International Journal of Innovation Scientific Research and Review
Vol. 05, Issue, 08, pp.5037-5043, August 2023
Available online at http://www.journalijisr.com
SJIF Impact Factor 2023: 6.599

Research Article

ISSN: 2582-6131

MMX (MARS MET X) A FAST HUMAN MISSION TO MARS ARCHITECTURE BASED ON NEW PROPULSION AND SPACE POWER TECHNOLOGIES: A PRELIMINARY STUDY

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Received 28th June 2023; Accepted 29th July 2023; Published online 30th August 2023

ABSTRACT

MMX (Mars MET X) is a Human Mars mission architecture capable of rapid development and rapid space transit to Mars. The results of a preliminary analysis are shown here. MMX is based on revolutionary developments in space launch, propulsion, and well established solar power technologies. This article concentrates on the core mission element of the Earth-to-Mars-surface mission and assumes prepositioning of supplies and habitats on Mars and-or in Mars orbit. A three-person crew is assumed to minimize mass. The SLS (Space Launch System) is used to put the basic CMS (Crewed Mars Spacecraft) into Earth orbit and boost it into an escape trajectory from Earth. Immediately the CMS docks with a SEP (Solar Electric Propulsion) spacecraft with 3MW of Solar Power and carrying a cluster of 75kW MET (Microwave Electro-Thermal) plasma thrusters and tanks of water propellent. The MET generates plasma thrust at 9km/sec Isp and takes advantage of the density and storability of water in space. In full Sunlight the combined CMS-SEP then performs a 14 day burn to achieve TMI (Trans Mas Injection) onto a fast trajectory to Mars , arriving in approximately 90 days. Near Mars the CMS detaches from the SEP and deploys a large IHS (Inflatable Heat Shield) in order to achieve aerocapture and aerobraking into Mars decent, to a pre-prepared Mars base. Return from the Mars surface is assumed to be achieved by a prepositioned MAV(Mars Ascent Vehicle) to rendezvous with a SEP MRC (Mars return Craft) to Earth.

Keywords: Mars, Earth, Solar panel, Delta V, Earth orbit, Aerocapture, Space Launch, MMX mission.

INTRODUCTION:

NEW TECHNOLOGIES MAKE A RAPIDLY DEVELOPED AND RAPIDLY TRANSITING HUMAN MARS MISSION POSSIBLE.

Because of recently demonstrated space technologies, a fast tracked, fast transit human Mars mission, called MMX (Mars MET eXpress) is now possible. The SLS (Space Launch System) has been demonstrated to perform flawlessly as did the Orion Capsule [1] (see Figure 1) , allowing a rapid American return to the Moon.

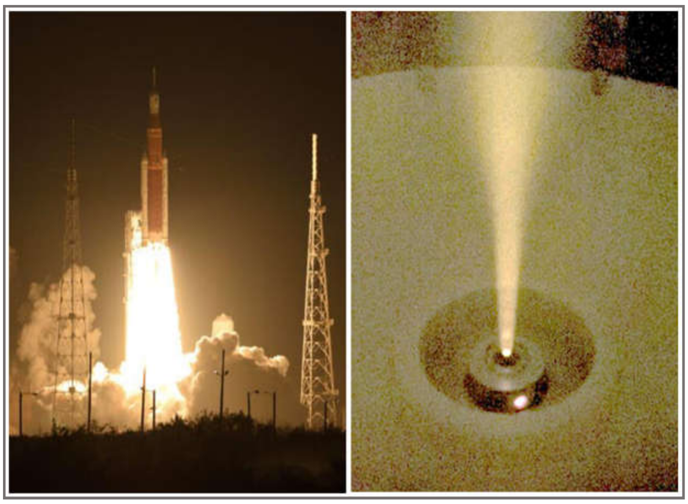

Figure 1 (L) The SLS launch as part of the Artemis 1 mission. (R) the MET plasma thruster in operation.

