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VASIMR® Human Mission to Mars

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Abstract. Variable Specific Impulse Magnetoplasma Rocket (VASIMR®) powered by multi megawatt advanced nuclear reactor power source can deliver human occupied vehicle to Mars in 39 days, as calculated by mission trajectory optimization software. The simulations demonstrated the following desired conditions for the fast mission: using Earth-Moon Lagrange point as a departure state, variable specific impulse regime of the thruster at the maximal constant power, and arriving to Mars sphere of influence with maximal allowed for aero-braking speed.

Keywords: VASIMR; Electric Propulsion; Plasma Radiofrequency Heating; Interplanetary Transportation; Mars; Variable Specific Impulse; Space Vehicles
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INTRODUCTION

For humans to travel safely to Mars and beyond, it will be important to make the trip as quickly as possible and thereby reduce the crew's exposure to weightlessness and space radiation. With today's chemical rockets, a round-trip mission to Mars would take over two years, with much of that time spent waiting for the right planetary alignment to return. A more rapid transit is possible with a Variable Specific Impulse Magnetoplasma Rocket (VASIMR®) propulsion system powered by a nuclear-electric generator. At the 200 megawatt power level, a human mission to Mars can be accomplished with VASIMR® technology in 39 days. The possibility of such a short trip to Mars was lauded by NASA administrator, Charles Bolden (Grossman, 2009), in July 2009. The document represents studies by Ad Astra Rocket Company (AARC) on VASIMR® conceptual human missions to Mars. The effect of initial departure position, initial mass, minimum specific impulse, and mass of payload on the mass-to-power ratio and trip time requirement was analyzed.

We first present the VASIMR® performance parameters assumed for all of the mission analysis. In the following section we show that 2 to 50 MW of input electrical power will allow for heliocentric transfers to a simplified Martian orbit with trip times from 1 to 8 months, which gives guidance for selecting the power levels in the subsequent sections. Next, we demonstrate that variable specific impulse does indeed provide propellant mass savings benefits for energetic missions to Mars. A quick study of the effect of using a solar rather than a nuclear power source shows that solar power increases trip time by about 10% if the specific masses are the same due to the decrease in solar power further from the sun. Next, we show that a 12 MW nuclear power source with a moderately aggressive technology advancement that assumes a specific mass of 4 kg/kW yields 3 month human mission to Mars with approximately the same initial mass as a chemical mission. Finally, we demonstrate that ambitious advancements in nuclear power sources at the 200 MW power level can enable 39 day human missions to Mars.

VASIMR® PERFORMANCE PARAMETER ASSUMPTIONS

Human interplanetary missions with the VASIMR® require power levels in the multi-megawatt for reasonably short transit times.. As we show in this paper, a 12 MW mission can take less than 4 months and a 200 MW mission less than 2 months. The nominal parameters for these missions are variable specific impulse, I_{sp}, from 4,000 to 30,000 s with a total power efficiency, η, of 60%, and a specific mass, α (total), less than 4 kg/kW. Ad Astra Rocket

Company demonstrated 50% net efficiency of the 200 kW VASIMR® lab experiment, VX-200 (Cassady, *et al*, 2010) with a predicted 60% efficiency at high specific impulse. An accurate VASIMR® model, considering the power efficiency to be a function of specific impulse and power, is beyond the scope of these studies.

Ad Astra Rocket Company employs several software for simulation of the variable specific impulse conceptual missions, including Human mission to Mars. They include tools from low to high fidelity, developed inside the company or by different institutions.

The most recent studies were accomplished with Copernicus software (Ocampo, 2002), which is a generalized spacecraft trajectory design and optimization system developed by the University of Texas at Austin. This software has been released to NASA centers and affiliates. It is supplied with a sophisticated GUI (Graphic User Interface), and includes variable I_{sp} capability for electric propulsion (EP) mission. Copernicus is an n-body tool with a high degree of flexibility. The user can model a number of different missions, with varying gravitational bodies, objective functions, optimization variables, constraint options, and levels of fidelity. Additionally, it can model multiple spacecraft, as well as optimize for both constant and variable specific impulse trajectories. Copernicus employs multiple shooting and direct integration methods for targeting and state propagation. Ad Astra Rocket Company has been using Copernicus for various VASIMR® interplanetary mission design applications (Ilin, *et al*, 2010) and to demonstrate the advantage of variable specific impulse over constant specific impulse missions (Dankanic, Vondra and Ilin, 2010).

EFFECT OF INPUT POWER

Here we compare Earth-Mars nuclear powered missions, assuming an initial mass of 100 mT at the Earth's Sphere of Influence (SOI). Both Earth and Mars orbits are assumed circular with their orbit radius equal to their semi-major axis in order to exclude effect of the departure date and give a quick, general result. The propulsion system is assumed to be working continuously with a constant specific impulse of 5,000 sec and a net efficiency of 60%. Figure 1 (a) demonstrates heliocentric transfer trajectories for five power levels: P = 2 MW, 5 MW, 10 MW, 20 MW and 50 MW. For each power level, the direction of the thrust is optimized in order to minimize the transit time. Figure 1 (b) demonstrates the dependence of input power and the corresponding specific mass estimate (α_{power} + $\alpha_{propulsion}$). on mission trip time.

