} = \alpha_{SA}\ max(P)$, where α_{SA} is the mass-to-power ratio for the solar arrays in *kg/kW.* Note, that in SEP case, the power P depends in the distance to sun as $P(R) \sim 1 / R^2$, but solar arrays, in general, are built for absorbing power of certain maximal value of $P_{max} = P(R_E)$. In NEP case, the power system is presented by the nuclear reactors, which has the following mass: $M_P = M_{NR} = \alpha_{NR} P$, where α_{NR} is the mass-to-power ratio for the nuclear reactors including radiators in *kg/kW.*

The thruster specific mass, α_T is the mass-to-power ratio for the thruster package including all power handling and heat rejection equipment. In general, the thruster specific mass is a decreasing function of power $\alpha_T(P)$. For missions inside the Earth's gravitational sphere of influence (SOI) and near Earth, we consider VASIMR® power (P) levels ranging from 200 – 800 kW. Significant performance improvements are possible as α decreases, nevertheless, we use conservative nominal parameters. For near Earth missions, a conservative $\alpha_T = 10$ *kg/kW* and $\alpha_{SA} = 7$ *kg/kW,* yielding a total mass-to-power ratio, $\alpha = \alpha_{SA} + \alpha_T$, of *17 kg/kW*. A more recent study of the VF-200 design, that incorporates the latest superconducting tape technology and custom, rather than ISS radiator design, shows that the α_T*(200 kW)* is likely to be 8 rather than 10 kg/kW. The power scaling study shows that α_T*(10 MW)* is likely to be less than 4 kg/kW. Also, a recent study of solar panels by Entec company[5] indicates that SEP specific mass α_{SA} maybe in the range of 2 to 4 kg/kW.

The experimental data provided by the VX-200 in 2011 demonstrated a system efficiency, $\eta > 60\%$ and specific impulse, I_{sp}, close to *5,000 sec.*[5][6] So, for missions near Earth, with power levels in 200 – 800 kW range, we assume the total power efficiency, η, of *60%* and a specific impulse, I_{sp}, of *5,000 s*.

For robotic or cargo interplanetary missions, we consider VASIMR® power levels ranging from 1 - 5 MW. The nominal parameters for these missions are a specific impulse, I_{sp}, of *4,000* or *5,000 s* with a total power efficiency, *η,* of *60%, and a mass-to-power ratio, α (total),* of *4 kg/kW.*

For robotic precursor interstellar missions, we consider VASIMR® power levels ranging from 1 - 8 MW. The nominal parameters for these missions are a specific impulse, I_{sp}, in the range between 8,000 and 50,000 s with a total power efficiency, η, of *70%,* and a mass-to-power ratio, *α (total),* of *8 kg/kW.*

For human interplanetary missions, we consider VASIMR® power levels ranging from 10 - 200 MW. The nominal parameters for these missions are a variable specific impulse, I_{sp}, from *3,000 to 30,000 s* with a total power efficiency, η_P, of *60%,* and a mass-to-power ratio, *α* (total), less than *4 kg/kW.* A more accurate VASIMR® model, considering the power efficiency to be a function of specific impulse and power, is beyond the scope of these studies.

Section 4. GTO to GEO with plane change

4.1 GTO-GEO mission description

In this section we will describe strategies for the Geostationary Transfer Orbit to Geostationary Earth Orbit transfer mission and introduce trajectories calculated by Copernicus and AdAstra3DTraj codes. We assume that the satellite is launched from the Kennedy Space Center (KSC), with the following starting parameters:

mass $M_0 = 6,600\ t$,
perigee altitude $h_{p0} = 185\ km$,
apogee altitude $h_{a0} = 35,786\ km$,
inclination $i_0 = 28.5\ deg$,
semi-major axis $a_0 = R_E + (h_{p0} + h_{a0})/2 = 24,363\ km$,
eccentricity $e = (h_{a0} – h_{p0}) / (2a) = 0.7306$,
period $T = 37,800\ sec$,
apogee velocity $V_{a0} = sqrt(GM_E(2/(R_E + h_{a0})\text{-}1/a)) = 1.596\ km/sec$.

The ending orbit (GEO) should be circular with the following parameters:

altitude $h_f = 35,786\ km$,
inclination $i_f = 0\ deg$, and
orbital velocity $V_f = sqrt(GM_E / (R_E + h_f)) = 3.076\ km/sec$.

First, we will describe a GTO-GEO mission using two chemical burn maneuvers. Then we will present different low thrust missions, with a VASIMR engine running at power *P = 200 kW,* total power efficiency $\eta = 60\%$ and specific impulse $I_{sp} = 5,000\ sec$, i.e. VASIMR parameters measured for VX-200[5][6]. That mission produces continuous thrust $F = 2P\eta/(gI_{sp}) = 4.89\ N$, yielding initial acceleration of $A = 0.00074\ m/sec^2$. The transfer involves changes in three orbital elements of the satellite orbit: 1) semi-major axis (SMA) *a*, 2) eccentricity *e* and 3) inclination *i*. As a start, we will consider a mission with 3 stages, each one devoted to changing one orbital element, which we call as a separate maneuver strategy mission. Then we will describe combined maneuver strategy low thrust missions. We will present low thrust missions optimized for the propellant use as well as optimized for the duration time. Finally we will summarize the advantages of low-thrust missions compared with chemical burn missions.

4.2 GTO-GEO mission using chemical burns

In a case of a chemical system, the mission involves two chemical burns, as shown in Figure 1. The 1st chemical burn is needed for the GTO – Geo-Synchronous Orbit (GSO) maneuver, and performs the velocity increase at GTO apogee (in order to increase perigee) equal to $\Delta V_p = V_f - V_{a0} = 1.48$ *km/sec.* The second chemical burn is needed for the orbit inclination change (at node) $\Delta V_i = 2V_f \sin(\Delta i/2) = 1.51$ *km/sec.* The sum of two chemical burns produce the total $\Delta V = 2.99$ *km/sec.* Using the Briz-M chemical thruster[7] with specific impulse $I_{sp} = 326$ *sec,* amount of propellant needed for the GTO-GEO transfer can be calculated from the rocket equation, as $M_{pr} = M_0 (1 - \exp(\Delta V/(I_{sp}g))) = 4{,}011$ *kg.* Assuming chemical thruster mass $M_{th} = 1{,}420$ *kg* (as for Briz-M booster[7]), the payload delivered to GEO using chemical thruster is $M_{PL} = 1{,}169$ *kg*. As demonstrates by Copernicus calculation, the trajectory shown in Figure 1 for the described mission[a], takes 0.6 days.

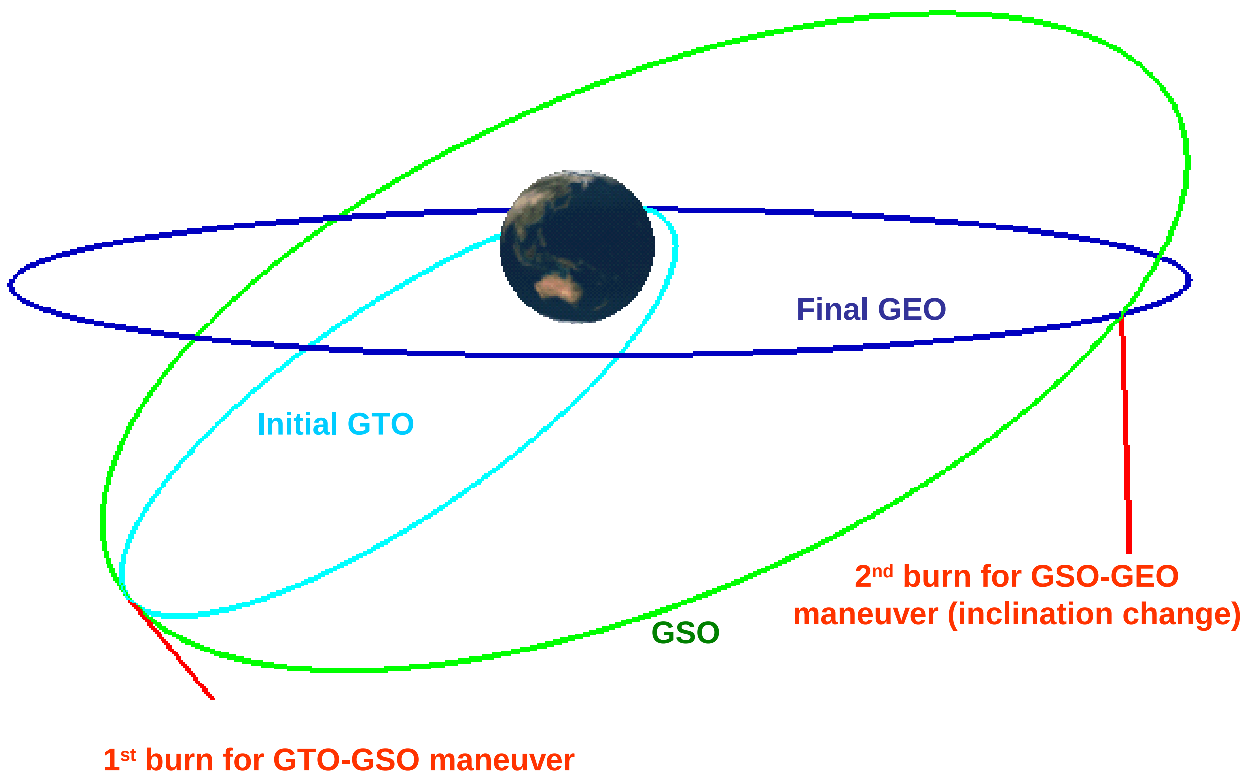


Figure 1: GTO – GEO mission using two chemical burns[a], taken *4,011* kg of propellant, lasted 0.6 days.

4.3 Separate maneuver strategy low thrust transfer

The GTO-GEO transfer involves changes in three orbital elements of the satellite orbit: 1) semi-major axis (SMA) a, 2) eccentricity e and 3) inclination i. In case of the separate maneuver strategy mission, the transfer includes 3 stages, each one devoted to changing one orbital element.

In order to increase the semi-major axis, using low thrust engine, the thrust vector should be pointed along velocity vector: $\mathbf{F}_a \parallel \mathbf{V}$. That maneuver takes $t_1 = 16.97$ *days* and *146 kg* of propellant ([b], segment 1). The Delta-V for this stage is $\Delta V_1 = t_1 A = 1.11$ *km/sec.* The Copernicus trajectory is calculated using Finite Burn Maneuver with VUW Control Frame with zero angles.

In order to reduce eccentricity (converting elliptical orbit into a circular one), the thrust vector should be directed orthogonally to the eccentricity vector $\boldsymbol{e}$ as well as to the angular momentum vector: $\boldsymbol{F_e} \perp \boldsymbol{e}$, $\boldsymbol{F_e} \perp \boldsymbol{H} = \boldsymbol{V} \times \boldsymbol{R}$. That maneuver takes $t_2 = 19.52$ *days* and *168 kg* of propellant ([b], segment 2). The Delta-V for this stage is $\Delta V_2 = t_2 A = 1.3$ *km/sec.* Since Copernicus does not have control frames related to the eccentricity vector, the IJK control frame was used, and the thrust direction angle was equal to the longitude of ascending node Ω

In order to reduce inclination, the thrust vector should be directed orthogonally velocity and radius vectors i. e. along angular momentum vector $\boldsymbol{H} = \boldsymbol{V} \times \boldsymbol{R}$ with switching direction every half-orbit (F_{iz} should have opposite sign with V_z) $\boldsymbol{F_i} \parallel -\mathrm{sign}(V_z)\, \boldsymbol{H}$. That maneuver takes $t_3 = 36.54$ *days* and *314 kg* of propellant ([b], segments 3). The Delta-V for this stage is $\Delta V_3 = t_3 A = 2.5$ *km/sec.* The 2011 version of the Copernicus (v. 3.0.1) includes capability for control frame for the inclination change: RST(INC).

The changing of all three orbital elements as described above requires total Delta V of 4.76 km/sec, 611 kg of propellant and takes 71 days. The payload delivered to GEO, using the separate maneuver strategy is 2,589 kg. Figure 2 demonstrates the separate maneuver trajectory, calculated with Copernicus code. The AdAstra3DTraj code was used as well for the same trajectory. The difference in the results of both codes is less than 2%. Each maneuver segment has duration and propellant use as shown in the Table 1. Each segment ends, when the corresponding orbital parameter: *a, e* or *i* reaches the goal value within small margin.

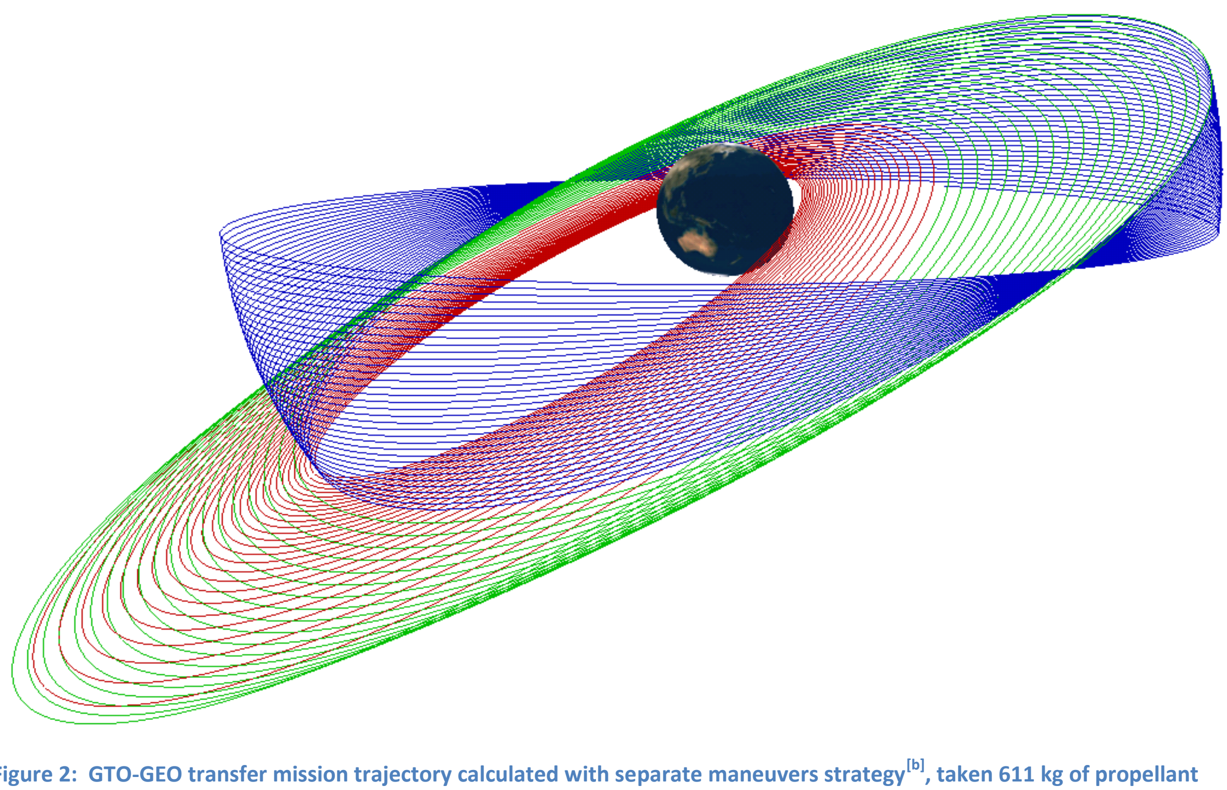


Figure 2: GTO-GEO transfer mission trajectory calculated with separate maneuvers strategy[b], taken 611 kg of propellant and lasted 71 days.