However, also of deep significance has been the successful demonstration in space of the MET (Microwave Electro-Thermal) plasma thruster using water propellant [2], as seen in Figure 1 and 2. This plasma thruster is the invention of the author. [3] The efficiency and 9km/sec exhaust velocity, of the thruster, combined with the excellent space storage and transfer characteristics of water, now enable a SLS-with Solar Powered MET upper stage to become the core system of a near-term human Mars mission. This system can send a CMS (Crewed Mars Spacecraft) to Mars on a fast trajectory minimizing space radiation exposure for the crew. The problems of a high-speed approach to Mars that results from this fast trajectory are then mitigated by another recent space technology development, demonstration of an IHS (Inflatable Heat Shield). [4] (Figure3.) This greatly reduces the need for propellant to be carried from Earth for the Mars capture and landing phase of the mission. Therefore, it will seen, that these new advanced technologies enable a simple MMX mission architecture capable of rapid development and rapid transit to Mars. Here, is this brief article, the results of a preliminary study are presented, using the approximations of short impulse burns and circular orbits. Gravity effects of the planets are neglected except for escape velocity considerations. Numerical values should therefore be considered approximate and preliminary, with more sophisticated analysis to be presented later. However, even with these caveats, a basic rapid development plan-rapid transit to Mars program appears eminently feasible.

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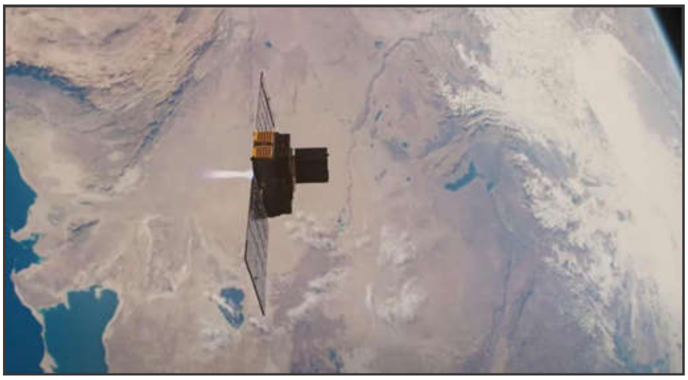

Figure 2. The Vigoride 5 vehicle propelled by the MET thruster using water propellant.

Figure 3 the Inflatable Aerobrake Shield

The revolutionary new heat shield technology will allow a landing component of the CMS to aerobrake on the Mars atmosphere and land, without the need of rocket propellants at Mars. Aerobraking at Mars has been successfully demonstrated by the Mars Reconnaissance Orbiter. [5] This discussion of the mission architecture assumes the prepositioning of habitats and supplies, including a MAV (Mars Ascent Vehicle) for a later rendezvous with a prepositioned SEP MRC (Mars Return Craft) for a crew return to Earth. In the body of this article we will discuss the core system, concentrating on propulsion and trajectories.

GETTING TO MARS FAST :BOTH PROGRAMMATICALLY AND IN TRANSIT TIME

A rapid program to place humans on Mars, can make use of present technologies, saving much development time. Concurrent efforts to develop more advanced technologies such as Nuclear Thermal Propulsion can also proceed to make the follow -on missions to Mars even more rapid and easy. However, present technologies can create a Fast Track human Mars mission in the near term. We can review these present technologies in detail as well as their role in the MMX.

The SLS block 1B cargo (see Figure 4) can put 42 tonnes (Metric Tons) on a trajectory to the Moon, essentially an Earth escape trajectory. We will assume for these discussions a CMS based around the Orion Capsule mass ((10,400 kg) with an additional ASM (Augmented Support Module) mass ((16,600 kg), for a total mass of 27 metric tons. This ASM will include 4 tonnes (metric tons) of water as radiation protection, emergency life support (water has many uses including being a source of oxygen) and emergency MET propellant. The crew size is three. The Orion is currently under development for crewed spaceflight and can serve as a backup spacecraft to bring astronauts safely back to Earth in case of a Mars mission abort.

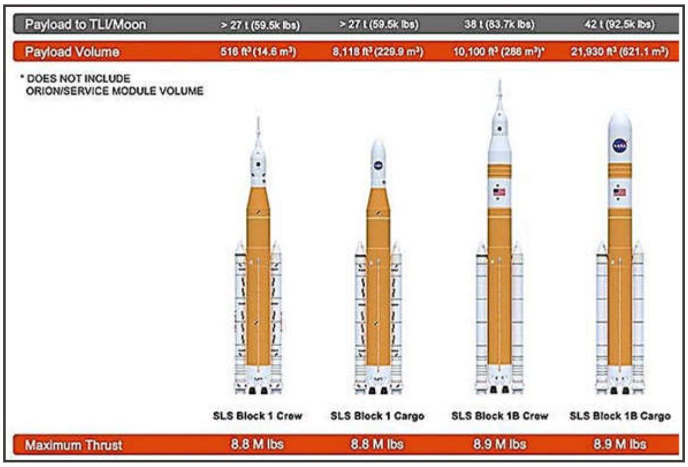

Figure 4 The demonstrated Space Launch System and its projected evolution.