In order to estimate specific mass, the following mass model was assumed:

$$M_{depart} = M_{propellant} + M_{arrival}, \tag{1}$$

$$M_{arrival} = M_{PL} + M_{tank} + M_{struct} + M_{power} + M_{propulsion}. \tag{2}$$

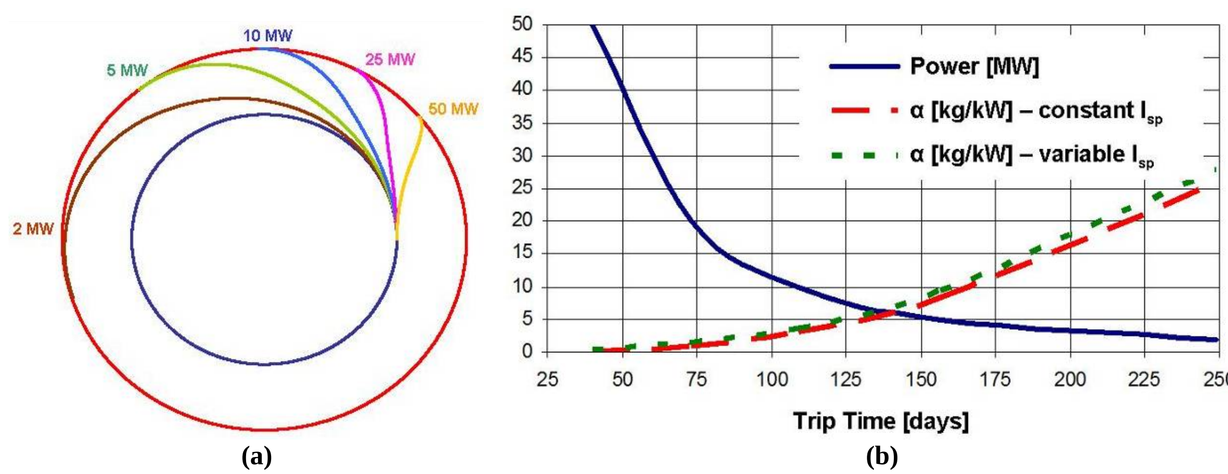


FIGURE 1. Effect of input power on the mission trip time. (a) Constant I_{sp} of 5,000 sec, initial mass of 100 mT and const power efficiency of 60% are assumed for all Nuclear power levels for trajectories shown; (b) Right plot includes specific mass (alpha) for both constant and variable I_{sp} cases.

The payload (PL) mass is assumed to be proportional to the arrival mass:

$$M_{PL} = M_{arrival} * \alpha_{PL} , \tag{3}$$

the propellant tank mass is assumed to be proportional to the propellant mass:

$$M_{tank} = M_{propellant} * \alpha_{PT}, \tag{4}$$

the structure mass is assumed to be proportional to the arrival mass:

$$M_{struct} = M_{arrival} * \alpha_{st}, \tag{5}$$

and both power and propulsion systems have mass proportional to the power:

$$M_{power} = P * \alpha_{power}, \quad M_{propulsion} = P * \alpha_{propulsion}. \tag{6}$$

From the above formulas, the power and propulsion specific mass can be estimated as follows:

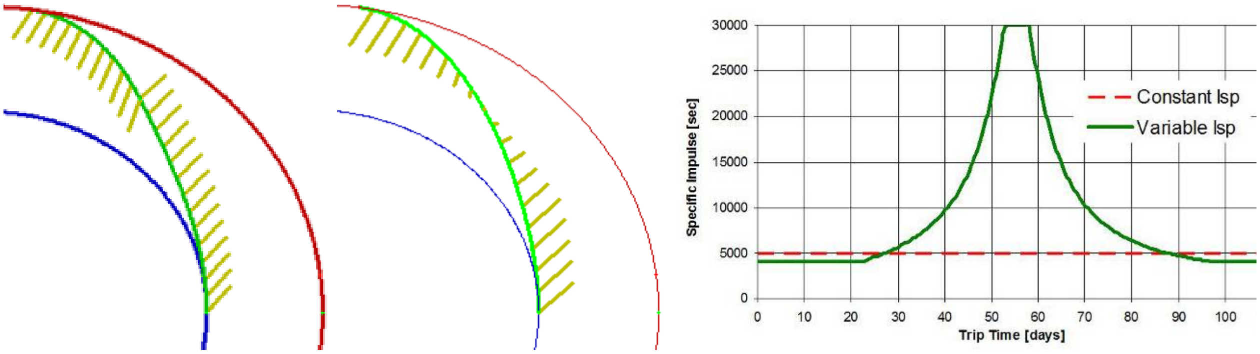
$$\alpha_{power} + \alpha_{propulsion} = (M_{depart} (1 - \alpha_{PL} - \alpha_{st}) - M_{propellant} (1- \alpha_{PL} - \alpha_{st} + \alpha_{PT})) / P. \tag{7}$$

Figure 1 (b) shows the specific mass for propulsion and power assuming $\alpha_{PL} = 0.25$, $\alpha_{PT} = 0.1$, $\alpha_{st} = 0.1$ [kg/kW]. For each power level, missions were optimized for minimum trip time, assuming that VASIMR® thrusters work continuously. The direction of the thrust vector was an optimized variable. With these assumptions, the trip time, T, goes down with the power, P, going up, so $T^2 P$ is almost a conserved quantity. Also, the higher the power, the more propellant is used: $M_{propellant} = \dot{m} T \sim P T \sim P^{1/2}$, where $\dot{m}$ is a propellant mass flow rate. Since the initial mass M_{depart} is fixed at 100 mT, the arrival mass and specific mass decrease with increasing power.

EFFECT OF VARIABLE SPECIFIC IMPULSE

Variable specific impulse is a major technological advantage of the VASIMR® propulsion system. Variable I_{sp} missions can save a significant amount of propellant relative to constant specific impulse missions. A comparison of a constant I_{sp} (at 5,000 sec) and a variable I_{sp} (between 4,000 and 30,000 sec) missions to Mars for the same initial mass of 100 mT, power of 10 MW and trip time of 109 days is shown in Figure 2. The trip time was chosen as a result of Copernicus optimization for the constant I_{sp} mission with minimum propellant utilization. Variable I_{sp} profile was calculated as a result of Copernicus optimization for the same trip time. Variable I_{sp} mission requires 40 mT of propellant versus 47 mT for the constant I_{sp} mission.