The reader can observe, that the low thrust mission for the GTO-GEO transfer is much longer than the chemical mission but takes much less of propellant, so much heavier payload can be delivered to GEO.

Both duration and propellant use can be reduced when the mission trajectory is optimized, as shown in the following sections.

#	Change	Thrust Direction	Duration [days]	M_0 [kg]	Propellant [kg]	Δ V [km/s]
1	SMA	Along V	16.95	6600	146	1.09
2	Eccentricity	Across E	19.47	6454	167	1.29
3	Inclination	Along H=VxR	34.63	6287	298	2.38
	Total		71.05		611	4.76

Table 1: Time and propellant budget for the separate maneuver strategy GTO-GEO transfer

4.4 Combined maneuver strategy transfer

In order to reduce the transfer time and propellant use, the transfer should combine maneuvers. As the Table 1 shows, the changing of inclination takes most time and propellant, so it is smart to introduce the inclination change maneuvers during the SMA and eccentricity change segments. Previous publications demonstrate that the inclination change is most effective when the orbit reaches a big radius and for the angular position to be close to the orbital nodes (where the orbit crosses the equatorial plane). The same conclusion can be made from looking at the Gauss form of the Lagrange Planetary equations for the inclination (shown in the section 4.6). Figure 3 demonstrates the trajectory calculated with Copernicus code with combined maneuver strategy, which takes 501 kg of propellant and lasted 58.3 days. The payload delivered to GEO, using the combined maneuver strategy is 2,699 kg.

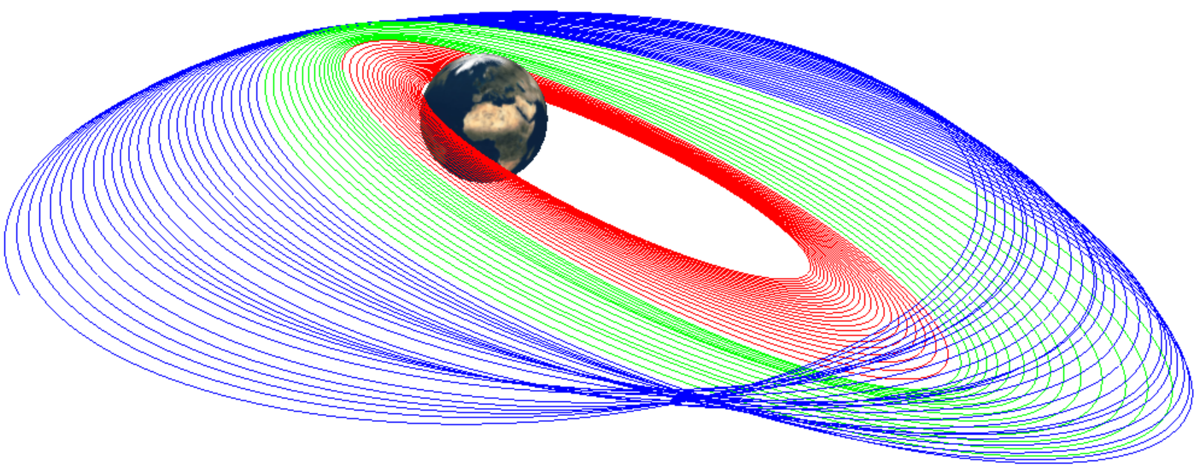


Figure 3: GTO-GEO transfer trajectory calculated with Copernicus code with combined strategy maneuvers, which takes 501 kg of propellant and lasted 58.3 days

Each maneuver segment has duration and propellant use as shown in the Table 2. The first segment utilizes only SMA increase maneuver with thrust direction along $\boldsymbol{V}$ ([c], segment 1). The second and third segments utilize rotation of the thrust vector with the frequency corresponding one rotation per current orbit. At the nodal point of the orbit, the thrust vector is oriented along angular momentum vector $\boldsymbol{H} = \boldsymbol{V} \times \boldsymbol{R}$ with a switching direction (F_{iz} should have opposite sign with V_z) $\boldsymbol{F}_i \parallel -sign(V_z)\, \boldsymbol{H}$. Second segment ends when the semi-major axis reaches the goal value ([c], segments 2 – 4 (ω = [400, 350, 300] deg/day)). The third segment ends when the eccentricity and inclination reach the goal value within a small margin ([c], segments 5 – 8 (ω = [-290, -320, -350, -365] deg/day)).

#	Change	Thrust angle	Duration [days]	M_0 [kg]	Propellant [kg]	Δ V [km/s]
1	SMA	$\alpha = 0, \beta = 0$	14.13	6600	122	0.91
2	SMA+Incl	$\beta = 90\sin(\omega(t-t_0))$	15.50	6478	133	1.02
3	Ecc+Inc	$\beta = 180+\omega(t-t_0)$	28.68	6345	247	1.94
	Total		58.31		502	3.87

Table 2: Time and propellant budget for the combined maneuver strategy GTO-GEO transfer

The reader can observe that the combined maneuver strategy mission uses 20% less propellant and 20% faster than the separate maneuver mission for the GTO-GEO transfer. The further optimization can make the mission even shorter.

4.5 Optimization for minimal trip time with continuous thrust

The AdAstra3DTraj code was used for optimizing the GTO-GEO transfer. The optimal thrust vector was used as a linear combination of the described thrust vectors: $\boldsymbol{F}_o = (w_a\boldsymbol{F}_a + w_e\boldsymbol{F}_e + w_i\boldsymbol{F}_i)/w(t)$. The weight parameters w_a, w_e, w_i and $w(t)$ are satisfied with the following requirements: 1) each of three weight parameters w_a, w_e and w_i is a positive constant until the corresponding orbital parameter reaches the target value, then it becomes equal to zero, 2) w(t) is calculated in order to enforce that the absolute value of the thrust vector equals to the given value: $|\boldsymbol{F}_o| = F$, 3) initial values of weight parameters w_a, w_e and w_i were optimized iteratively to minimize the total mission time. Figure 4 demonstrates AdAstra3DTraj trajectory calculated using combined maneuvers strategy.

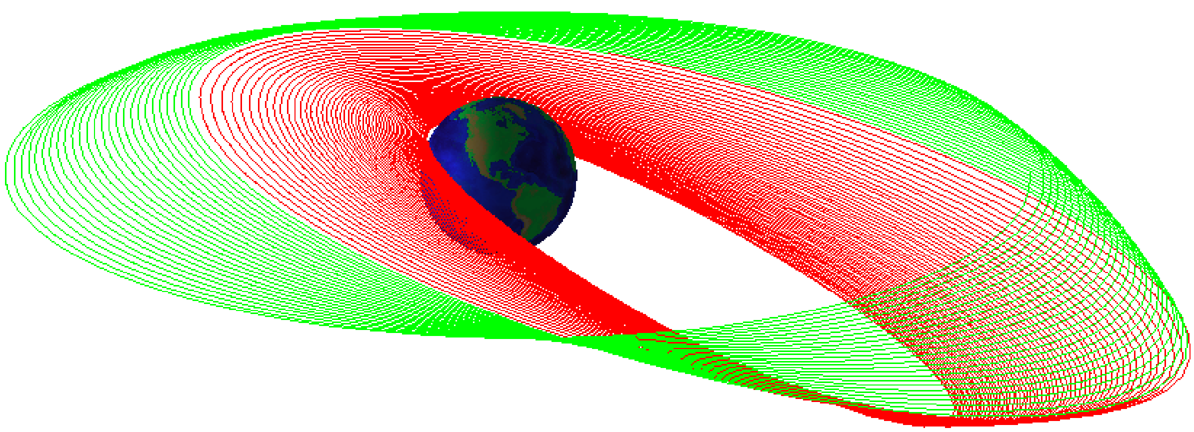


Figure 4: GTO-GEO transfer trajectory calculated with AdAstra3DTraj code with combined strategy maneuvers optimized for minimal trip time with continuous thrust, which takes 434 kg of propellant and lasted 50.5 days

#	Change	Thrust parameters	Duration [days]	M_0 [kg]	Propellant [kg]	Δ V [km/s]
1	SMA+Ecc+Incl	w_a=0.34, w_e=0.26, w_i=0.4	29.7	6600	255	1.93
2	Ecc+Incl	w_a=0, w_e=0.4, w_i=0.6	20.8	6344	179	1.40
	Total		50.5		434	3.33

Table 3: Time and propellant budget for the GTO-GEO transfer optimized for minimal duration

The transfer is presented by two segments shown in the Table 3. The first segment ends when the semi-major axis reaches its target value and the weight parameter w_a becomes zero. The optimal (for minimal propellant) low thrust mission, shown in Figure 4, takes 50.5 days (bringing Delta-V down to

3.33 km/sec) and amount of propellant is only 434 kg. The payload delivered to GEO, using the optimal duration strategy is 2,766 kg.

One can observe that the duration-optimized mission, demonstrated in this section is 30% faster than the separate maneuver mission and 14% faster than non-optimized combined maneuver mission for the GTO-GEO transfer.

4.6 Optimization for minimal propellant use with discontinuous thrust

In order to design a mission with minimal propellant use, the requirement of continuous thrust needs to be replaced by the requirement of the thrust control with highest effectiveness. Most efficient use of propellant for changing perigee maneuver corresponds to trajectories with a constant apogee. When the thrust has a relatively little effect on the perigee increase or inclination decrease, the mission should have a coasting phase with the engine turned off, as it is described by discontinuous-thrust control law derivation[8], i.e. the coasting criteria satisfies to the following conditions:

$$\frac{dr_p}{dt} < \eta_F \max\left(\frac{dr_p}{dt}\right), \quad \left|\frac{di}{dt}\right| < \eta_F \max\left(\left|\frac{di}{dt}\right|\right) ,$$

where the thrust effectiveness parameter $\eta_F < 1$.

Consider the Gauss form of the Lagrange Planetary equations for *a, e* and *i* orbital parameters[9]:

$$\frac{da}{dt} = \frac{2a^2}{\sqrt{GM_E a(1-e^2)}}\left\{e\sin(\nu)a_r + \frac{a(1-e^2)}{r}a_\theta\right\} ,$$

$$\frac{de}{dt} = \frac{1}{\sqrt{GM_E a(1-e^2)}}\left\{a(1-e^2)\sin(\nu)a_r + ((a(1-e^2)+r)\cos(\nu)+re)a_\theta\right\} ,$$

$$\frac{di}{dt} = \frac{r\cos(\omega+\nu)}{\sqrt{GM_E a(1-e^2)}}a_h .$$

The constant apogee requirement can be rewritten mathematically, as follows:

$$\frac{dr_a}{dt} = \frac{d(a(1+e))}{dt} = \frac{da}{dt}(1+e) + a\frac{de}{dt} = 0 ,$$

which defines the thrust direction, as follows

$$\frac{a_r}{a_\theta} = -\frac{2a^2(1-e^2)+(ra(1-e^2)+r^2)\cos(\nu)+r^2 e}{ra(1+e)^2\sin(\nu)} .$$

The perigee maneuver thrust is applied when the following criterion is met:

$$\frac{dr_p}{dt} = \frac{d(a(1-e))}{dt} = \frac{da}{dt}(1-e) - a\frac{de}{dt} > \eta_F \max\left(\frac{dr_p}{dt}\right) .$$

The inclination maneuver thrust is applied when the following criterion is met:

$$\left. \left|\frac{di}{dt}\right| \middle/ \max\left|\frac{di}{dt}\right| \right. = \frac{\sqrt{a\left(1-e^2\right)}\left|\cos(\omega+\nu)\right|}{\left(1+e\cos(\nu)\right)\sqrt{a_f}} > \eta_F \ .$$

Figure 5 shows the AdAstra3DTraj trajectory calculated using the minimal propellant maneuvers strategy with the following value of the effectiveness: $\eta_F = 0.9$. That mission requires 421 kg of propellant and takes 152 days. The payload delivered to GEO, using the minimal propellant strategy is 2,809 kg.

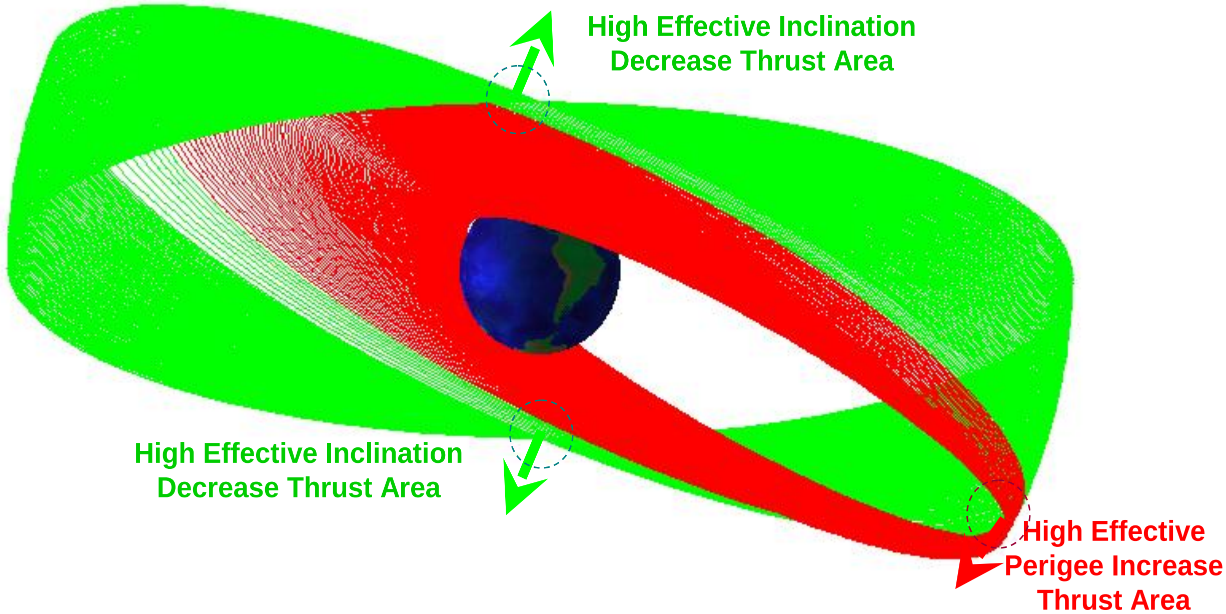


Figure 5: AdAstra3DTraj trajectory calculated using minimal propellant maneuvers strategy with effectiveness: η = 0.9.

Figure 6: Optimizing the mission for minimal propellant demonstrates the comparison of the minimal propellant strategies for different values of the effectiveness parameter. When η_F approaches 1, the propellant use approaches the minimal propellant value of 391 kg (10% less than for minimal trip time mission), corresponding to the same value of $\Delta V = 2.99$ *km/sec* as for chemical thruster mission. The disadvantage of the minimal propellant strategy is that the increase in the trip time is much higher than the reduction of the propellant use.