A rapid transit mission to Mars, desirable to reduce crew exposure to deep space radiation, needs non-chemical propulsion. This is provided by the demonstrated MET plasma thruster using water vapor propellant. The high density, equation of state, and ability to store and transfer water in space make it an ideal propellant for moving heavy payloads to Mars. Water is also deeply compatible with human beings and is vital to sustain life. For a human deep space missions it also makes excellent radiation shielding. The concept of using the MET thruster with water propellant for a Mars mission has been explored previously [6] for longer transit time missions (see Figure 5, 6 and 7).

Another important technology for the MMX is the well demonstrated deployment and operation of MW class Solar Power arrays on the ISS, which generate 215kW of electric power in full Sunlight [7] (see Figure 8.) Thus, plenty of power can be made available to power MET-water propulsion. Accordingly, the components of the SLS, the MET thruster with water propellant , and Megawatt class Solar Power Arrays to power an outgoing Mars mission are all demonstrated. Such large, Megawatt class, Solar arrays have long been contemplated for space based solar power sources for Earth and have motivated research to increase efficiency and kW/kg specific power, and long-term performance in the space radiation environment. It is believed that thin film Solar arrays can achieve 1kW/kg. [8] While enhanced performance with radiation resistance has been a long sought goal for such arrays [9], it is not a mission requirement for the Solar Arrays for the MMX mission, which requires only that performance at high power and low mass in the space environment be sustained for a year or less (see Figure 9.) Nominally, the MMX Solar arrays only need to power the MET propulsion units for a few months.

The MET plasma thruster utilizes a vortex-stabilized, electrode less, microwave-gas discharge to heat gases for propulsion. Because the plasma is largely isolated in the resonant microwave chamber it has high thermal efficiency, with little microwave energy lost to heat in the chamber walls. Measured thermal efficiencies were approximately 50% for the 1kW version that operated on 2.5GHz microwaves. [3]. This thermal efficiency was nearly the same for different molecular gases, including gases as different as Helium and Nitrogen, indicating that the primary heat loss is radiative from the hot central plasma and can be sharply reduced by gold plating the walls for the resonant chamber. 915MHz microwaves are being used, which we will assume, when systems are optimized, can create microwaves at 98%

efficiency from DC. Therefore, we will assume a high thermodynamic efficiency of 93% for an optimized 75kW unit. Since the skin depth for 915 MHz microwaves is very short in conductors such as gold and aluminum, 2.7 μm, we will assume the 75kW MET units can be reduced in mass to approximately 5kg each and that 40 MET engines will be clustered to provide thrust at 3MW of solar power in full sunlight. We will thus assume 200kg for the MET propulsion array.

The thrust, T, per unit power, P, of an electric thruster is found as where ε=90% is the energy optimized system efficiency and V_{ex}=9km/sec is the exhaust velocity :

$$\frac{T}{P} = 2\varepsilon/V_{ex} \tag{1}$$

Thus, the T/P =0.2 N/kW for our optimized MET thruster and for a cluster of 40 engines consuming 3 MW of solar power we have a total thrust level, T_t, of 600N. The mass flow rate required for water propellant into the thrusters to produce this thrust is found :

$$\frac{dM}{dt} = T_t/V_{ex} \tag{2}$$

The total mass flow rate is thus dM/dt = 0.0667 kg/sec. Therefore the SEP system under full Sunlight near Earth will consume 5.79 tonnesper day. The mass density of Solar Power arrays is assumed to be 1kW/kg and thus is 3 tonnes for an approximate SEP propulsion system mass of 3.2 tonnes, neglecting water propellant mass.