Figure 1 (b) includes specific mass for both constant I_{sp} and variable I_{sp} missions. Since variable I_{sp} missions require



(a) (b)

FIGURE 2: Comparison of Constant Specific Impulse (at 5,000 sec) Earth-Mars mission with Variable Specific Impulse (in [4,000; 30,000] sec) mission for 10 MW nuclear power, an initial mass of 100 mT and 109 day mission time. (a) Mission trajectories, with thrust vector, are shown on the left, (b) specific impulse profiles are shown on the right.

less amount of propellant (by 6 – 7 mT) than constant I_{sp} missions, the specific mass for the variable I_{sp} missions can be higher or it can deliver higher payload.

NUCLEAR VERSUS SOLAR POWER SOURCE

Using nuclear power for human mission to Mars has several advantages when compared to solar powered EP mission. It is predicted that for megawatt power levels, nuclear power systems will have smaller specific masses, α, than sub-megawatt designs. Figure 3 demonstrates that even for equal values of the mission parameters for specific mass, power, efficiency, initial mass, and final mass solar powered missions to Mars are about 10% longer than nuclear powered missions.

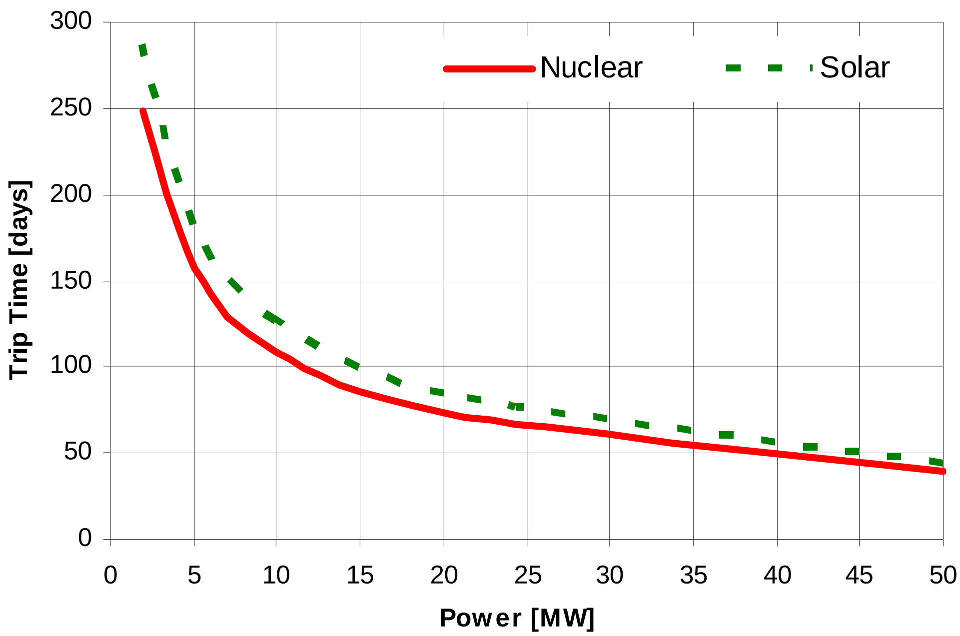


FIGURE 3: Comparing nuclear power Earth-Mars missions with solar power missions. Variable I_{sp} range of [4,000; 30,000] sec, initial mass of 100 mT and const power efficiency of 60% are assumed for all power levels for solar and nuclear cases. The solar power missions are about 10% longer for the same final mass.

12 MW HUMAN MISSION TO MARS

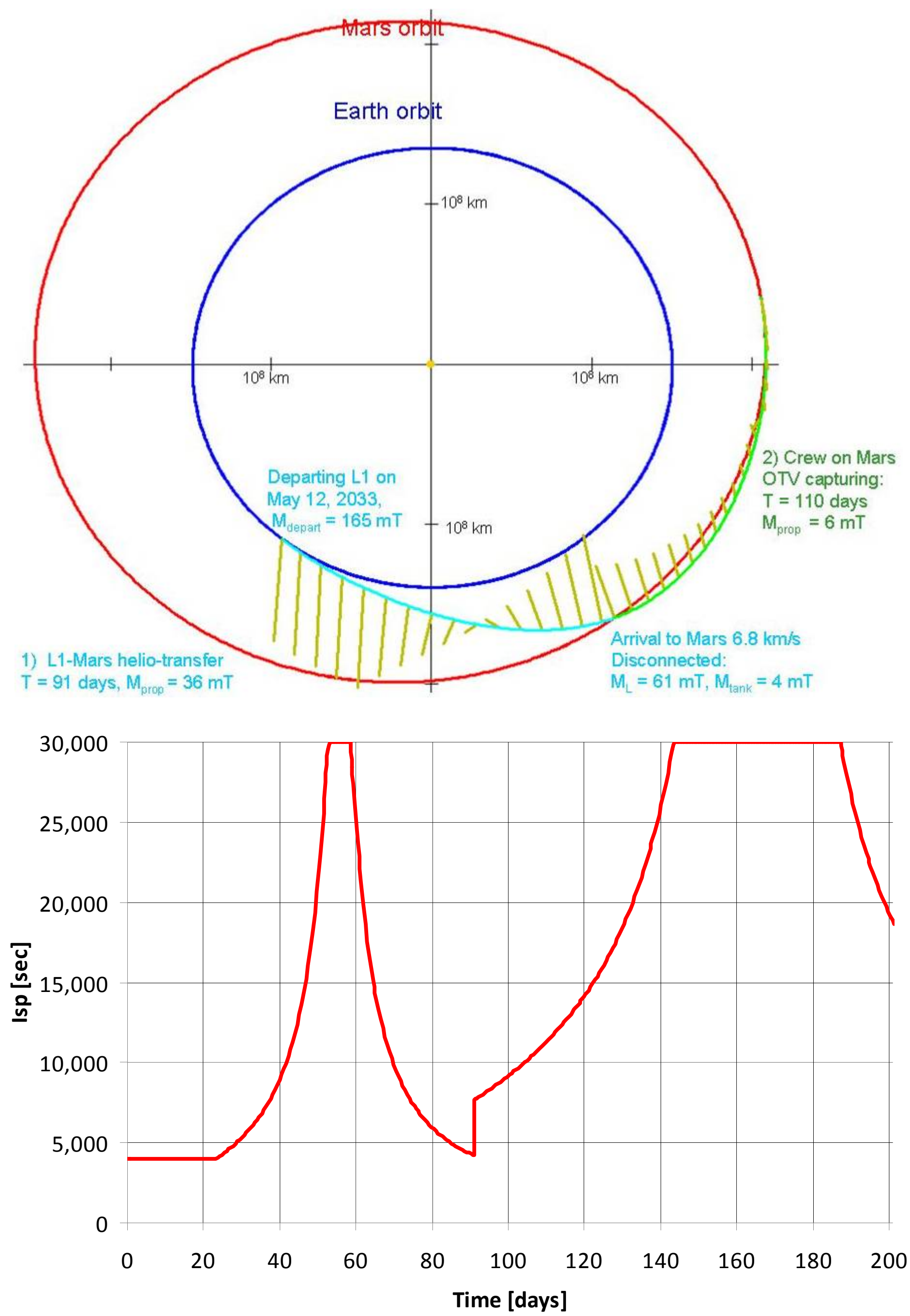
The major advantage of VASIMR® technology - use of variable specific impulse – can be fully appreciated for multi-megawatt interplanetary missions, including human mission to Mars. The first studies of human missions to Mars, based on VASIMR® propulsion technology, were conducted using HOT (Hybrid Optimization Technique) software (Chang Díaz, *et al*., 1995; Chang Díaz, 2000). The HOT software is a Fortran code written at NASA JSC. It uses a numerical optimization method for minimizing a performance function describing mission trajectory with variable specific impulse. HOT provides interplanetary simulation by integrating equations of motion and equations for Lagrange multipliers, numerically using an equation for the control values.