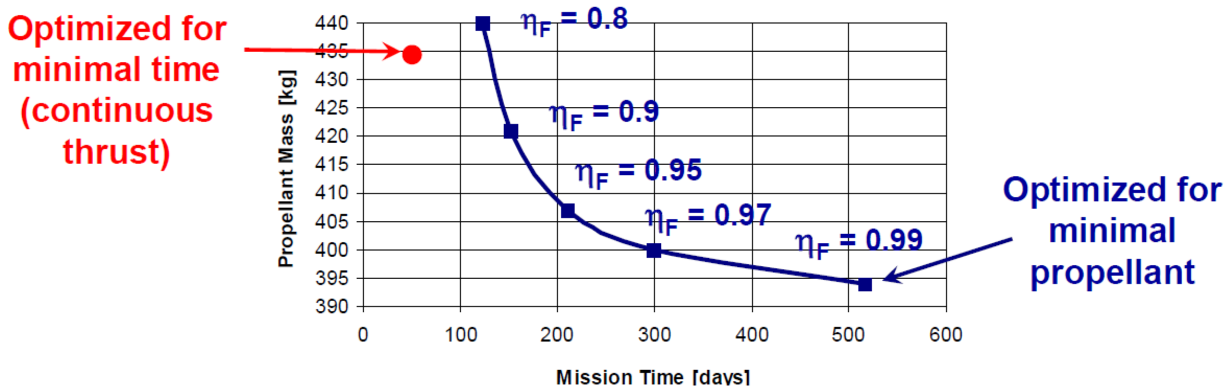


Figure 6: Optimizing the mission for minimal propellant

4.7 Impact of the specific mass on the maximal delivered payload

The optimal (for minimal mission time) low thrust mission, shown in Figure 4, takes 50.5 days (bringing Delta-V down to 3.33 km/sec) and the amount of propellant is only 434 kg. Assuming a total (propulsion and power) specific mass $\alpha = \alpha_{th} + \alpha_{SA} = 17\ kg/kW$, the payload delivered to GEO can be 2,728 kg which is 236 % of the payload delivered by a chemical thruster. Since the exact value for the specific mass is unknown, it makes sense to observe how much the delivered mass depends on the specific mass. Figure 7 demonstrates such dependence. It also shows that using Krypton, as a propellant, using 140 kW of VASIMR power, and 3,500 sec specific impulse, can provide the same 50 day mission from GTO to GEO, delivering a much higher mass.

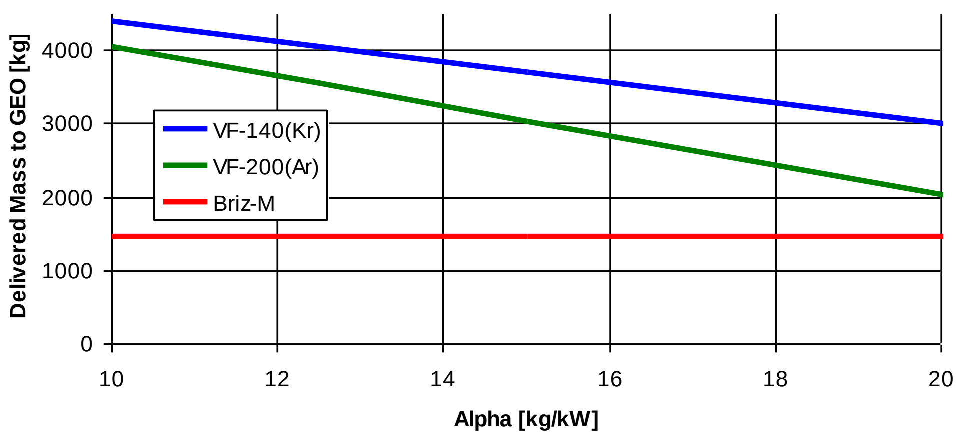


Figure 7: Impact of the VASIMR specific mass on the delivered payload to GEO

4.8 GTO-GEO mission Summary

In this section we have described different strategies for the Geostationary Transfer Orbit to Geostationary Earth Orbit transfer mission, summarized in the Table 4: Summary for GTO-GEO missions. It was shown that GTO-GEO mission using two chemical burn maneuvers, while being the fastest, utilizes the most of propellant, so can deliver the least mass of the payload. Low thrust missions use much less propellant than chemical mission and can deliver much more payload, so they are preferred when the mission duration is not critical. The fastest 200 kW VASIMR mission using Argon as a propellant can deliver within 50 days a payload 140% heavier than a chemical mission. Using Krypton as a propellant and a VASIMR power reduced from 200 kW to 140 kW, results ins the same 50 day mission from GTO to GEO, delivering a heavier payload (210% more than the chemical mission).

Mission Option (section #)	I_{sp}[sec]	ΔV[km/sec]	M_{pr}[kg]	M_{PL}[kg]	t[days]	P[kW]
Chemical Burn Mission (4.2)	326	2.99	4011	1169	0.6	
Low thrust, separate maneuver (4.3)	5,000	4.76	611	2589	71	200
Combined maneuver, non-opt. (4.4)	5,000	3.87	501	2699	58	200
Duration-optimized, Ar (4.5)	5,000	3.33	434	2766	50	200
Propellant-optimized, Ar (4.6)	5,000	2.99	391	2809	500	200
Duration-optimized, Kr (4.7)	3,500	3.33	611	3609	50	140

Table 4: Summary for GTO-GEO missions

Note, that the described low-thrust missions were calculated for continuous solar power. Shadowing was also neglected. For the high elliptical orbit, the inclusion of the shadowing effect does not significantly lengthen the mission, as the shadowing occurs only over a limited part of the orbit, when the vehicle is close to perigee and velocity is the highest. The Copernicus calculation of the same mission with shadowing effect taken into consideration makes the mission time by 6% longer (53 days for duration-optimized mission) and using 3kg more propellant.

Section 5. SEP Missions to NEA

In this section we will describe Solar Electric Propulsion missions to Near-Earth Asteroids. First we will define the objectives and list two targets for missions to NEA. Then we will describe robotic missions to NEA. We will present results on human missions to NEA, including chemical burn mission and SEP missions with VASIMR propulsion. We will demonstrate that SEP missions can be accomplished with the same duration or less, than chemical mission, and using less propellant.

5.1 NEA mission objectives

As of May 2010, more than 7,000 Near-Earth Asteroids (NEA) are known, ranging in size from 50 meters to 32 kilometers. Some NEA's orbits are very close to Earth orbits and hence have a significant impact probability in the future. Before 2004, 2000 SG_{344} was considered to have the highest (though still very low) likelihood of any NEA to impact Earth in the next 100 years. The current record for highest Torino rating (categorizing the Earth impact hazard) is held by 99942 Apophis (also known as 2004 MN_4), an about 270 m diameter NEA. Studies by NASA, ESA and RSA have described a number of proposals for deflecting Apophis or similar objects, including gravitational tractor, kinetic impact, and nuclear bomb methods.[10]

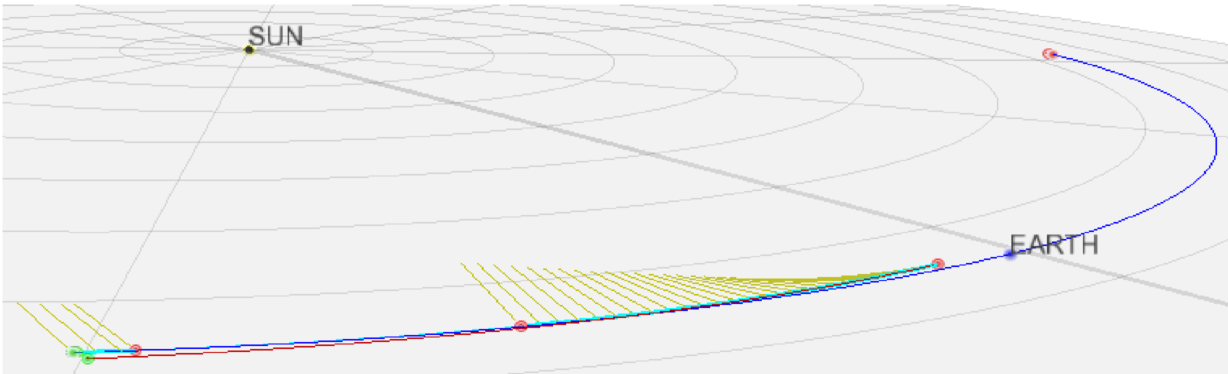


Figure 8: Heliocentric Transfer from Earth SOI to 2000 SG_{344}

5.2 Robotic missions to NEA

The robotic mission is assumed to depart from LEO with altitude of 370 km. Solar Electric Power of 200 kW and 60 % total power efficiency was assumed at constant specific impulse of 5,000 sec. Shadowing effect and inclination changes were neglected for simplicity. IM_{LEO} of 6 mT, 7 mT and 8 mT were considered.

Figure 8 demonstrates a heliocentric transfer orbit from Earth SOI to 2000 SG_{344} asteroid for IM_{LEO} of 6 mT. The mission was calculated by Copernicus, optimized for minimal propellant use. The LEO-ESOI part of the mission takes 95 days and uses 820 kg of propellant. The heliocentric transfer takes only 27 days including 10 days of the coasting and uses only 150 kg of propellant. LEO departure date was Aug 23, 2014. Note that the optimizations for minimal propellant use and for minimal duration produce very similar trajectories for this mission. Heliocentric transfer is fastlittle because the relative difference between the orbital parameters for 2000 SG344 and Earth is very small: 2% for *a,* 1% for *e* and 0.1% for *i*.

Table 5 summarizes LEO – 2000 SG_{344} robotic missions for different IMLEO (6 mT, 7 mT and 8 mT). The Payload Mass was estimated, assuming 200 kg for Avionics, 600 kg for the Solar Arrays, 800 kg for the Propulsion Module and 10 % Tank/Propellant mass fraction.

	Propellant Mass [kg]					Trip Time [days]		
IMLEO [kg]	**Spiral**	**Helio**	**Total**	**M_f[kg]**	**M_{PL}[kg]**	**Spiral**	**Helio**	**Total**
8000	**1073**	**232**	**1305**	**6695**	**4965**	**123**	**27**	**151**
7000	**952**	**215**	**1167**	**5833**	**4116**	**110**	**27**	**136**
6000	**820**	**198**	**1018**	**4982**	**3280**	**95**	**27**	**119**

Table 5: Summary for LEO - 2000 SG_{344} transfer

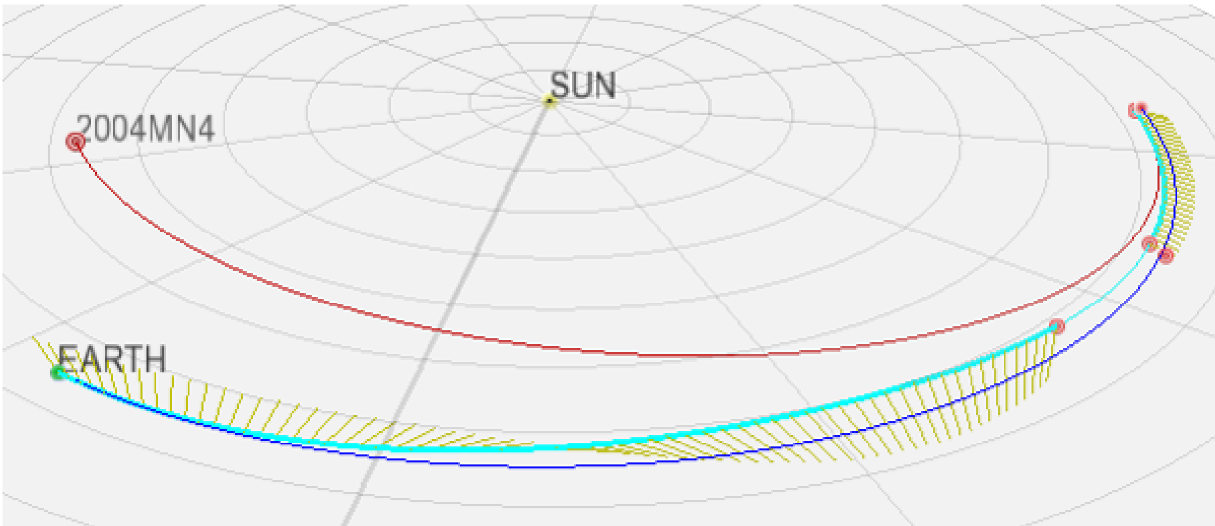


Figure 9: Helio Transfer from Earth SOI to 2004 MN_4

Figure 9 demonstrates the Copernicus trajectory for heliocentric transfer orbit from Earth SOI to 2004 MN_4 asteroid for IMLEO of 6 mT. For this asteroid, the heliocentric transfer takes only 125 days including 18 days of the coasting and uses only 975 kg of propellant. The LEO departure date was Aug 29, 2028. Again, the optimizations for minimal propellant and for minimal duration produce very similar trajectories. Note, that the relative difference between orbital parameters for 2004 MN_4 and Earth is following: 8% for *a,* 17% for *e* and 4% for *i*. That is much higher than for the previous case. Therefore, the duration and propellant use for heliocentric transfer went up by factor of 5. Table 6 summarizes LEO – 2004 MN_4 robotic missions for different IMLEO (6 mT, 7 mT and 8 mT).

	Propellant Mass [kg]					Trip Time [days]		
IMLEO [kg]	Spiral	Helio	Total	M_f[kg]	M_{PL}[kg]	Spiral	Helio	Total
8000	1073	1112	2185	5815	3997	123	140	263
7000	952	1037	1989	5011	3212	110	134	243
6000	820	975	1795	4205	2426	95	125	220

Table 6: Summary for LEO - 2004 MN_4 transfer

5.3 Human mission to NEA

In this section we will demonstrate Copernicus results on human roundtrip missions to the same nearest asteroids: 2000 SG_{344} and 2004 MN_4. The duration of the mission should be less than a year with 8 day stay on the asteroid and the payload mass should be 15 mT (which includes 10 mT for Command Module and 5 mT for the Service Module).

5.3.1 Human missions to 2000 SG_{344} asteroid on a chemical booster

NASA has conducted extensive studies of the roundtrip human mission to 2000 SG_{344} asteroid using chemical burn maneuvers. Figure 10 demonstrates the parametric scan for the departure date between 2025 and 2033 and mission duration up to 15 months. The studies demonstrated the minimal propellant mission which lasted 5 months and had departing mass 132 mT at LEO of 400 km altitude.