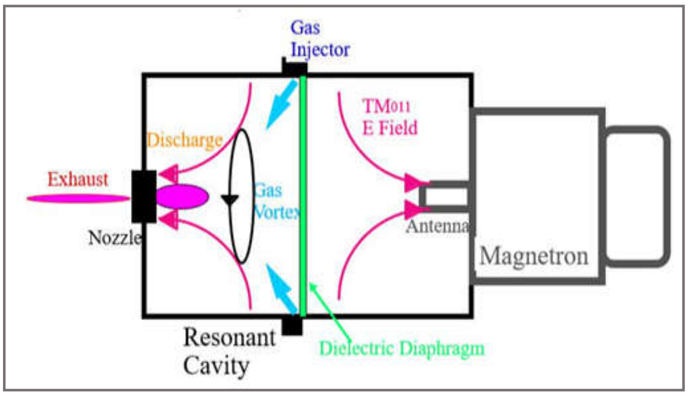

Figure 5A schematic of the MET plasma thruster. The thruster can be driven directly by a Magnetron microwave source.

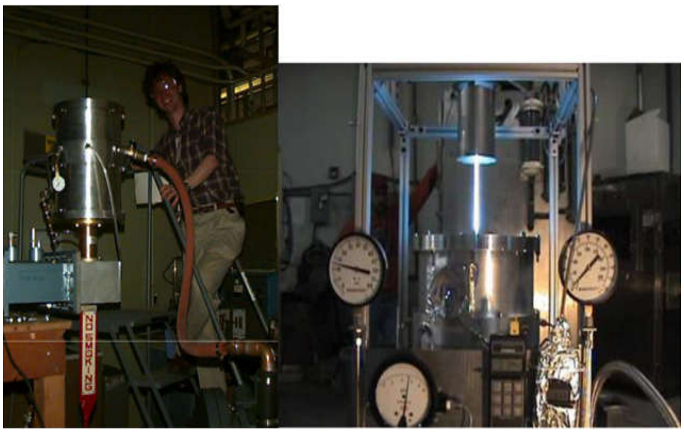

Figure 6 The 75kW MET thruster using water vapor propellant. The 75kW MET operates on 915MHz Microwaves that can generated with 95% efficiency.

We will now proceed to calculate the time of flight and propellant mass required given these levels of thrust and exhaust velocity achievable with a near term-Mars mission.

Let us assume we wish to achieve an Earth to Mars transit time of approximately 3 months, ¼ of a year. This is much less than the standard, minimum energy, Hohmann transfer transit time of approximately 8.5 months and thus requires a much greater Delta V at departure from Earth than the minimal, approximately 3.5km/sec above escape velocity used for the Hohmann.

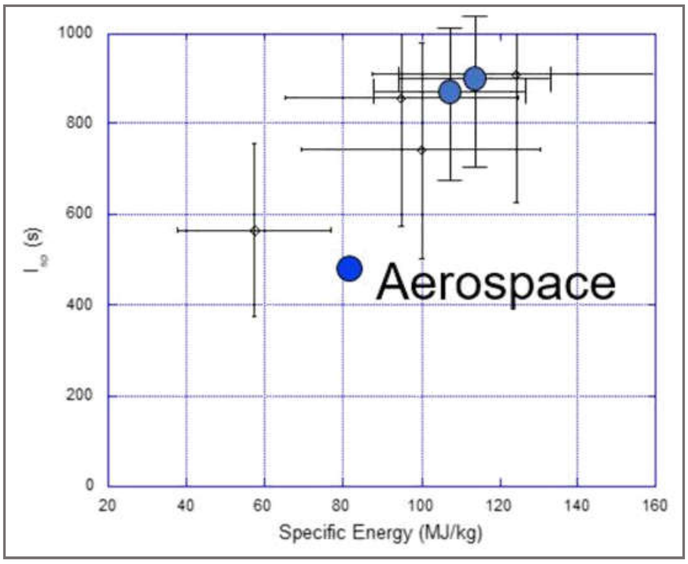

Figure 7 Measured Specific impulse for the MET using water vapor propellant.

Figure 8 The 215 kW Solar Power arrays of the ISS.

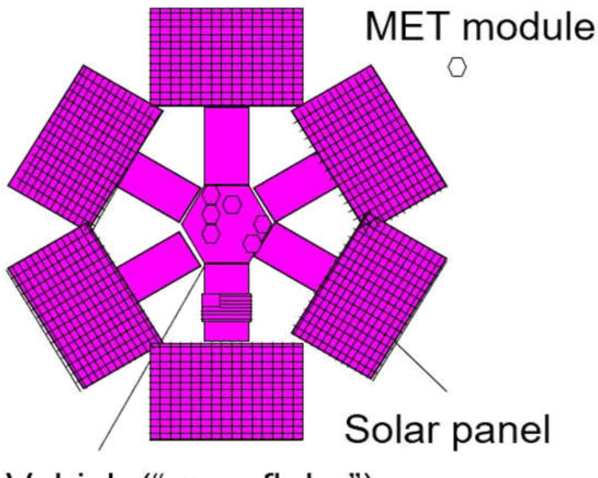

Figure 9 MET SEP Mars Vehicle before full Solar array deployment.