Those studies demonstrated the capability of a 12 MW mission to transit to Mars within 3 months, which is about twice as fast as the DRM (NASA Design Reference Mission) to Mars, assuming chemical propulsion technology [Drake, 1998]. Recent calculations of the mission using Copernicus software confirmed the results generated with the HOT software, but with higher fidelity.

In order to decrease the human transit time of the Earth-Mars mission, the departure point was chosen to be the L1 Lagrangian point. Spiraling from Low Earth Orbit (LEO) to L1 would be conducted without a crew with the

VASIMR® operating in a low-thrust,high-specific impulse mode to minimize the propellant usage.. The ship has a mass of 165 mT at L1 and 12 MW NEP (Nuclear Electric Propulsion system) with an α of 4 kg/kW. The 61 mT Mars Lander (ML) mass breakdown is: 31 mT Habitat, 13.5 mT Aeroshell, and 16.5 mT Descent System. The crew will be delivered to the Mars ship at L1 on a fast, chemically propelled vehicle. Figure 4 shows the mission heliocentric trajectory.

The Earth-Mars heliocentric transfer, which takes 91 days and utilizes 36 mT of propellant, was calculated by Copernicus software with an optimized, variable I_{sp} schedule in the range of 4,000 to 30,000 s, which delivers the specified payload with minimum propellant in the required time. Maximum power is maintained throughout all



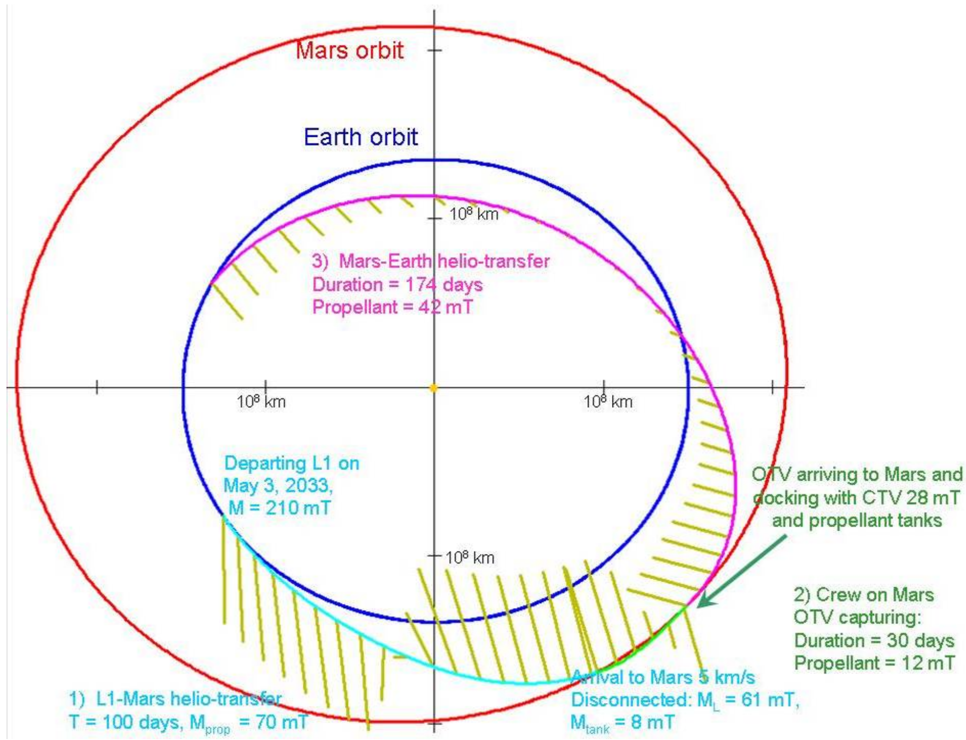
(a)

(b)