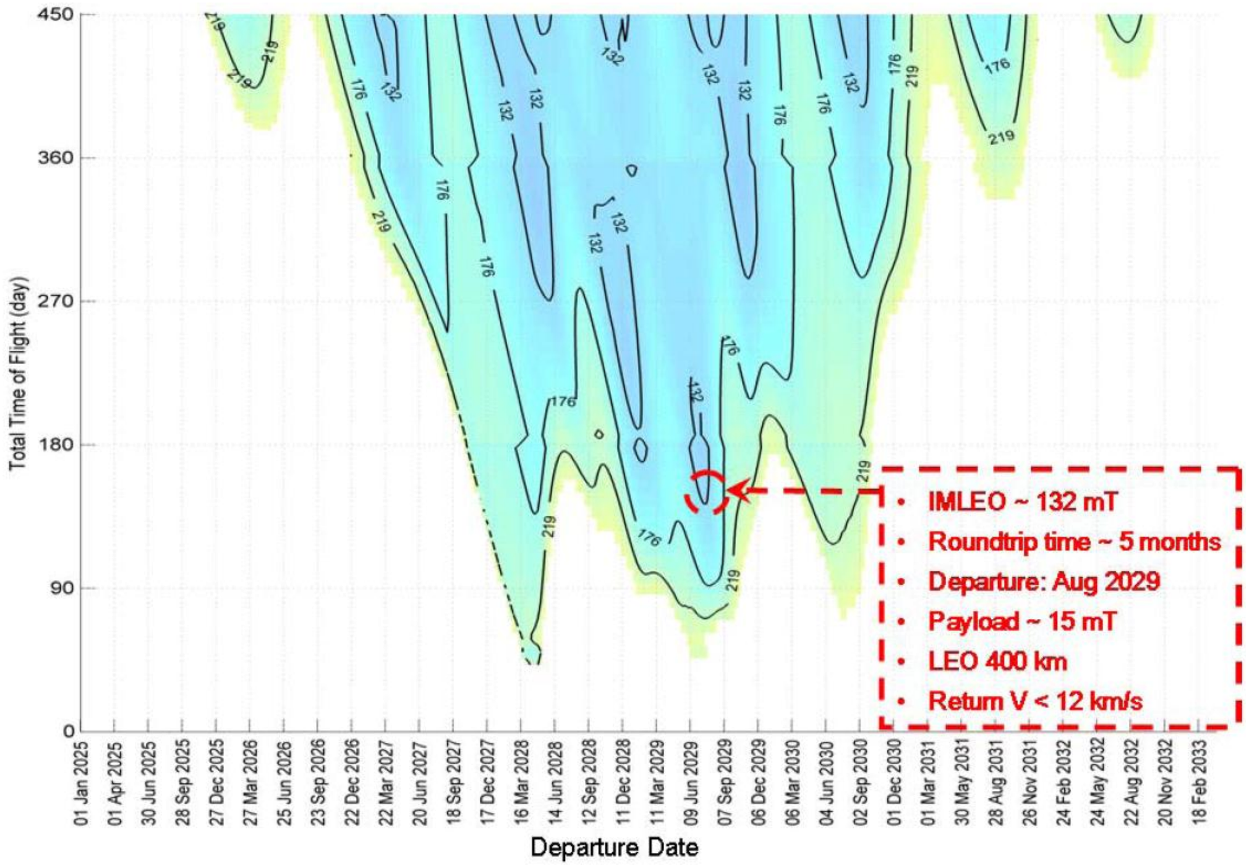


Figure 10: Parametric scan for human mission to the 2000 SG_{344} asteroid with chemical thruster. Shown are contour lines of the IMLEO, as a function of the departure date and the mission time

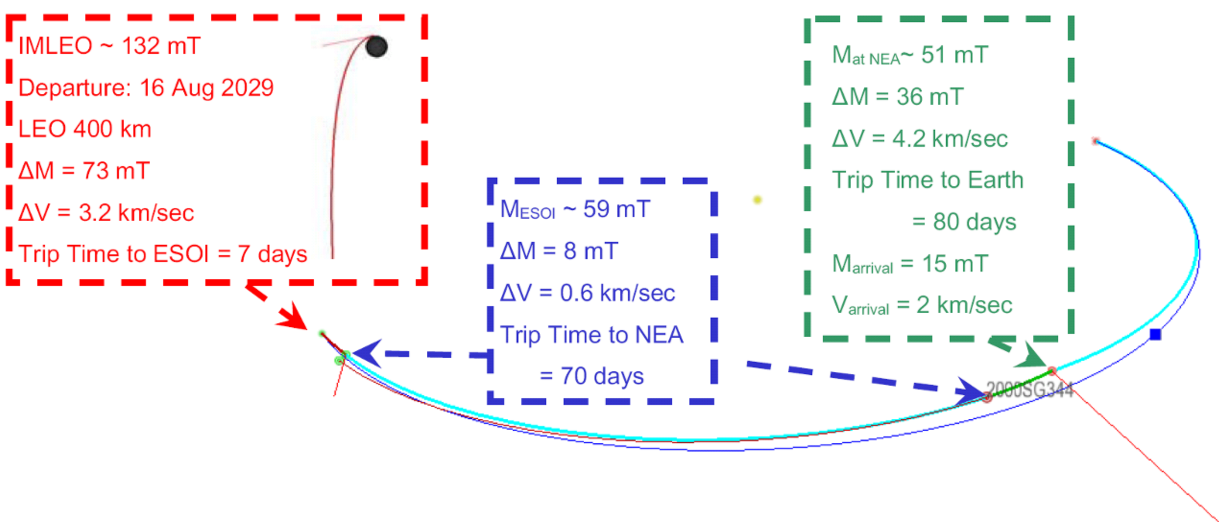


Figure 11: Human roundtrip mission to 2000 SG_{344} asteroid with 400 sec I_{sp} chemical engine

Figure 11 demonstrates the fastest chemical mission. This mission was calculated by Copernicus, optimized for the minimal duration ([o]). Note, that the human mission to 2000 SG_{344} asteroid with a chemical thruster involves three chemical burns: 1) at LEO with ΔV = 3.2 km/sec and ΔM = 73 mT in order to get to the Earth SOI, 2) at Earth SOI with ΔV = 0.6 km/sec and ΔM = 8 mT in order to get to the asteroid and 3) at the asteroid with ΔV = 4.2 km/sec and ΔM = 36 mT in order to return back to Earth.

5.3.2 SEP Human missions to 2000 SG_{344} asteroid

A solar electric propulsion mission has been designed based on a VASIMR® thruster running on Argon propellant with 5,000 sec specific impulse and 60 % power efficiency. That mission is compared with the VASIMR® mission running on Krypton with 3,500 sec specific impulse and Hall thruster with 2,000 sec specific impulse and chemical propelled mission. In order to minimize the radiation exposure at Van Allen belt, the chemical burn with 400 sec I_{sp} will be used for escaping Earth SOI or Nuclear Thermal Rocket with 900 sec I_{sp}. Several (2, 3 or 4) VF-200-type engines will be used for propulsion, so the input power will be 400, 600 or 800 kW.

Figure 12 demonstrates the heliocentric transfer portion of the human mission from Earth to 2000 SG_{344} asteroid for an IM_{SOI} of 33.5 mT and power of 800 kW. The heliocentric transfer lasts 133 days, including a 32 day Earth-NEA segment, 8 day stay on NEA and 93 day return. The mission was calculated by Copernicus, optimized for a minimal duration.

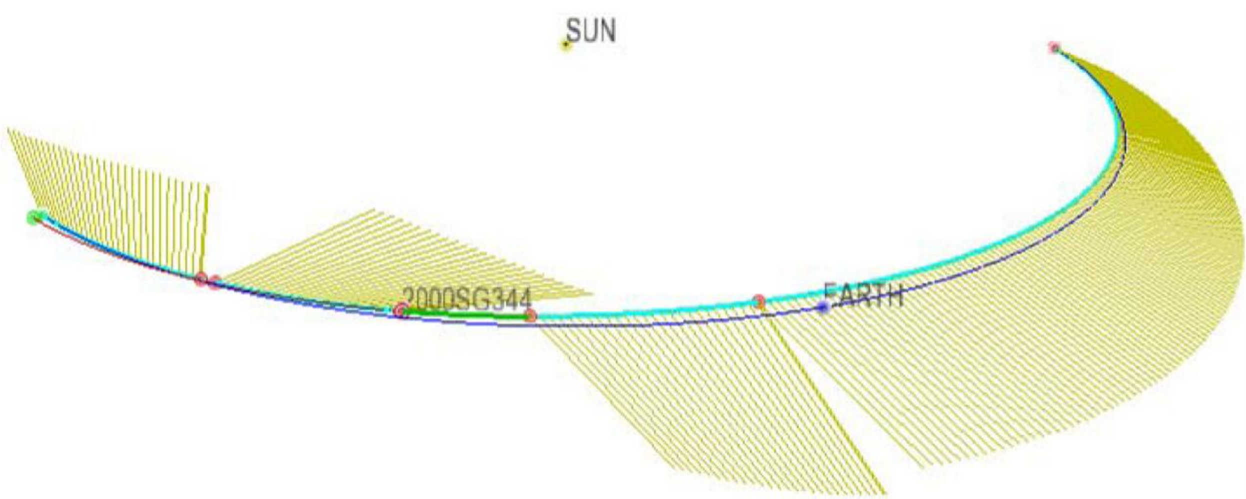


Figure 12: Human roundtrip mission to 2000 SG_{344} asteroid with 5,000 sec I_{sp} 800 kW VASIMR® engine

IM_{LEO}[mT]	P[kW]	T[days]	M_{PL}[mT]
50	800	181	29
50	600	253	33
50	400	334	37
35	800	138	16
35	600	195	20
35	400	293	23

Table 7: Parametric scan for 5000 sec I_{sp} human mission to 2000 SG_{344}

Table 7 demonstrates the duration and payload mass for human mission to the 2000 SG_{344} asteroid with different IM_{LEO} (35 or 50 mT) and VASIMR® power (400, 600 or 800 kW). Lowering the power (or increasing IM_{LEO}) makes the mission longer, but allows for higher payload.

When Krypton is used as a propellant, the specific impulse can be reduced to 3,500 sec, and the heliocentric transfer time goes down to 3 months (instead of 4.5 months for Argon with 5,000 sec specific impulse), but it requires 2 mT more propellant, so the initial mass at ESOI is 35.5 mT. Table 8 summarizes the heliocentric transfer segment of the human mission to 2000 SG_{344} asteroid using four different propulsion options: 1) 5,000 sec I_{sp} VASIMR® on Argon, 2) 3,500 sec I_{sp} VASIMR on Krypton, 3) 2,000 sec I_{sp} Hall thruster on Xenon and 4) 400 sec I_{sp} chemical booster. Note, that P = 800 kW power, 60 % efficiency was assumed for all three SEP missions.

Propulsion Option	I_{sp} [sec]	Time [months]	IM_{SOI} [mT]	M_{pr} [mT]	$V_{arrival}$ [km/sec]
VASIMR-Ar	5,000	4.2	33.5	4.3	2.9
VASIMR-Kr	3,500	3	35.5	6.0	2.5
Hall-Xe	2,000	2	42	11.8	2.5
Chem-LOX	400	5	59	44	2

Table 8: Summary for heliocentric transfer part of the human mission to 2000 SG_{344}

The following Mass Model was assumed for the results shown in Table 7: $IM_{SOI} = M+PL + M_{pr} + M_{SEP} + M_{PT} + M_{avi}$. Payload mass M_{PL} consisted of the Service Module (SM, M_{SM} = 5 mT) and the Command Module (CM, M_{CM} = 10 mT). The mass of the Solar Electric Propulsion was calculated ($M_{SEP} = \alpha_{SEP} * P$) using Specific Mass α_{SEP} = 17 kg/kW. Propellant tank mass is assumed as $M_{PT} = 0.1\ M_{pr}$. Avionics mass is assumed as M_{avi} = 0.2 mT.

The LEO-SOI transfer can be performed using a chemical booster with 400 sec specific impulse (lasting 7 days) or a Nuclear Thermal Rocket (NTR) with 900 sec specific impulse. Table 9 summarizes the initial mass requirements for LEO-SOI part of the human mission to the 2000 SG_{344}.

Propulsion Option for LEO– ESOI Transfer	Propulsion Option for Heliocentric Transfer	LEO-ESOI I_{sp} [sec]	IM_{LEO} [mT]	IM_{SOI} [mT]	M_{pr} [mT]
NTR	VASIMR-Ar	900	52.0	33.5	15.8
NTR	VASIMR-Kr	900	55.0	35.5	16.7
NTR	Hall-Xe	900	64.8	42	19.7
Chem-LOX	VASIMR-Ar	400	89.6	33.5	50.0
Chem-LOX	VASIMR-Kr	400	94.8	35.5	52.9
Chem-LOX	Hall-Xe	400	111.6	42	62.3
Chem-LOX	Chem-LOX	400	132	59	73

Table 9: Summary for LEO-ESOI transfer part of the human mission to 2000 SG_{344}

The following Mass Model was assumed: $IM_{LEO} = IM_{SOI} + M_{prop\text{-}to\text{-}SOI} + M_{tank\text{-}to\text{-}SOI} + M_{booster}$, with mass of the booster $M_{booster}$ = 1.1 mT, mass of the tank $M_{tank\text{-}to\text{-}SOI} = 0.1\ M_{prop\text{-}to\text{-}SOI}$.

5.3.3 Human mission to 2004 MN_4 asteroid

For the mission to 2004 MN_4 asteroid, NASA studies demonstrated chemical burn roundtrip mission with duration 11 months and departing mass 176 mT at LEO of 400 km altitude, as shown in the parametric scan picture on the Figure 13.

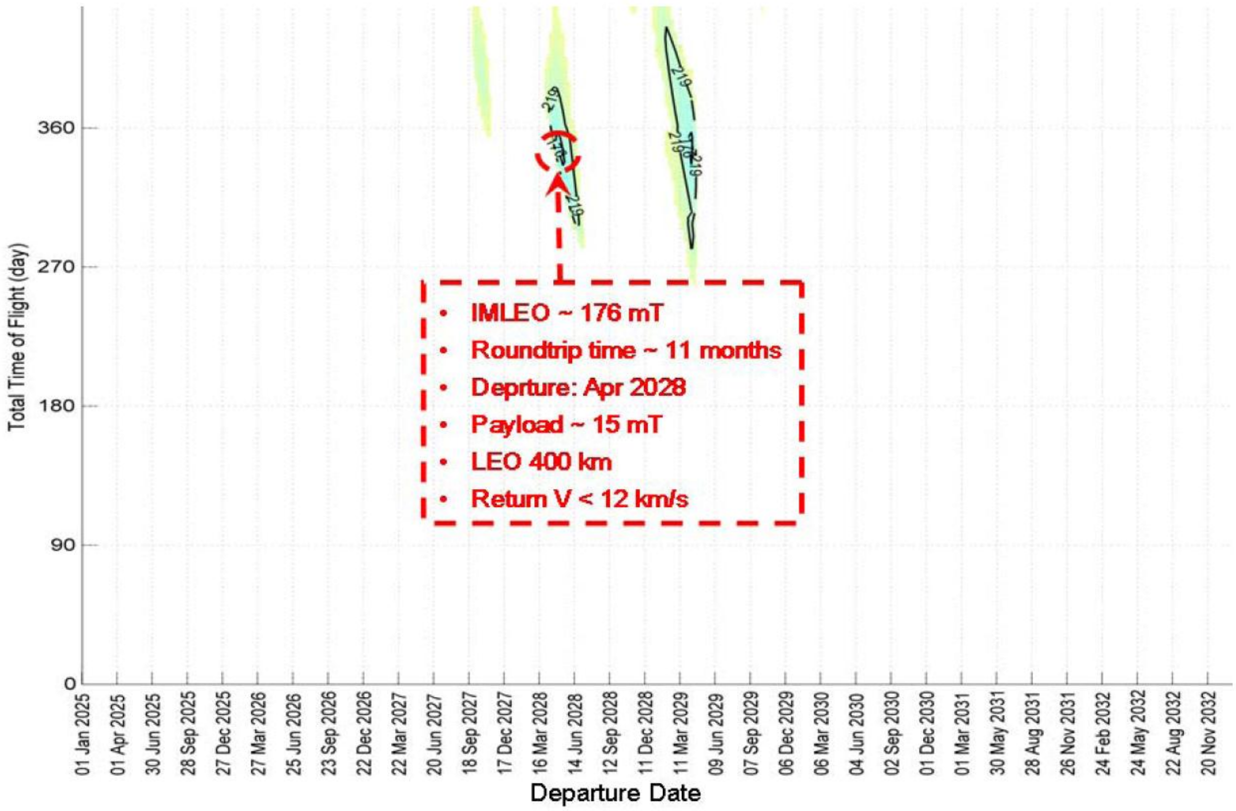


Figure 13: Parametric scan for the human mission to the 2004 MN_4 asteroid using a chemical booster. Shown are contour lines of the IM_{LEO}, as a function of the departure date and the mission time

Figure 14 demonstrates heliocentric transfer part of the human mission from Earth to 2004 MN_4 asteroid for IM_{SOI} of 35 mT and power of 600 kW. The heliocentric transfer will lasts 342 days, including 170 day Earth-NEA segment, 8 day stay on NEA and 164 day return. The mission was calculated by Copernicus, optimized for the minimal duration use. If the LEO-ESOI transfer is implemented using chemical burn, the initial mass at LEO needs to be 78 mT. If the transfer to ESOI is implemented with NTR, the IM_{LEO} needs to be 44 mT. The conclusion is that human missions to NEA with SEP for the heliocentric transfer can be implemented with much less IM_{LEO} with a possible significant reduction in the mission duration.