APPROXIMATE ESTIMATES OF MASSES AND TIMESCALES FOR THE MTI (MARS TRANSFER INJECTON) PHASE OF THE MMX MISSION.

For the purposes of gaining estimates of the masses, flight times, Delta Vs, and closure speeds at Mars on arrival, we will make the following approximations: first, we will assume perfectly circular orbits for Earth and Mars , second, since the SLS and MET are high thrust propulsion technologies, we will make the approximation that the Delta V events occur over very short times relative to the mission transit time between Earth and Mars. All major Delta V events will be assumed to occur near Earth, because of the storability problems of liquid hydrogen in the SLS in Earth orbit and the intense sunlight near Earth to enable maximum power for the SEP-MET propulsion system, and thus maximum thrust.

We will assume an SLS puts a MCS (Mars Crewed Spacecraft) of 30 tonnes on an escape trajectory from Earth orbit at a direction tangent to the Earth's orbit to maximize heliocentric orbital energy. The MCS will then dock and join with a SEP-MET propulsion unit containing Solar Panels, the MET engine cluster and water propellant. The SEP system with then be activated to add Delta V to the approximate Earths heliocentric orbital velocity of the MCS. The MET thruster cluster will produce approximately 600N of thrust operating on water vapor at 3MW in full sunlight and add Delta V to the assembled spacecraft to achieve MTI (Mars Transfer Injection) based on a new heliocentric elliptical orbit.

As is well known, the new heliocentric orbit's path will depend on only two parameters : the Specific Energy after the Delta V:

$$W=\frac{v^2}{2}-\frac{\mu}{r_e} \tag{3}$$

Where V= (V_θ + V_D) , the Earths orbital velocity being V_θ =29.8 km/sec plus the Delta V = V_D added by the SEP system, μ=GM_S where M_S is the mass of the Sun, all evaluated at the heliocentric Earth orbit radius, r_e, where the SEP burn was performed.

The other key parameter for the new heliocentric transfer orbit is the new specific angular momentum at the heliocentric Earth orbit radius, r_e

$$= (\boldsymbol{V_\theta} + \boldsymbol{V_D})\boldsymbol{r_e} \tag{4}$$

These quantities are used to calculate the new heliocentric orbit eccentricity

$$e = \sqrt{1+\frac{2Wh^2}{\mu^2}} \tag{5}$$

Following the standard path of analysis [10], we define the orbital size parameter p = h^2/μ and the semi major axis a= - μ/(2W) for the new heliocentric orbit of the MCS after the Delta V.

The equation for the transfer orbit radius then reduces to

$$r = \frac{p}{1+eCos\,v} \tag{6}$$

Solving for Cos ν_C , the value of the cosine of the heliocentric angle at the crossing point where r= r_m , the radius of Mars orbit, we obtain

$$Cos\nu_C=\left[\frac{p}{r_m}-1\right]/e \tag{7}$$

To calculate the time of flight to intercept Mars we must find the elliptic angle E_c of the crossing point:

$$E_c = Cos^{-1}\left(\frac{ae + r_m Cosv_c}{a}\right) \dots\dots\dots \tag{8}$$

We define the heliocentric orbit period T_p =$2\pi(a^3/\mu)^{1/2}$

So that we can use the standard Time of Flight equation to obtain the transit time T_t:

$$\boldsymbol{T_t} = T_p(E_c - eSinE_c) \tag{9}$$

Accordingly, we can generate a curve of T_t versus Delta V (see Figure 10.) It is seen that the time of transit decreases from the standard value of approximately 8.5 months, required for a Hohmann transfer at the minimum Delta V of approximately 3.5km/sec, to a situation of minimized returns at near 12km/sec, which is near the Delta V for escape from the Solar System from Earth orbit. The minimum T_p that can be achieved with this method of applying Delta V near Earth, where Solar Power is maximized, is then approximately ¼ of a year or 3 months. This is a considerable improvement over a Hohmann transfer trajectory, but requires large quantities of water propellant and also produces a high closure velocity for the MCS as it approaches Mars. Fortunately, water is easy to store in space in large quantities, and the Inflatable Heat Shield can easily produce Aerocapture and aerobraking at Mars for the MCS even at high closure velocities.