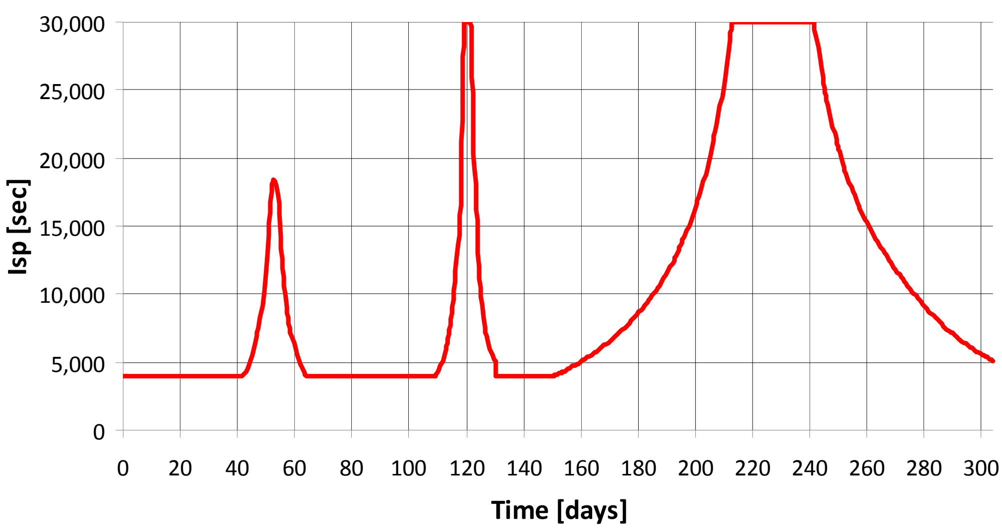
FIGURE 4: (a) Human piloted mission at 12 MW heliocentric transfer from Earth SOI to Mars and (b) I_{sp} profile

phases of flight. At Mars arrival, the relative velocity is 6.8 km/sec, the Mars lander (61 mT) and empty propellant tanks (4 mT) are separated from the orbital transfer vehicle (OTV). The Mars Lander descends directly to the surface, as the orbital transfer vehicle continues past Mars without a crew and rendezvous with Mars after 110 days requiring 6 mT of propellant. The I_{sp} schedule used in the mission is shown in Figure 4 (b).

The separation of the ML from the propulsion system at Mars arrival and its direct entry are operationally reasonable. The lander descent maneuver is identical to that performed in the DRM. The delay in achieving orbital insertion of the propulsion module at Mars results in considerable fuel and time savings. While some risk is involved in this, the crew has a potential backup. The Cargo Vehicle in Low Mars Orbit (LMO) contains the Earth Return Vehicle and the return propellant, as well as a fully functional, albeit lower power, VASIMR® module. This



(a)



(b)

FIGURE 5: Roundtrip human mission to Mars using 12 MW VASIMR®. (a) Helio transfer trajectories are shown on the top. (b) Specific impulse profiles are shown in the bottom.

configuration could be used in a contingency, should the prime propulsion system fail to achieve LMO. Such an option will result in a longer return trip time. Note that the 91 day Earth-Mars transfer was calculated without optimizing round-trip mission, and the return mission was assumed to take place during the following Earth-Mars approach cycle.

The mass budget for the described mission (M_{depart} = 165 mT) includes propellant ($M_{propellant}$ = 42 mT), Mars Lander (M_{PL} = 61 mT), structure (M_{struct} = 9 mT), tanks (M_{PT} = 5 mT), power and propulsion corresponding 4 kg/kW specific mass. The power and propulsion system (48 mT) includes reactors (21 mT), thrusters (21 mT), thruster radiators (2.4 mT) and reactor radiators (3.6 mT).

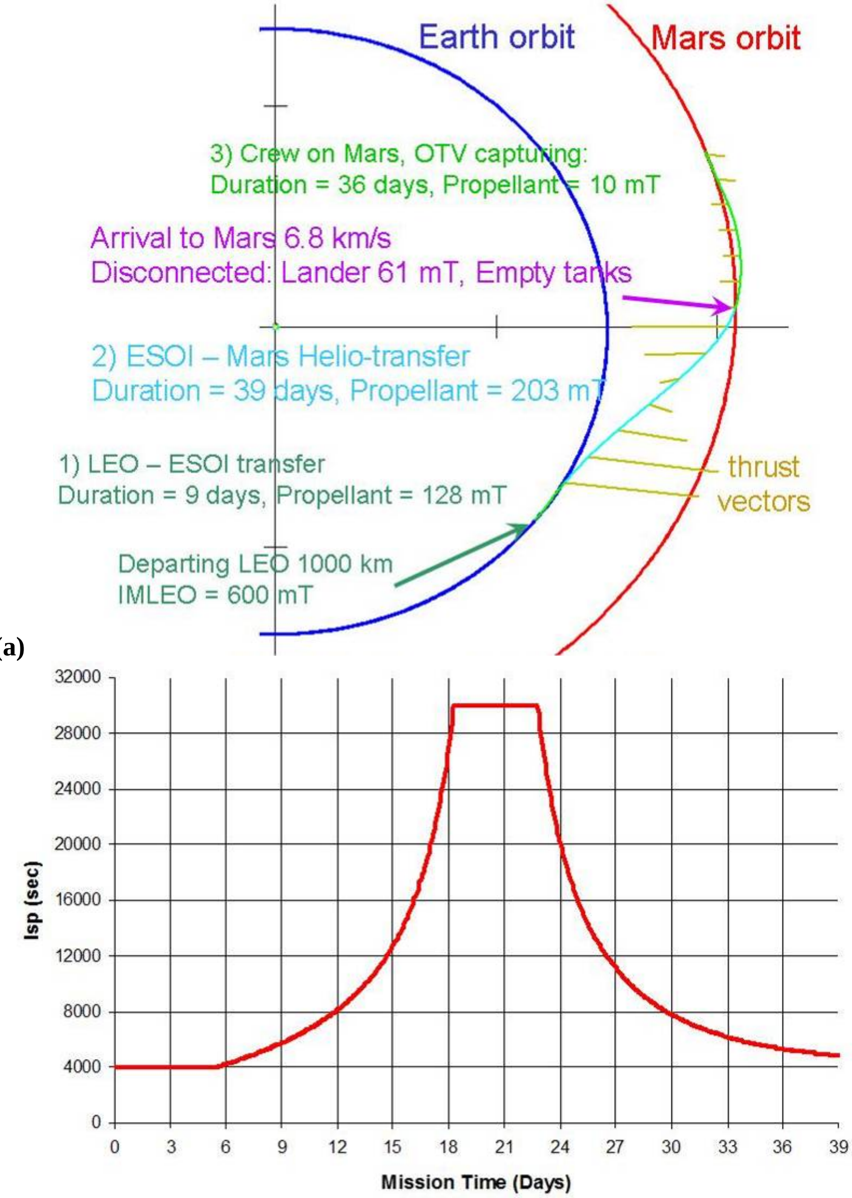
In order to accomplish 12 MW human mission to Mars within one year, assuming the same payload (M_{PL} = 61 mT for the Mars Lander) and specific mass (α = 4 kg/kW), the following modifications need to be made: 1) Initial mass for the space ship should be higher – 210 mT; 2) Trip times are longer than before: 100 days for the Earth-Mars transfer and 174 days for the return; 3) The stay time on Mars should be limited to 1 month. Figure 5 demonstrates Helio-transfer trajectories for the roundtrip mission to Mars departing from L1 on May 3, 2033. The mass budget for the 12 MW roundtrip mission to Mars was assumed as following. Initial mass at Earth-Moon L1 of (M_{depart} = 210 mT) includes propellant mass 82 mT (70 mT for Earth-Mars transfer and 12 mT for OTV capturing), propellant tank mass 10% of propellant, structure, power and propulsion systems of the same mass. Initial mass for the return from Mars to Earth (131 mT) includes propellant mass (42 mT) and payload 28 mT (Crew Transfer Vehicle, CTV). The propellant for the return trip and CTV should be prepositioned on the Mars orbit and delivered on a previous cargo mission.