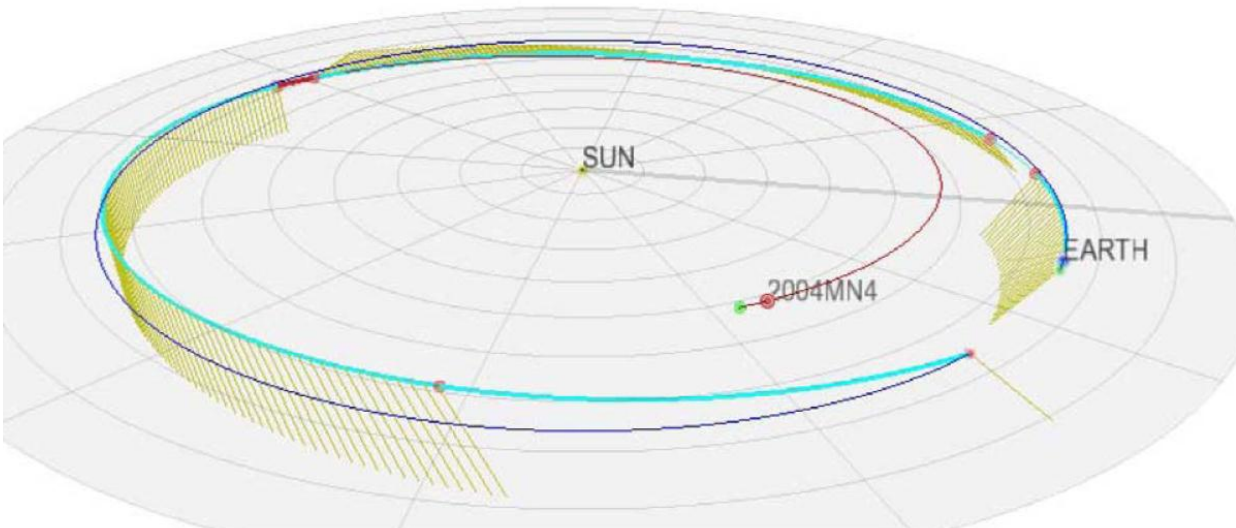


Figure 14: Human roundtrip mission to 2004 MN_4 asteroid

Section 6. SEP Human Mission to Mars

In this section we describe human missions to Mars implemented with Solar Electric Propulsion technology with a VASIMR® thruster of a few megawatts of total power. A Manned mission to Mars needs to be accompanied with cargo robotic missions, which don't need to be as fast as the manned portion of the mission, but require large delivered payload. So we will analyze SEP robotic cargo missions to Mars as well. Assume that the missions depart from a 400 km altitude Low Earth Orbit with 28.5 deg inclination, corresponding to KSC lunch site. Arrival Mars Orbit is GSO with 20,500 km altitude. For simplicity, the shadowing effect is neglected for planet orbiting and de-orbiting segments.

6.1 Power Scan for SEP VASIMR robotic missions to Mars

This subsection analyzes the robotic missions to Mars with an Initial Mass of 100 mT in LEO. The VASIMR® thruster is assumed with a constant specific impulse: I_{sp} = 5,000 sec. The SEP power is scanned in the range of 0.5 – 4 MW. The total system efficiency is assumed of 60%. The power and propulsion Specific Mass is 10 kg/kW.

Figure 15 and Table 10 summarize the power scan results for the SEP robotic mission to Mars. The scan demonstrates that the mission duration goes down very fast in the power range between 0.5 MW and 2 MW and relatively stays flat between 2 MW and 4 MW. The propellant use is almost the same for all analyzed power values. The delivered payload is linearly going down with power increased. The ratio of payload to duration has maximal value of 3.7 mT/month corresponding 2MW of the power, which makes the 2 MW power level be the optimal power value for the given mission assumptions.

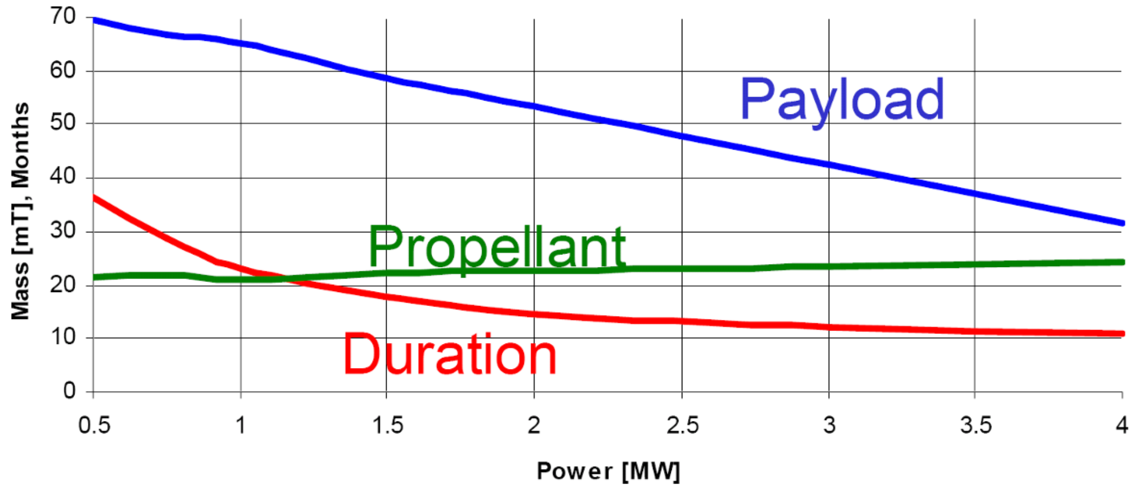


Figure 15: Robotic SEP mission to Mars Power Scan

LEO Power [MW] =	4	2	1	0.5
Initial Mass / Initial Power [kg/kWe] =	25	50	100	200
LEO-ESOI Spiral Time [days] =	80	153	290	537
Heliocentric Transfer Time [days] =	221	234	287	370
Propulsive Capture Time [days] =	31	60	124	201
Total Time from LEO to MGSO [months] =	11	15	23	36
LEO-ESOI propellant [mT] =	14	13	13	12
Heliocentric transfer propellant [mT] =	8	7	7	7
Propulsive Capture propellant [mT] =	2	2	2	2
Argon Propellant Total (propulsive capture) [mT] =	24	22	22	21
17% Tankage [mT] =	4	4	4	4
Power + Propulsion @ 10 kg/kW [mT] =	40	20	10	5
Mass remaining for payload [mT] =	32	54	64	70

Table 10: Mars SEP Power Scan

Figure 16 demonstrates 2 MW SEP Mission to Mars, which takes 15 months of the trip time and can deliver payload of 54 mT. The higher power mission, like a 4 MW Mission, has a duration of 11 months (30% less), but can deliver payload of 32 mT (40% less). The lower power mission, gain in payload but loose in the trip time at much higher rate. For example, 0.5 MW Mission can deliver payload of 70 mT (30% more) but has duration of 36 months (140% more).

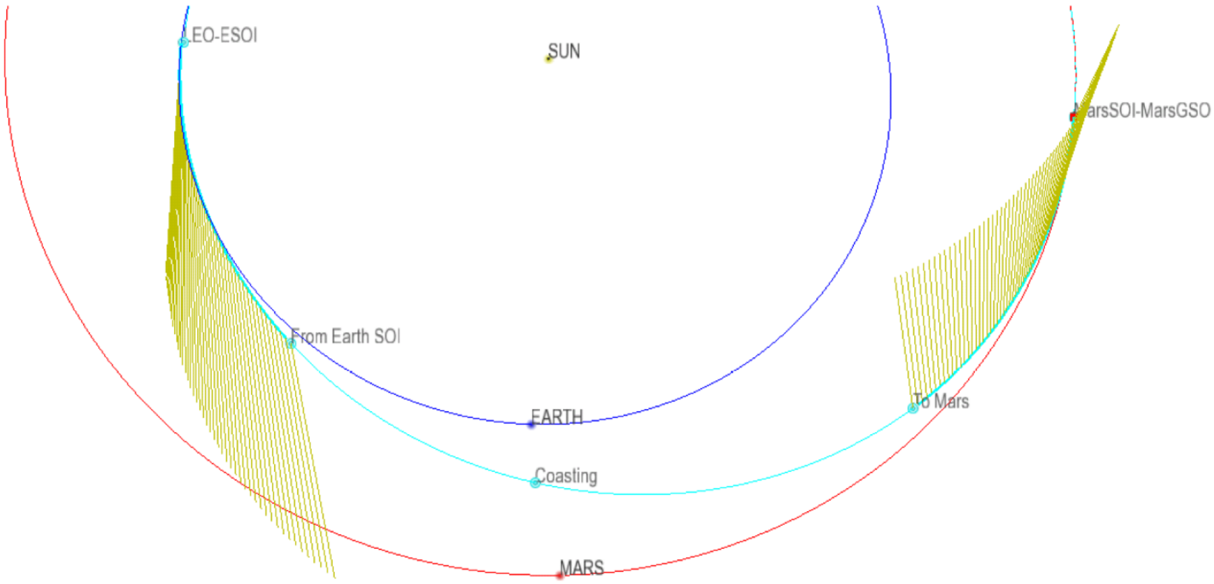


Figure 16: 2 MW SEP Mission to Mars

6.2 Comparison of the SEP robotic mission with Chemical System

In this subsection we will compare the 1 MW SEP robotic mission results with a chemical mission results with the same initial mass, typical specific impulse (I_{sp} = 450 sec for LOX/LH2), and Tank/Propellant ratio of 0.17. For the chemical system, it is assumed that there are separate stages for the TMI (Trans Mars Injection) and MOI (Mars Orbit Insertion) burns. For the TMI burn, there is a significant performance gain from staging. Tankage (17%) and stage mass ratio (equal) here are based on NASA Mars Design Reference Architecture DRA 5.0 2007. [11]

Figure 17 shows an example of the chemical system mission from LEO to Mars, which involves three chemical burns:

1) TMI Stage 1 burn with ΔV_1 = 1.4 km/sec,
2) TMI Stage 2 burn with ΔV_2 = 2.0 km/sec, and
3) MOI burn ΔV_3 = 1.4 km/sec.

All three burns require 63 mT of the propellant and 10 mT of the propellant tank. Thus delivered payload can be 27 mT. The duration of the mission is 8 months. The Table 11 compares the results of the chemical and 1 MW SEP missions. If the duration of 2 years is OK for the cargo mission, than the SEP case is much more attractive due to higher delivered payload.

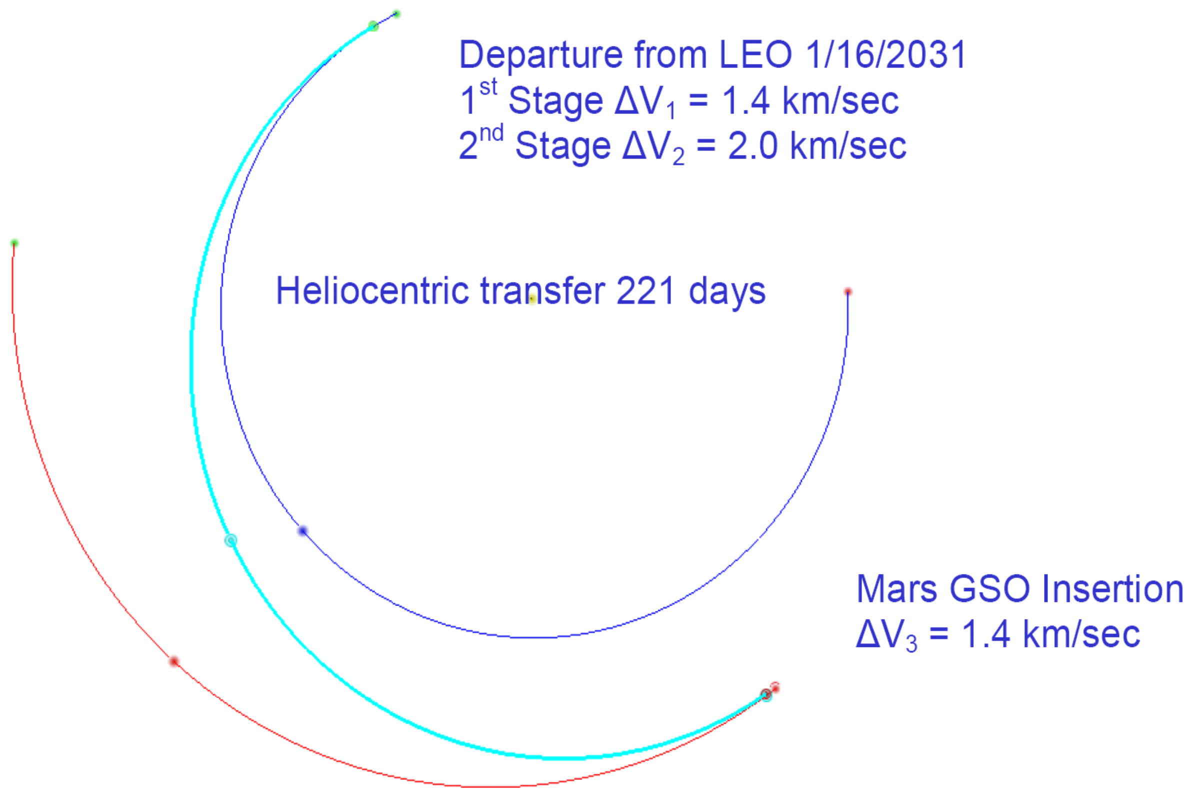


Figure 17: Chemical System Cargo Mission to Mars

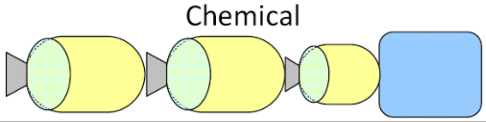
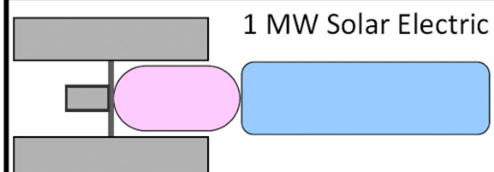
100 mT IMLEO System	Specific Impulse (seconds)	Payload to 1-sol Mars orbit (mT)	Transit Time (months)
Chemical 	450	27	8
1 MW Solar Electric 	5000	64 *2.4 x chem*	23

Table 11: Comparison of Cargo Missions to Mars: 1 MW SEP vs Chemical System

6.3 Variable I_{sp} Earth – Mars Human roundtrip mission

In this subsection we will describe the fastest Earth – Mars roundtrip Heliocentric Transfers for human SEP mission assuming stay time at Mars at least 40 days, payload mass of 30 mT, VASIMR Thruster and power mass of 40 mT, Variable specific impulse range [3,000 – 30,000] sec, net power efficiency 60%, power 2 MW (Specific Mass 20 kg/kW for Power & Propulsion), solar power is capped at 2 MW, when inside the Earth orbit. Return CTV and propellant are delivered to Mars on separate cargo mission and does not need to be taken on the Earth – Mars human transfer. The arrival speed is assumed to be less than 6.8 km/sec which is within requirements for the air breaking landing on the Mars surface.