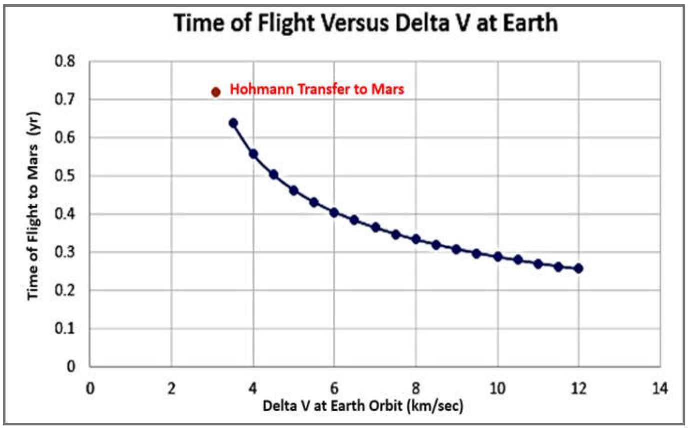

Figure 10 Time of flight to Mars versus Delta V. Note that the minimum Delta V shown here is approximately that for the Hohmann transfer.

Since we desire a Delta V of 12km/sec produced near Earth by our SEP system, how much time and how much mass of water is required by our system to place the MCS on the rapid transit trajectory to Mars?

The spacecraft fuel mass fraction of M_p, propellant mass , over the core spacecraft mass M_{sc}, assumed here to be 30 tonnes, is determined from the Tsiolkovsky Equation

$$\frac{\boldsymbol{M_p}}{M_{sc}} = exp(V_d/V_{ex}) - 1 \tag{10}$$

When this is done for M_{sc} =30 tonnes, which include the IHS and SEP MET propulsion unit, we obtain approximately 85 tonnes of water (see Figure 11.) This water is assumed to be placed on an escape trajectory from Earth and thus will require the equivalent of two SLS launches to preposition them for later docking and propellant transfer to the MCS for its trip to Mars. However, given the easy storability and transfer of water in space near Earth, this prepositioning of propellant on an escape trajectory can be accomplished by several smaller and less expensive launchers.

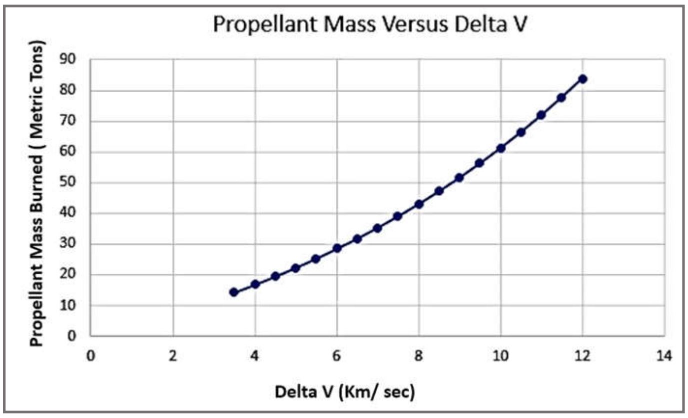

Figure 11 Water propellant mass requirements versus Delta V for the MMX mission.

We must now find the Burn time , T_B, for the SEP-MET system near Earth in order to achieve this Delta V of 12km/sec. To produce a thrust level of 600N the water propellant must be consumed at a rate of dM/dt = 0.0667 kg/sec.

The time for consumption of the required propellant mass is found from this flow rate and mass of propellant required.

$$Mp/T_B = {}^{M_p}\Big/\left(\frac{dM}{dT}\right) \tag{11}$$

For the mass of 85 tonnes the required burn time is approximately 14.6 days. Burn time is only 15% of time of flight to Mars at Delta V of 12km sec on this fast ransit trajectory (see Figure 12) so even for this large value the approximation of a point Delta V does not introduce significant errors. The variation of SEP burn-time in this model with Delta V is shown in Figure 13.