200 MW HUMAN MISSION TO MARS

The first results of a 200 MW human mission to Mars were generated in 2001 using the OptiMars program (Karavasilis, 2001). OptiMars is a variable I_{sp} Earth – Mars transfer optimizer of low fidelity. The software was developed in 2000 – 2002 at University of Maryland. The code consists of two stages. Both of them simulate 2D trajectories in a plane, which is assumed to include both Earth and Mars orbits. The first stage is a spiraling from LEO to Earth's Sphere of Influence with a constant I_{sp}, assuming Earth is the only source of gravitation. The second stage is heliocentric transfer from Earth orbit to Mars orbit, assuming that Sun is the only source of gravitation. The positions of the planets and departure date are not considered, so the solution trajectory is an "ideal" case for the most optimistic departure date. The heliocentric transfer is simulated using variable specific impulse, and assuming polynomial expression for the radial and tangential accelerations. For a given initial state and heliocentric transfer time, the polynomial coefficients are optimized, and the final mass and arrival speed are calculated. The code is very fast and can be used for quick estimation.

The OptiMars results needed to be verified by a high fidelity tool. The available high fidelity software implemented at AARC is Copernicus. Figure 6 demonstrates Copernicus results for optimized 39-day heliocentric transfer stage of the mission to Mars. The optimal departure date (within the nearest 20 years) from Earth LEO (1000 km) was calculated as June 11, 2035. The spiraling phase with a constant I_{sp} of 4,000 sec from the Earth LEO to SOI takes 9 days and requires 128 mT of the propellant. This mission was designed with the assumption that the crew will be delivered in a fast, chemically propelled vehicle, which will joint the Mars ship near the edge of the Earth SOI, where the ship has achieved the speed of 4.7 km/s. It makes sense to detach used empty tanks before the start of the second phase. The 39 day heliocentric transfer phase requires 203 mT of propellant. Copernicus optimizes the thrust direction and variable specific impulse profile in the range of 4,000 to 30,000 s, in order to minimize the use of the propellant. Note that the 39 day Earth-Mars transfer was calculated without optimizing the round-trip mission.

At Mars arrival, the relative velocity is 6.8 km/sec, the Mars lander (61 mT) and empty propellant tanks (21 mT) are separated from the orbital transfer vehicle. The Mars Lander descends directly to the surface, as the orbital transfer vehicle continues past Mars without a crew and rendezvous with Mars after 36 days requiring 10 mT of propellant. The corresponding optimized variable specific impulse profile is shown in Figure 6 (b). The final mass after the heliocentric transfer is 164 mT, which includes the structure, power, and propulsion systems with a total specific mass less than 0.8 kg/kW.



(a)

(b)

FIGURE 6: Roundtrip human Mission to Mars using 200 MW VASIMR®. (a) Heliocentric transfer trajectories are shown on the top. (b) Specific Impulse profile is shown in the bottom.

The specific mass requirements can be relaxed by increasing the arrival speed, mission time, and also by avoiding spiraling from Earth. Another scenario can be realized by constructing the spacecraft at one of the Earth-Moon Lagrangian points. Figure 7 shows the effect of L1 departure versus LEO departure for different mission times, assuming zero arrival velocity at Mars. The minimum I_{sp} required to achieve the mission and estimated specific mass (for power and propulsion systems) shown in Figure 7. Figure 8 shows the effect of arrival velocity and required trip time on the required specific mass. It was demonstrated, that a 60-day mission with zero arrival velocity or 55-day mission with 10 km/s arrival velocity from L1 to Mars can be achieved with alpha of approximately 2 kg/kW.