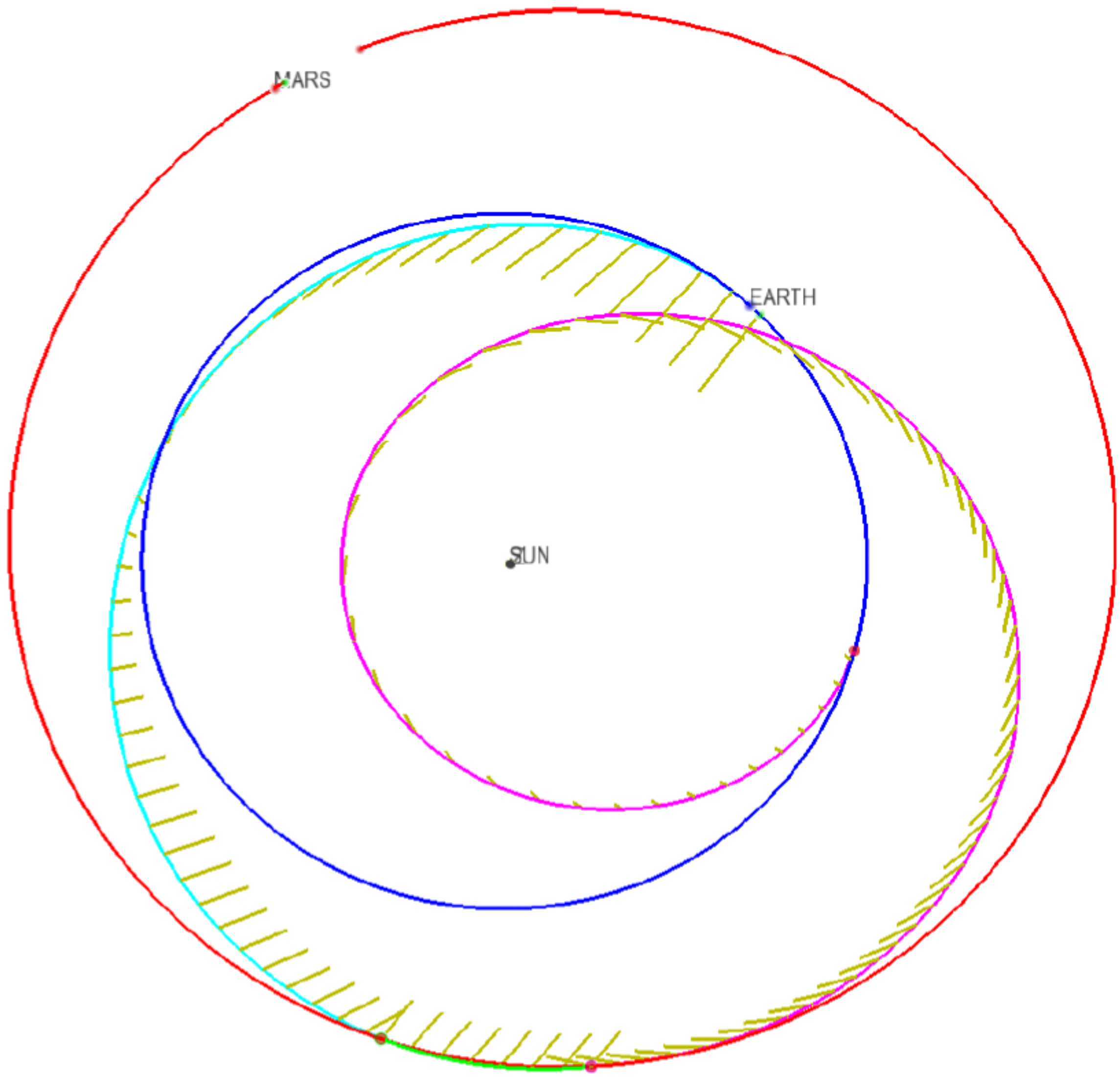


Figure 18: 2 MW SEP Manned Mission to Mars

Figure 18 shows an example of the Variable Isp mission from Earth to Mars optimized for a minimal propellant use. The roundtrip mission has the duration of 22 months, requires 48 mT of propellant and includes the following steps:

1) ESOI Departure: Mar 15, 2035, IM = 96 mT;
2) Earth – Mars coasting takes 235 days and uses 21 mT of propellant
3) At MSOI Arrival, the relative V is 1.6 km/s. The payload of 30 mT gets separated and landed on Mars, OTV goes by Mars
4) Stay on Mars for 40 days, while the OTV rendezvous with MSOI, using 5 mT of propellant
5) MSOI Departure: IM = 92 mT
6) Mars – Earth segment takes 395 days and uses 22 mT of propellant

7) ESOI Arrival V = 6 km/s: Jan 13, 2037

Figure 19 shows specific impulse profile for the mission described above.

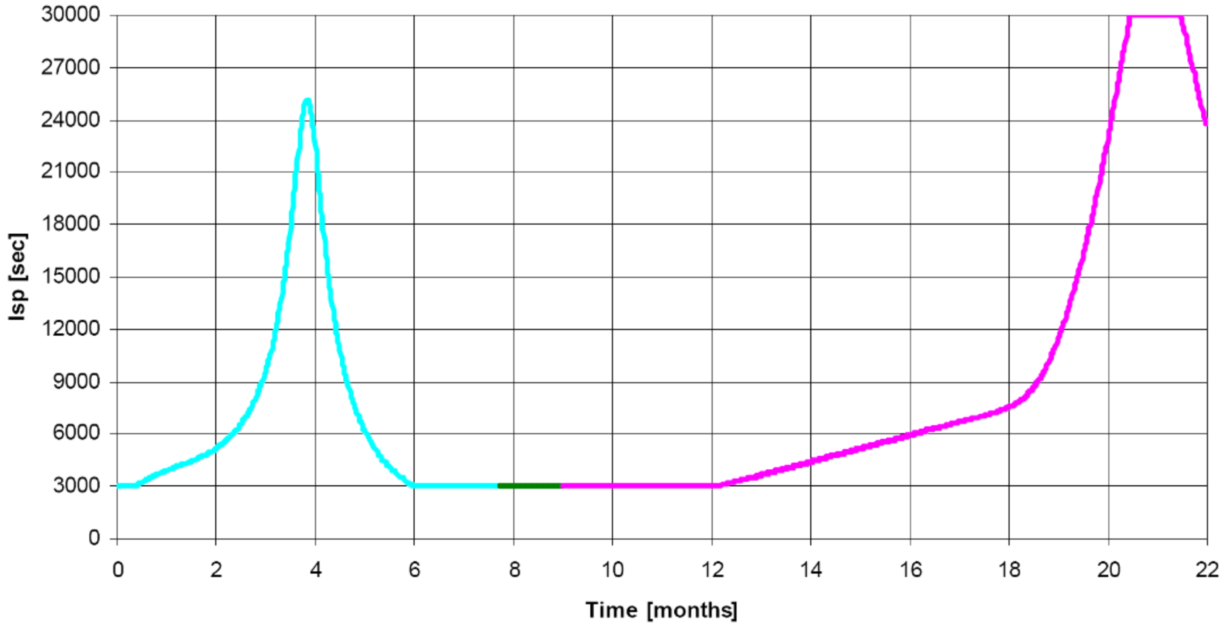


Figure 19: Specific impulse profile for 2 MW SEP Manned Mission to Mars

6.4 Comparison with constant I_{sp} mission and Chemical System mission

In this subsection, the SEP mission is compared with human roundtrip mission based on a chemical system with the following assumption: I_{sp} = 350 sec, chemical thruster mass is 3 mT. The payload mass, departure, stay time and arrival speed assumptions are the same as for the SEP mission.

Figure 20 shows an example of the chemical system mission from Earth to Mars optimized for a minimal propellant use. The roundtrip mission has the duration of 28.4 months, requires 107 mT of propellant and includes the following steps:

1) ESOI Departure: Feb 5, 2029, IM = 85 mT, ΔV=3.2 km/s, Prop = 52 mT
2) Earth – Mars coasting takes 373 days
3) ESOI Arrival V = 6.8 km/s: June 21, 2031
4) Stay on Mars for 184 days
5) MSOI Departure: IM = 88 mT, ΔV = 3.4 km/s, Prop = 55 mT
6) Mars – Earth coasting takes 309 days
7) ESOI Arrival V = 6.8 km/s: June 21, 2031

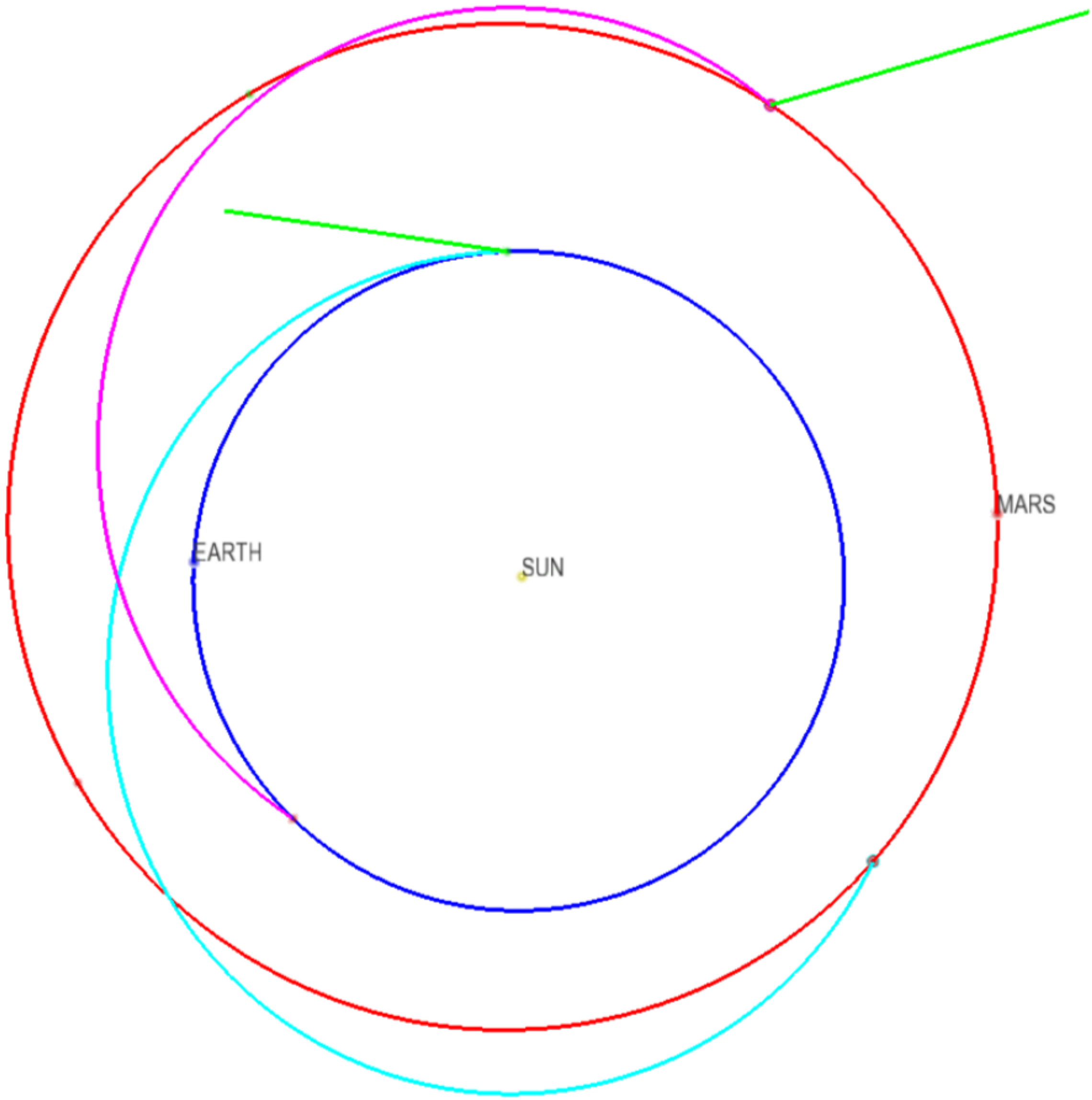


Figure 20: Chemical System Manned Mission to Mars

Isp Type	Variable	2-value	W/Coasting	Constant	W/Coasting	Chem	Chem
Minimal Isp [sec]	3000	3000	3000	3000	2000	350	350
Maximal Isp [sec]	30,000	15,000	Coasting	3000	Coasting	Coasting	Coasting
Roundtrip Time [mo]	22.0	23.3	24.0	23.3	21.8	24.0	28.4
Mass Earth Dep [mT]	96	98	104	118	143	146	85
Propellant To Mars [mT]	26	28	34	48	73	113	52
Mass Mars Dep [mT]	92	119	125	147	127	75	88
Propellant Return [mT]	29	49	55	77	57	42	55
Total Propellant [mT]	48	77	89	125	130	155	107

Table 12: Comparison of Manned Missions to Mars: 2 MW SEP vs Chemical System

Table 12 compares different cases for Human roundtrip Earth-Mars heliocentric transfers:

1) Variable specific impulse in the range of [3,000; 30,000] sec,
2) Two I_{sp} value 3,000 or 15,000 sec,
3) Constant I_{sp} 3,000 sec with a coasting,
4) Constant I_{sp} of 3,000 sec without coasting,
5) 2,000 sec I_{sp} with coasting,
6) Chemical system with restricted duration of 24 months,
7) Chemical system optimized for a propellant use.

One can see that Variable Isp SEP mission is a winner in the mission time as well as in propellant use. Two value I_{sp} missions last about the same but require more propellant. Chemical missions are slower and require even more of propellant.

Section 7. NEP Mission to Jupiter

In this section we describe robotic missions to Jupiter implemented with Nuclear Electric Propulsion technology with a VASIMR® thruster of a few (2 – 15) megawatt of total power. Net Power Efficiency is 60% and Specific Mass is 8 kg/kW for Power and Propulsion. Assume that the missions depart Earth Sphere of Influence so only heliocentric transfer is optimized. We will be looking for missions with Earth SOI – Jupiter SOI Transfer time not more than 2 years. Arrival velocity to the Jupiter SOI is assumed to be zero. The initial mass will be optimized in order to deliver required payload with mass of 50 mT.

7.1 Minimal Propellant mission to Jupiter

The Table 13 demonstrates results of the Power scan for NEP missions to Jupiter optimized for a minimal propellant, assuming mission duration of 24 months. The VASIMR® thruster will be working with Variable Isp in the range of [5,000 – 30,000] sec or with constant Isp of 5,000 sec with optional coasting period.