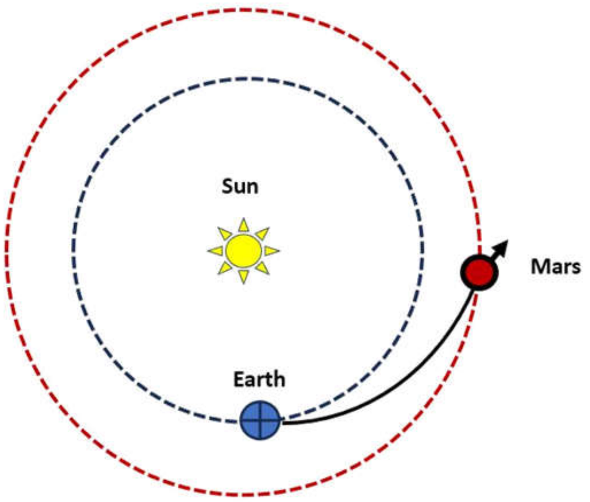

Figure 12A schematic of fast transit time transfer orbit for the MMX mission.

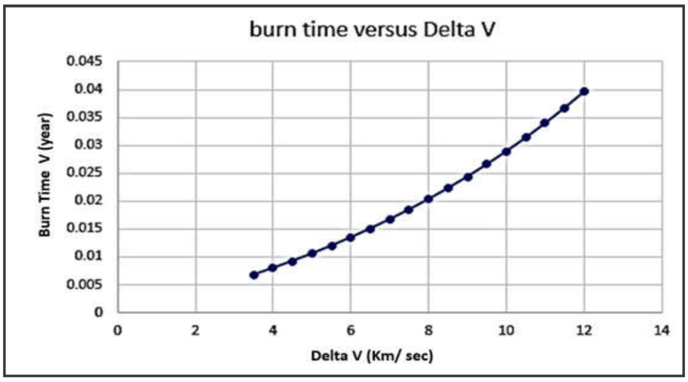

Figure 13A schematic of SEP-MET burn time for the fast transit time transfer orbit for the MMX mission.

The upward turn of the curve of the burn time versus Delta V for a fixed thrust level seen in Figure 13 at first seems counter-intuitive, however, it stems from the Tsiolkovsky equation's requirement that fuel mass fraction of the space craft must increase exponentially with Delta V.

APPROACH TO MARS AND LANDING

We can find the closure velocity of the MCS as it approaches Mars versus Delta V at Earth orbit (see Figure 14.)

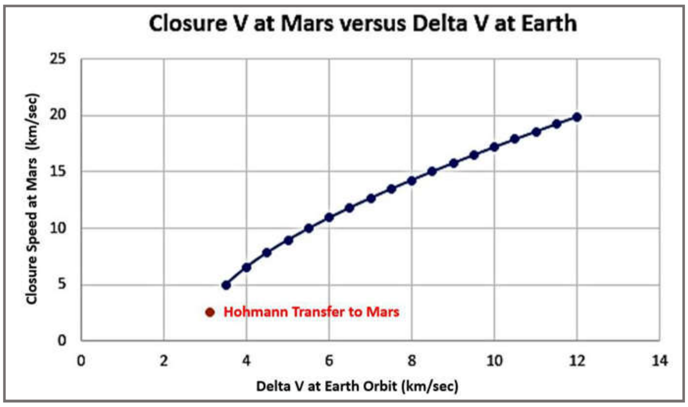

Figure 14 A graph of closure velocity versus Delta V at Earth for the fast transit time transfer orbit for the MMX mission. Approximate values for a Hohmann transfer are also shown.

As the CMS approaches Mars it does so at a high rate of speed, approximately 20km/sec , for a Delta V of 12km/sec at Earth orbit, to which will be added another 5km/sec as it approaches Mars due to Mars gravity. The IHS (Inflatable Heat Shield) will be the major source of braking for an insertion into a Mars orbit and later, a Mars landing, however the SEP unit will also be used, and this can be further optimized. For instance, more water propellant can be carried from Earth to increase SEP braking and lower g-forces on the crew at Mars aerobraking. However, we will assume here that, with properly chosen aerobraking trajectories, that a large SEP burn Mars, requiring more water to be carried or pre-positioned on a Mars transit trajectory, will be unnecessary. This will adhere to our overall concept of a simple core mission concept.