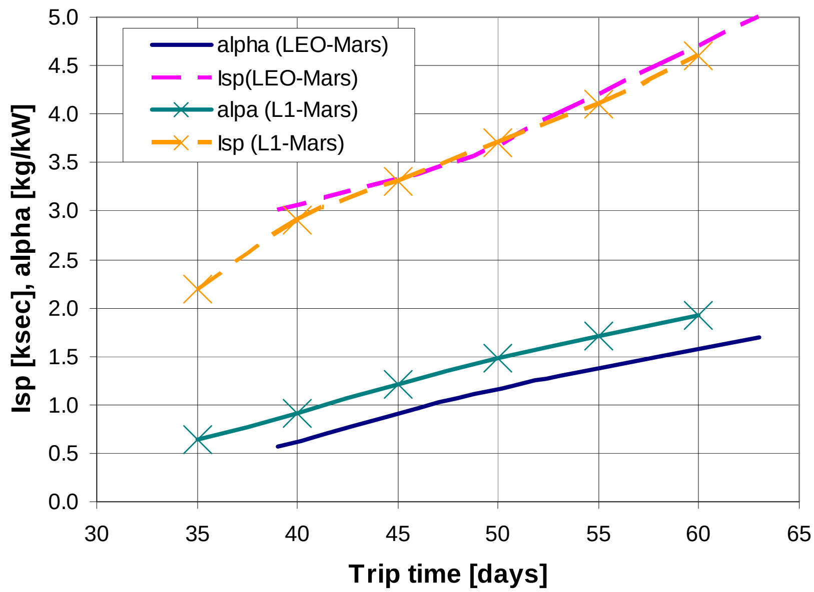


FIGURE 7. Parametric study of human mission to Mars trip time. Observed effect of the departure point, trip time on the required minimal specific impulse and specific mass.

It is not expected that turbo-Brayton reactor systems will fall below 10 kg/kW, even for larger power output systems. However, alterative lower technology readiness level (TRL) options exist - including a Magneto-hydro-dynamic (MHD) system with a projected alpha approaching 1 kg/kW (Litchford, Harada, 2005). Even lower system alpha systems have been proposed using Fission Plasma Core Reactor (FPCR) with MHD power generation (Smith, Knifht and Anghaie, 2002).

Figure 9 demonstrates heliocentric transfer trajectories for the roundtrip mission to Mars departing from L1 on May 30, 2033. The mass budget for the 200 MW roundtrip mission to Mars was assumed as following: initial mass of 600 mT (for both forward and return parts of the mission), propellant mass 300 mT (Earth-Mars) or 330 mT (return), propellant tank mass 10% of propellant, thruster radiator mass 40 mT, reactor radiators mass 60 mT, payload 61 mT (Mars Lander) or 28 mT (Crew return vehicle), structure mass 9 mT, reactor mass 50 mT, thruster mass 50 mT. Total mass for power and propulsion system of 200 mT corresponds to the specific mass of 1 kg/kW. The propellant for the return trip should be prepositioned on the Mars orbit and delivered on a previous cargo mission.

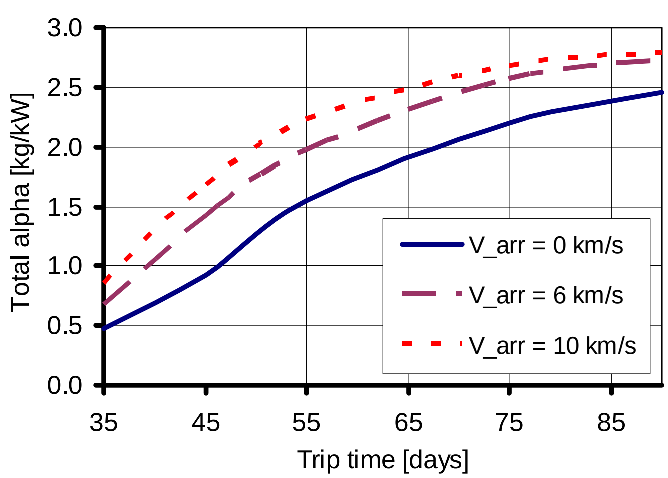
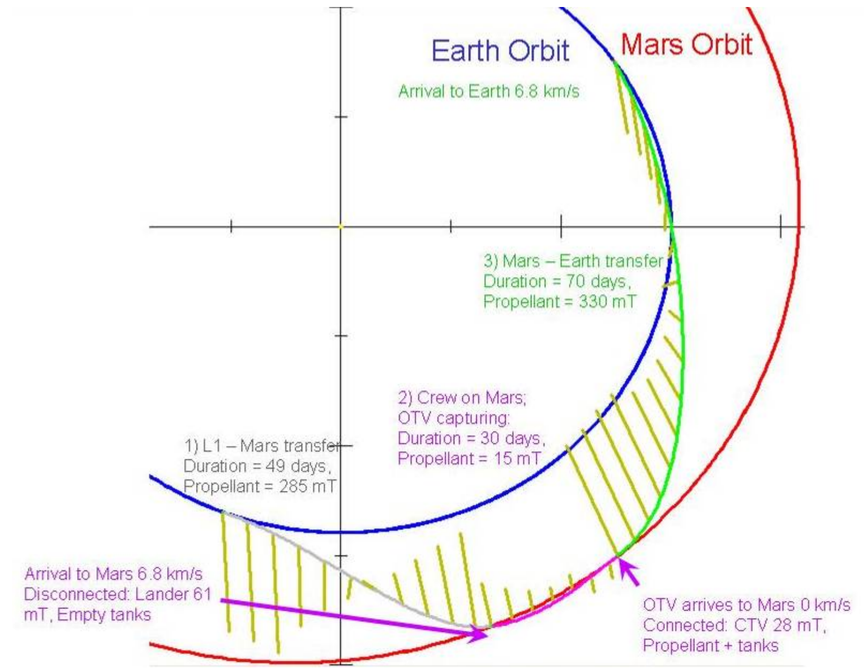
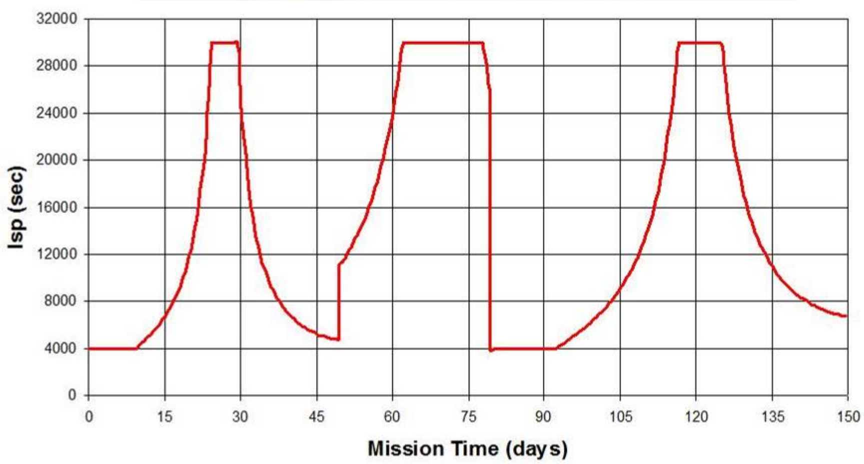


FIGURE 8. Parametric study of human mission to Mars as a function of trip time. Observed effect of the arrival velocity and required mission time on the required specific mass.