Power [MW]	2	3	5	10	15
IM (Var I_{sp}) [mT]	101	97	108	150	199
Prop (Var I_{sp}) [mT]	35	23	18	20	29
IM (Const I_{sp}) [mT]	110	108	129	192	249
Prop (Const I_{sp}) [mT]	44	34	39	62	79
P&P Mass [mT]	16	24	40	80	120

Table 13: Power scan for NEP missions to Jupiter optimized for a minimal propellant

The power scan gives us the optimal power value of 3MW for both Variable and constant I_{sp} missions, which correspond to minimal initial mass. Low power missions use more propellant due to the fact that the thruster needs to be working at lower I_{sp} in order to achieve 24 months mission trip time. High power missions use more propellant because the total mass of the spacecraft goes up. One can see that Variable I_{sp} can save significant amount of propellant vs the constant I_{sp} mission. The advantage of Variable I_{sp} becomes more evident for high power missions.

Figure 21 demonstrate optimal 3 MW mission with variable I_{sp}, optimized for a minimal propellant. It has ESOI Departure initial mass of 97 mT and uses 23 mT of the propellant.

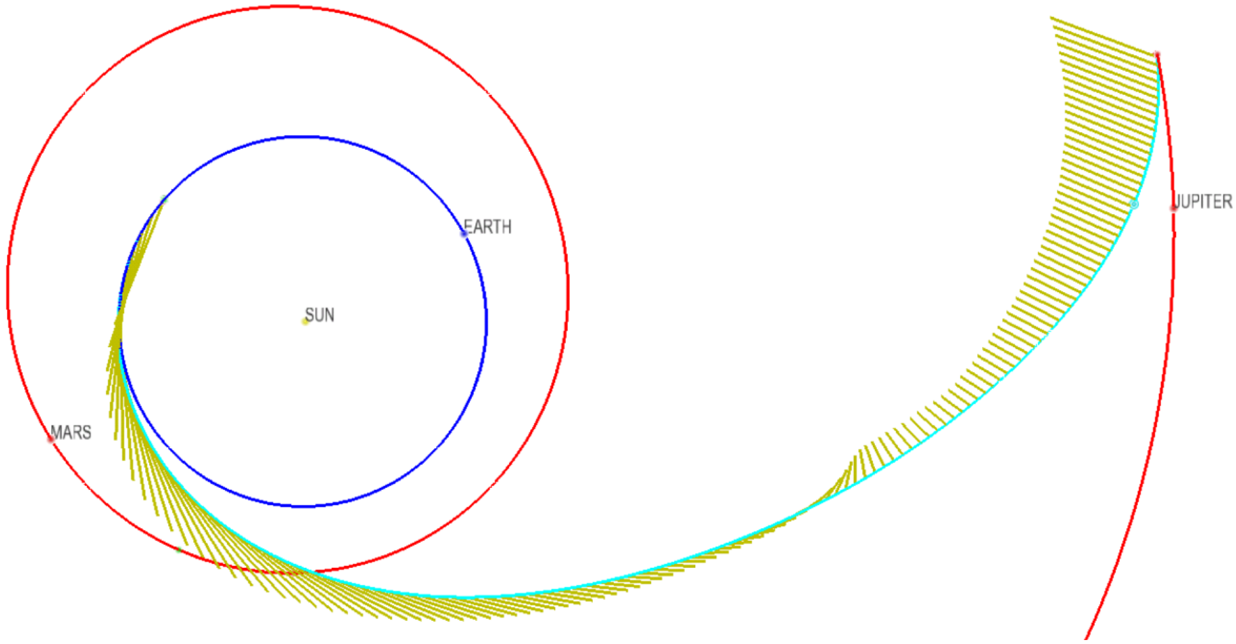


Figure 21: 3 MW NEP Cargo Mission to Jupiter with variable I_{sp}, optimized for minimal propellant

Figure 22 demonstrates the variable I_{sp} profile for the optimal mission above. One can see that it includes long intermediate period corresponding to maximal specific impulse with negligible propellant use.

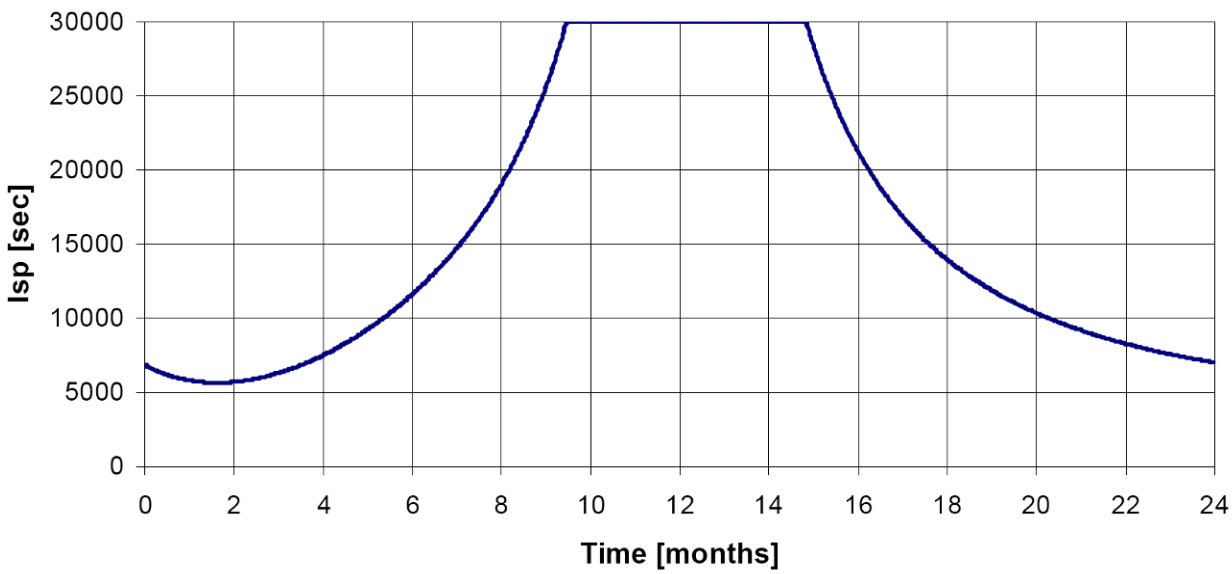


Figure 22: I_{sp} profile for 3 MW NEP Cargo mission to Jupiter

Figure 23 demonstrate optimal 3 MW mission with a constant I_{sp}, optimized for a minimal propellant. It has ESOI Departure initial mass of 108 mT and uses 34 mT of the propellant.

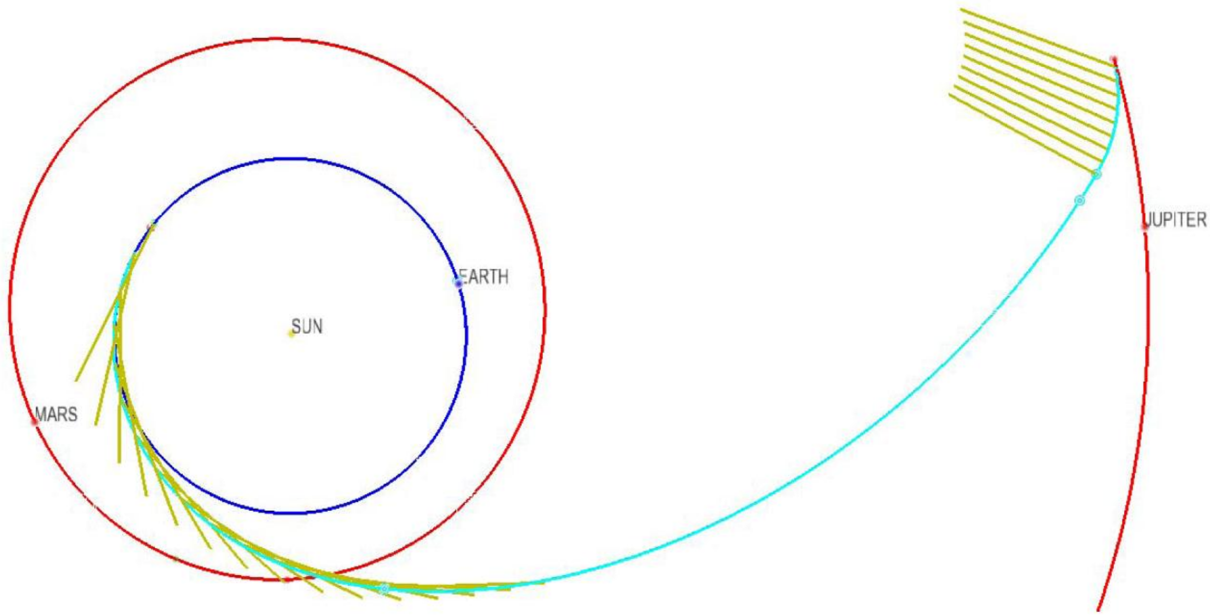


Figure 23: 3 MW NEP Cargo Mission to Jupiter with constant I_{sp}

7.2 Minimal Duration mission to Jupiter

The Table 14 demonstrates results of the Power scan for NEP missions to Jupiter optimized for a minimal duration assuming mission duration of 24 months. The VASIMR® thruster will be working with Variable I_{sp} in the range of [3,000 – 30,000] sec or [5,000 – 30,000] sec. As expected, the fastest missions correspond to the highest power, but they use a lot of propellant.

Power [MW]		2	3	5	10	15	20
[3,000-30,000] sec Variable I_{sp}	T [mo]	21.9	19.3	16.8	15.9	15.5	15.4
	IM [mT]	111	137	203	266	325	375
	Prop [mT]	45	63	113	136	155	165
[5,000-30,000] sec Variable I_{sp}	T [mo]	24.0	21.4	18.6	16.1	15.3	15.4
	IM [mT]	101	111	149	247	333	381
	Prop [mT]	35	37	59	117	163	171
P&P Mass [mT]		16	24	40	80	120	160

Table 14: Power scan for NEP missions to Jupiter optimized for a minimal duration

From the Table 14, one can see that the missions with minimal I_{sp} 3,000 sec are faster than missions with 5,000 sec minimal I_{sp} but they use more propellant. The difference is more evident at low power levels.

Section 8. NEP Interstellar Mission

In this section we describe robotic missions from Earth LEO outside of the solar system to the distance of 1000 AU implemented with Nuclear Electric Propulsion technology with VASIMR® thruster of 1 – 8 megawatts of total power. Net Power Efficiency, η, is 70% and Specific Mass, α, is 8 kg/kW for Power and Propulsion. The initial mass, M_0, will be scanned between 50 mT and 150 mT. The mission analysis is accomplished at different constant values of the specific impulse, I_{sp}, in the range between 5,000 sec and 50,000 sec. The power, initial mass and specific impulse will be optimized in order to deliver required payload with mass of 5 mT at minimal time.

Figure 24 demonstrates the fastest mission to 1000 AU with specific impulse of 20,000 sec, power of 5 MW, initial mass of 150 mT (400 km), calculated with Copernicus code. The mission involves LEO – ESOI transfer, lasted for 11 months and required 5.4 mT of propellant. The whole mission involves thruster operation for 18.2 years, until all propellant is used: 105 mT. After all propellant is used, the dry mass of the vehicle becomes 45 mT (40 mT of Power and thruster and 5 mT of payload). The maximal reached velocity is 224 km/s. The mission lasts 32.6 years in order to get to the distance of 1000 AU.

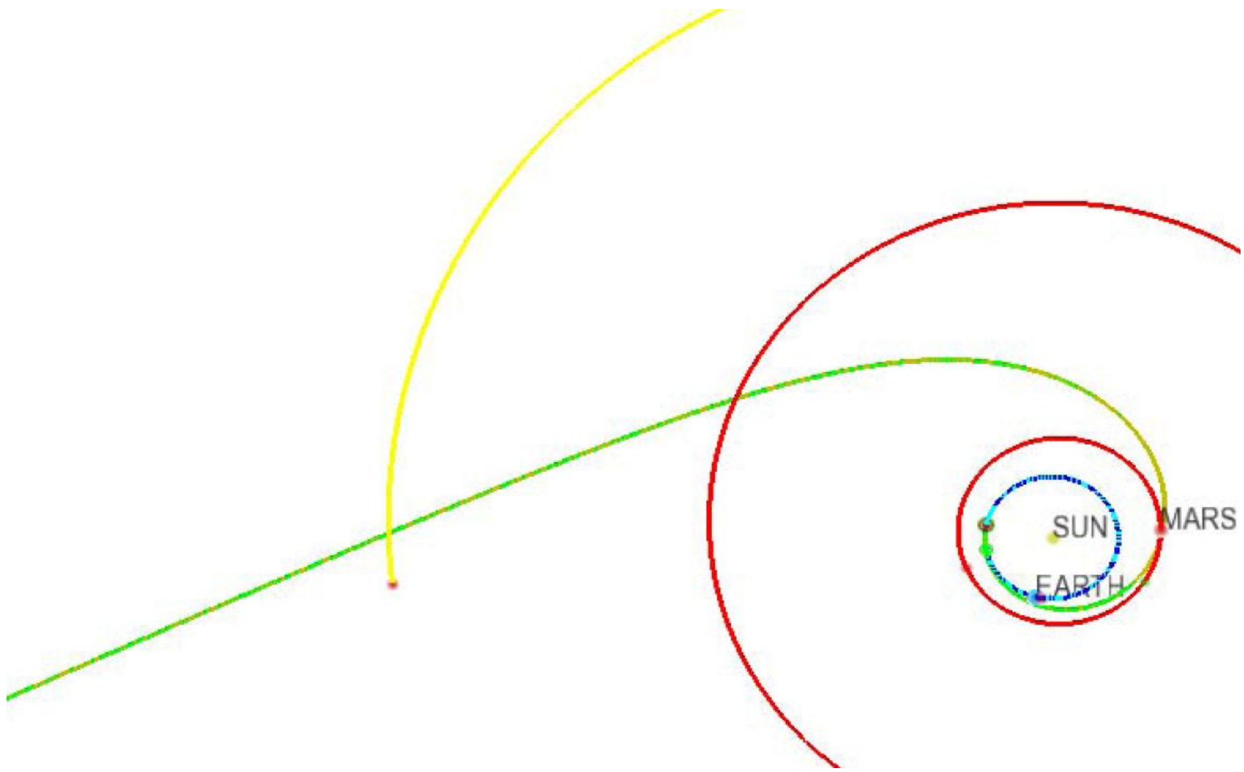


Figure 24: Precursor interstellar NEP mission trajectory

Figure 25 demonstrates results of the Parameter Scan. The upper part of the figure shows the optimal I_{sp} for each value of power and the initial mass. The lower part of the figure shows the mission times for the optimal I_{sp}.