Before the CMS approaches the Mars atmosphere the CMS will engage its SEP propulsion and burn the 4 tonnes of water , formerly used as radiation shielding for the crew, in a braking maneuver and to make final course corrections. This produces approximately 1 km/ sec of braking Delta V and also lightens the ship. The crew will then withdraw into the Orion capsule with its attached IH Sand the other portions of the ship will be discarded onto a trajectory that misses Mars atmosphere and thus proceeds out into a deep space orbit.

The Aerocapture (see Figure 15) will obviously be the most stressful event of the Mars mission for both its crew and the population back on Earth and will require the CMS-IHS to shed approximately 19 km/sec in order to be captured by Mars gravity. This is roughly 70% above the 11km/sec experienced by the Apollo crews on their return from the Moon but should not present grave difficulties. Many different strategies for trajectories and crew comfort can be pursued to lessen the g-forces effects experienced by the crew. Here we will take a basic and approximate approach of assuming that the Aerocapture maneuver, resulting in the CMS being in an elongated orbit will be successful and that subsequent aerobraking maneuvers to lower the orbit can be accomplished leading finally to a decent to the Mars surface. (see Figure 15)

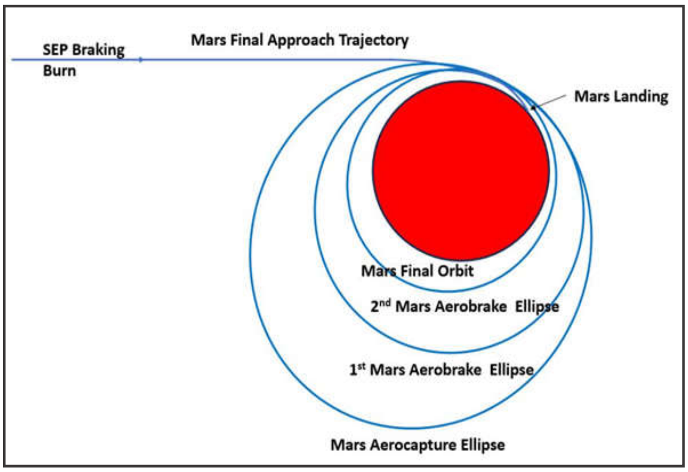

Figure 15A schematic of Mars aerocapture, aerobraking and landing trajectories

It is assumed that a prepositioned Mars habitat will be waiting at the landing site. The site should include a Mars rover capable of remote piloting to pick up the crew if they miss their landing zone. This habitat site, with Radioisotope Thermal Generators supplying power and heat will serve as the base for the crews exploration of Mars.

It is also assumed that a MAV (Mars Ascent Vehicle) will also be prepositioned at the habitat site to allow the crew to ascend to Mars orbit, and rendezvous with either a MRS (Mars Return Spacecraft), or even to reach a “forward base camp” crewed or uncrewed previously established on Phobos. Manufacture of Mars return fuel on Mars , as proposed by Robert Zubrin, is both highly feasible and highly desirable for follow on missions [11]. However, here, for this early rapid mission, we will assume a simple , no frills, approach with a MAV powered by storable liquid fuels like the Apollo Moon lander/ ascent vehicle [12].

SUMMARY AND DISCUSSION

Therefore, three important, newly demonstrated advances in space technology, the SLS, the MET with water propellant, and the inflatable heat shield, plus the already well demonstrated feasibility of Megawatt class Solar power in space, can all be combined into a simple Human Mars Mission architecture, with fast transit time to Mars and fast program development, and good crew safety.

Obviously, many variations and improvements in modeling on this preliminary, basic architecture can be explored and optimized. In particular, calculational exploitation of favorable launch windows using detailed orbits, and the exploitation of planetary gravity fields can result in lowered delta Vs, faster transit times, and lowered fuel masses and consequent increased payloads. Also, the prepositioning of a crewed “forward base camp” on Phobos confers many advantages and can be done using a similar architecture as the MMX. However, in general, the steady advance of space technologies now makes a straightforward fast-tracked and fast transit crewed mission to Mars appear quite feasible, based on this preliminary study. This can be regarded as the natural consequence of the human desire to become space faring, ‘first to Mars and then to the stars. ‘

Acknowledgements

The author wishes to thank Brent Nelson , Jason Dvorkin and Mike Smith of Kepler Aerospace for their encouragement and support of this research.

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