(a)



(b)

FIGURE 9. 200 MW Roundtrip Human Mission to Mars. (a) Helio transfer trajectories are shown on the top. (b) Specific Impulse profile is shown in the bottom.

Table 1 demonstrates results of the parametric study of power, mass, trip time and specific mass for roundtrip human mission to Mars. For all cases the same payload mass was assumed: 61 mT (Earth-Mars leg) and 28 mT (return leg). The roundtrip time includes 1 month stay on Mars. The mission assumed departed in 2033 from L1 Lagrangian Earth-Moon point. Initial mass is the same for both legs.

Table 1. Parametric study result for power, mass, trip time and specific mass of the roundtrip human missions to Mars.

Power [MW]=	200	100	50	25	12	6
Total Mass [mT]=	600	500	400	300	250	200
Roundtrip Time [months] =	5	6	6.5	8	10	13
Alpha_thruster+power[kg/kW] =	1.0	1.5	2.0	3.0	4.0	6.0

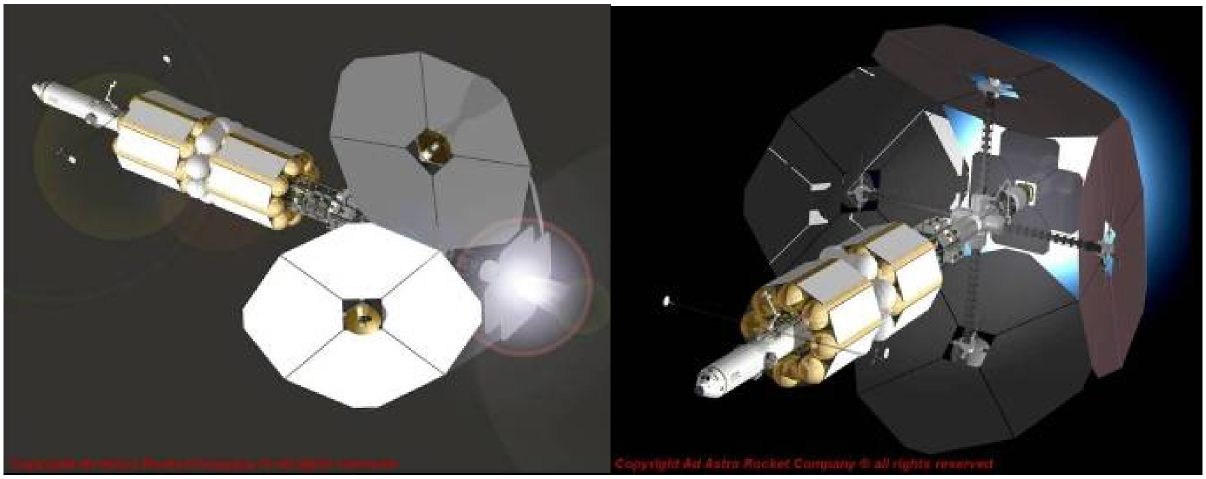


FIGURE 10. Bekuo: Human mission to Mars with four 50 MW VASIMR® thrusters.

SUMMARY

Perhaps the most laudable goals for the VASIMR® technology are that it should eventually enable human missions to Mars that are much faster and safer than can be achieved with chemical rockets. Trips to other near-Earth objects clearly warrant similar studies. The extensive study, presented in this paper, demonstrate how the increase in the available power effects in the decrease of the mission time and increase in the specific mass. Variable specific impulse can allow the savings of significant amounts of propellant over constant specific impulse missions. It was also demonstrated that nuclear power missions reduced the Earth-Mars trip time by 10% relative to solar power missions. Using 12 MW of power and a total specific mass for the entire power and propulsion system of a challenging, but presently realizable 4 kg/kW, allows for a scenario with a crewed one-way mission time of approximately 3 months, and a round-trip mission time of approximately 10 months (including 1 month stay on Mars). Assuming advanced technologies that reduce the total specific mass to less than 2 kg/kW, trip times of less than 60 days will be possible with 200 MW of electrical power. One-way trips to Mars lasting less than 39 days are even conceivable using 200 MW of power if technological advances allow the specific mass to be reduced to near or below 1 kg/kW. Round-trip missions at the same power level and specific mass can last around 5 months (including 1 month stay on Mars).

NOMENCLATURE

α = specific mass (kg/kW)
η = VASIMR® jet efficiency (%)
I_{sp} = Specific Impulse (sec)
M = mass (kg)
$\dot{m}$ = propellant rate (kg/sec)
P = input power (W)
T = trip time (days)

ACRONYMS

AARC - Ad Astra Rocket Company
CTV - Crew Transfer Vehicle
DRM - Design Reference Mission
EP - Electric Propulsion
FPCR - Fission Plasma Core Reactor
GUI - Graphic User Interface
HOT - Hybrid Optimization Technique
L1 - Lagrangian Earth-Moon point
LEO - Low Earth Orbit
LMO - Low Mars Orbit
MHD - Magneto-Hydro-Dynamic
ML - Mars Lander
NEP - Nuclear Electric Propulsion
OTV - Orbital Transfer Vehicle
PL - Payload
SOI - Sphere of Influence
TRL - Technology Readyness Level

VASIMR® - Variable Specific Impulse Magneto-plasma Rocket

VX-200 - VASIMR® experiment at 200 kW input power

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