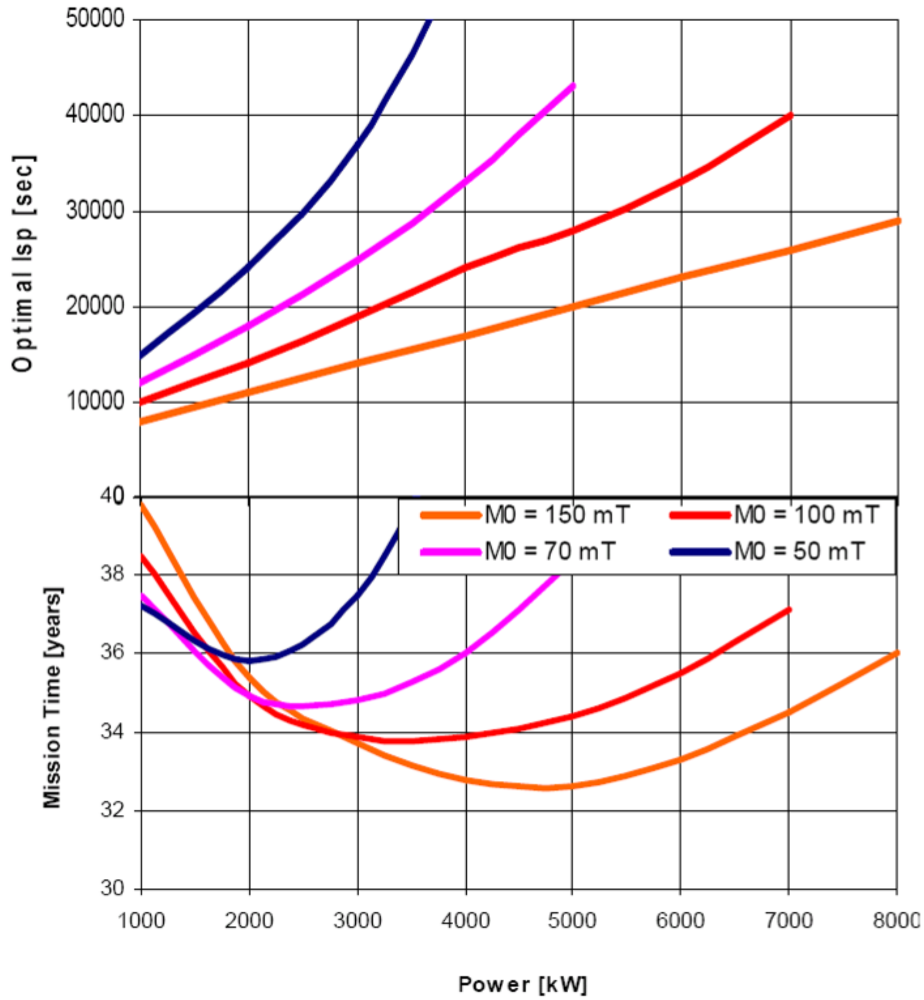


Figure 25: Parametric scan for precursor interstellar mission

Table 15 demonstrates the list of the fastest missions for different initial masses:

IM [mT]	T [years]	P [MW]	I_{sp} [s]
50	35.8	2	24,000
70	34.8	2.5	23,000
100	33.9	3.5	22,000
150	32.6	5	20,000

Table 15: Fastest precursor interstellar missions

Due to the high value of the optimal specific impulse, the use of the Variable Specific Impulse does not make the mission much faster. As the study shows, Variable specific impulse option for the precursor interstellar mission to 1000 AU can be 8% faster than the constant I_{sp} mission for the 50 mT initial mass case.

Let us compare the EP mission with fastest possible chemical system mission, assuming the realistic specific impulse of 450 sec. As previous study demonstrated, one of the optimal way to escape from the solar system is the use of the Oberth Maneuver[12], when the chemical burn is applied at the closest flyby near the sun, assuming R_{min} = 8.5 R_{Sun} (as for Solar Probe mission). [13] The Figure 26 demonstrates such a mission which requires total amount of propellant 200,000 mT (400 x ISS mass). The mission involves two chemical burns: ΔV_1 = 16 km/s at LEO, and ΔV_2 = 30 km/s at 8.5 R_{Sun}. When the vehicle escapes from the solar system it reaches the following velocity $V_{SolarSOI}$ = 108 km/s. The mission lasts 44 years in order to deliver the same payload of 5 mT to the distance of 1000 AU.

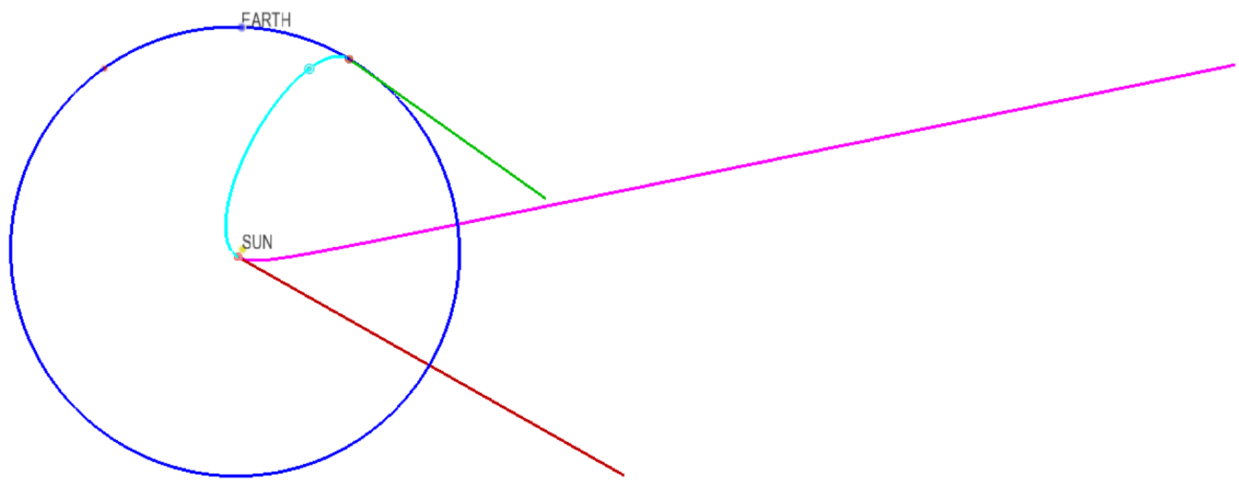


Figure 26: Precursor interstellar mission using chemical burn Oberth maneuvers

Section 9. Lotus – L1 Mission

The Earth – Moon Lagrange point L1 and surrounding stable orbits become attractive intermediate arrival targets for robotic missions to Mars and other planets. In previous research the GTO – L1 mission was analyzed using chemical burn.[14]

In this section we will present the VASIMR mission from GTO to L1, using the following assumptions: IMGTO = 6.6 mT, P = 200 kW, α=17 kg/kW, I_{sp} = 5,000 sec. The Copernicus analysis demonstrated that within described assumptions, and without shadowing effect, the GTO – L1 mission can be achieved with a transfer time of 47 days and consuming Propellant of 410 kg. The Figure 27 demonstrates the mission trajectory in the ecliptic frame and Figure 28 – in the Earth-Moon rotating frame.

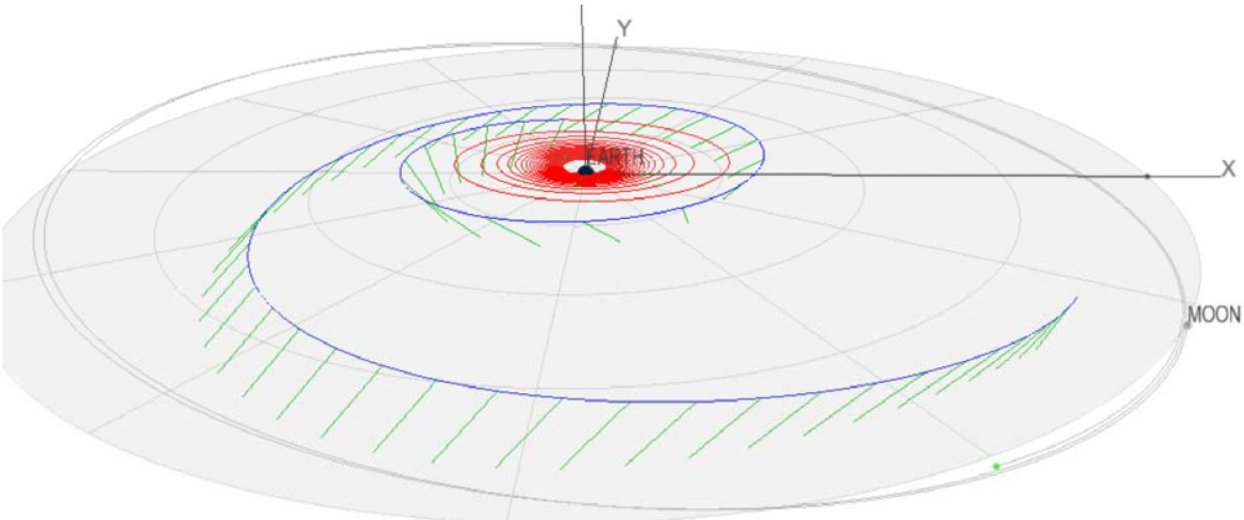


Figure 27: GTO - L1 mission trajectory in the ecliptic frame

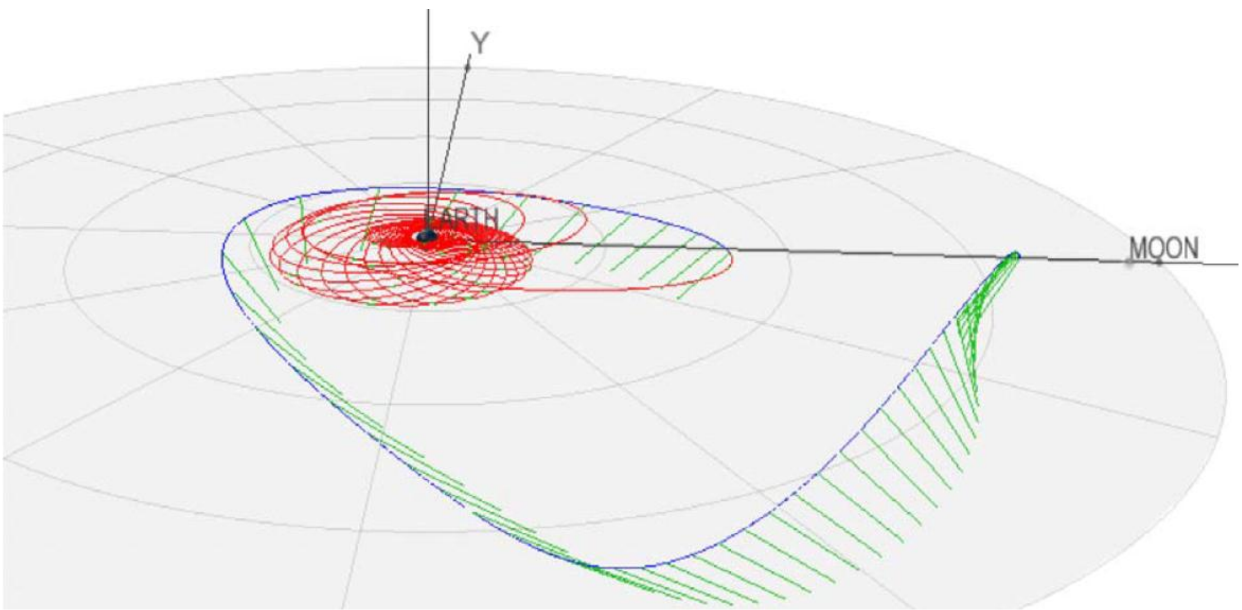


Figure 28: GTO - L1 mission trajectory in the Earth-Moon rotating frame

Section 10. Summary

A survey of space exploration missions enabled by VASIMR® technologies has been performed using a wide range of simulation tools including near-Earth, deep-space robotic and human missions. Using Ad Astra developed software, the optimized near-Earth orbital change missions were generated, and optimal low-thrust control strategies were formulated. This survey provides a map to guide more specific and detailed analysis of scenarios enabled by exploiting and developing VASIMR® technologies. It also provides guidance for evaluating the trade-offs, risks, and rewards inherent in developing new technologies.

A 200 kW OTV based on a VASIMR® driven spacecraft and existing state-of-the-art solar cell technology with a total propulsion alpha of 17 kg/kW, is able to achieve significant mass savings over all chemical thruster technology. For an IM_{GTO} of 6.6 mT, the VASIMR® running on Argon propellant can deliver 2,728 kg of payload to GEO which is 236 % of the payload delivered by a chemical thruster. Similar advantage for the same mission can be demonstrated for 140 kW VASIMR running on Krypton propellant.

Human mission to 2000 SG_{344} asteroid with an 800 kW VASIMR® is able to accomplish the 5 month roundtrip heliocentric transfer with about half the initial mass (IM_{SOI} = 33 mT) of the chemical thruster mission (IM_{SOI} = 59 mT).

The major advantage of Variable Specific Impulse technology is demonstrated for high power interplanetary missions. As demonstrated in this work, the 100 mT Cargo mission to Mars, using 1MW SEP VASIMR technology, can deliver 2.4 times more payload to LMO than a chemical system mission with the same initial mass. The 2-year roundtrip human mission to Mars, can be implemented with 2MW SEP VASIMR technology at initial mass (96 mT) much less than for chemical system mission (146 mT).

For the robotic multi-megawatt NEP VASIMR mission to Jupiter, the power level was optimized in order to deliver 50 mT payload within 2 year transit time. As it was found, for the optimal value of 3 MW of the available power, the mission can be accomplished with minimal initial mass using both constant specific impulse (IM = 108 mT) and variable specific impulse (IM = 97 mT) options.

Also the power level was optimized for multi-megawatt precursor interstellar VASIMR robotic mission. For 150 mT initial mass, the transit to 1000 AU can be implemented within 33 years using optimal power of 5 MW with constant I_{sp} = 20,000 sec. A chemical system mission to 1000 AU, lasting much longer (44 years) would require un-realistically large initial mass of 200,000 mT.

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Appendix 1. Related Copernicus ideck files

[a] GTO-GEO-6.6mT-326sec.ideck – GTO – GSO mission using two chemical burns with 326 sec specific impulse, taken 4,011 kg of propellant, lasted 0.6 days

[b] GTO-GEO-200kW-6.6mT-5000sec-6.ideck – GTO – GSO mission using low thrust separate maneuver strategy transfer, which takes 611 of propellant and lasts 71 days

[c] GTO-GEO-200kW-6.6mT-5000sec-5.ideck – GTO – GSO mission using combined maneuver strategy transfer, which takes 501 kg of propellant and takes 58.3 days

[d] Earth-2000SG344-5000sec-6mT-MinProp.ideck – robotic mission to 2000-SG_{344} asteroid

[e] Earth-2004MN4-5000sec-6mT-MinProp.ideck- robotic mission to 2004-MN_4 asteroid

[f] Earth-2000SG344-5000sec-800kW-35mT-MinDuration-Roundtrip.ideck – 800 kW human mission to 2000-SG_{344} asteroid, optimized for minimal duration

[g] Earth-2004MN4-5000sec-600kW-35mT-MinDuration-Roundtrip-2033.ideck – 600 kW human mission to 2004-MN_4 asteroid, optimized for minimal duration

[h] Earth-Mars-2MW-2033-2000sec-Isp+coasting-SEP.ideck - human mission to Mars lasting 22 months at constant 2,000 sec specific impulse.

[i] Earth-Mars-2031-350sec-Isp-2years.ideck - human mission to Mars lasting 24 months using a chemical system at 350 sec specific impulse.

[j] Earth-Mars-2029-350sec-Isp.ideck – human mission to Mars optimized for a propellant use using a chemical system at 350 sec specific impulse.

[k] ESOI-Jupiter-3MW-5000sec-VarIsp.ideck – 3 MW cargo mission to Jupiter at Variable Specific Impulse

[l] ESOI-Jupiter-3MW-5000sec-coasting.ideck – 3 MW cargo mission to Jupiter at constant I_{sp}

[m] LEO-PrecursorInterstellar-20000sec.ideck – precursor interstellar mission from LEO using NEP at 20,000 sec specific impulse

[n] LEO-PrecursorInterstellar-450sec.ideck – precursor interstellar mission from LEO using two chemical burns at 450 sec specific impulse

[o] GTO-L1-200kW-6.6mT-5000sec.ideck– GTO – L1 mission