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A SEP (solar electric propulsion) deep space rendezvous for mars mission concept based on the MET (microwave electro-thermal) thruster

Abstract

A human mission to Mars concept based around Solar Electric Propulsion (SEP) and theMET (microwave Electro-Thermal) thruster with water propellant is proposed and analyzed. This architecture utilizes water propellant transfer between spacecraft to increase payload delivered to Mars where the architecture is compared to a purely chemical baseline using the Saturn V booster from the Apollo missions. It is discovered that despite the advantage of the Oberth Effect early in the mission to lower DV for MTI (Mars transfer orbit injection) the requirement of storable chemical propellants for MOI (Mars Orbit Insertion) with lower I_{sp} greatly reduces the amount of useful payload delivered into Mars orbit. In contrast the use of the MET SEP, with much higher I_{sp} than chemical propellant, as a high energy upper stage on the Saturn V together with using the demonstrated transferability of water between spacecraft, allows a transfer of water propellant from a tanker after a DSR (Deep Space Rendezvous) before MTI and possesses a 3.1 times increase of useful payload delivered to Mars orbit.

INTRODUCTION

At this time, proposals for human missions to Mars are predominantly centered around mission plans or architectures using chemical or nuclear-thermal propulsion. The reason for this stems from the Oberth effect and advantages its gives to these architectures where delta V increments are delivered in brief periods of high thrust. The Oberth effect allows reasonable payload masses and short flight times to Mars, and indicates that human missions are feasible in the coming decades using present chemical propulsion technology. Electric propulsion, conversely, is limited mostly to low thrust delivered over long periods. This is especially true for SEP architectures which are limited in power to approximately a megawatt. Hence, SEP concepts cannot make use of the Oberth effect near Earth. This lowers payload and increases flight times for human crews, exposing them to more radiation from cosmic rays or solar flares. One SEP architecture will be examined that isa variantof the original Solaris(1)mixed-electric-chemical architecturebuilt around the MET thruster(2). This thruster offers high thrust compared to other electric propulsion technologies and also allows propellant transfer in space, as well ascompare an Oberth-assisted chemical architecture. We will use as our chemical baseline propulsion system, the Saturn V (3)used for Lunar missions, since it represents the demonstrated "state of the art" for man-rated heavy lift propulsion that is Mars-capable. We will also look at the problem of a standard 10 month Hohmann transfer to Mars (4)and Mars capture to provide a representative ΔV budget.
Given that we will be looking at minimum energy trajectories to Mars flight time, this will be considered for a fixed parameter of approximately 10 months. Faster trajectories are certainly available, but at the cost of payload. Insertion into Mars orbit at the orbit of Mars' moon Diemos will also be assumed since aerobraking on Mars atmosphere will be acceptable for cargo but will not be considered for early human Mars missions because of its danger.
Therefore we will consider as a baseline chemical architecture a Saturn V launch making full use of the

Oberth effect for a liquid hydrogen S-IV B stage of empty mass 13, 000kg (1) with two burns. Since liquid hydrogen and oxygen cannot be stored for 10 months in space because of Solar heating, we will assume a storable liquid propellantcombination upper stage, such as NTO (Nitrogen tetroxide) and UDMH (Unsymmetrical Di-Methyl-Hydrazine) is used for Mars orbit insertion with empty mass 2000kg. Using these assumptions we can arrive at an approximate payload mass placed in Mars orbit by delivering into lunar transfer orbit and assuming a slightly higher ΔV being required for Mars transfer injection. It is found that the Oberth effect is strongest in the initial Mars transfer orbit injection burn from LEO (Low Earth Orbit), but that the use of chemical propellants causes large payload penalties, relative to SEP, later in the mission at Mars orbit injection and capture.

SATURN V TO MARS CHEMICAL PROPELLANT BASELINE

Given variations in the Earth-Mars orbit relationship, the Mars transfer orbit injection DV will be approximately 4.3km/sec from LEO using the Oberth Effect, rather than the ΔV =3.0km/sec (2)to Mars from a deep spacetransfer from Earth's Solar orbit. This is higher than thelunartransfer orbit injection of ΔV= 3.1 km/sec and lowers the payload to Mars accordingly over that sent to the Moon: the Apollo CMS and Lunar Lander plusthe empty SIV-B tankage and structure of 13,000kg.

The payload delivered to Mars transfer orbit must also include in addition to crew module and supplies, a chemical upper stage using storable propellants. This produces aI_{sp} of 300seconds with an empty mass of 2000kg. The required ΔV for Mars capture and assumption of orbit at the orbit of Diemos is ΔV = 1.1km/sec. The storable propellant must make up 30% of the mass brought to Mars –Deimos orbit. Aero-braking at Mars can considerably reduce this required propellant but must also increase the danger to a human crew. Therefore, we will consider the mass delivered to Diemos orbit as a baseline for this study.

Total empty mass of the Saturn S-IV-B upper stage was59,460kg. The ΔV =3.3 km/sec was approximate delta V from LEO to Lunar transfer orbit, which is approximately Earth escape velocity. The S-IV-B delivering a useful payload into lunar orbit 45,960kg, which consisted of the Apollo Command Service Module and Lunar Lander neglecting S-IV-B tankage andstructure.

Using rocket equation to give mass fraction, with DV =3.3km/sec and V_{ex} = 4.1 km/ sec ($\Delta V/V_{ex}$=0.8) we obtain:

$$e^{(V/V_{ex})} = M_o/M_f = 2.2 \quad (1)$$

Or a ratio of fuel to empty mass M_f of 1 /2.2 =0.45 or (M_o/M_f= 2.2)

$$\frac{M_o}{M_f} = \frac{M_{prop} + M_f}{M_f} = \frac{M_{prop}}{M_f} + 1 \quad (2)$$

$$\frac{M_{prop}}{M_f} = 1.2 \quad (3)$$

This means for M_f = 59,460kg that a fuel mass of 71, 352kg is required, which we will assume is the maximum propellant load.

Assuming the same amount of propellant, and a S-IVB could send a total mass on a Mars transfer orbit of with ΔV =4.3km/sec we have then($\Delta V/V_{ex}$=1.0):

$$e^{(V/V_{ex})} = M_o/M_f = 2.9 \quad (4)$$

$$\frac{M_{prop}}{M_f} = 1.9 \quad (5)$$

This means payload to Mars transfer orbit that is 37, 554kg, whichmust include 13, 500kg of the SIV-B structure, leaving 24, 053kg of useful payload in Mars transfer orbit.

Upon reaching Mars space, the human Mars craft must achieve capture into a useful orbit around Mars, this

Figure 1 : The saturn V was very successful and is representative of a large chemical booster build to support mars exploration

requires an approximate $\Delta V = 1.1$km/sec to assume Deimos orbit, which we assume here is a useful orbit around Mars for staging various architectures to Phobos or the surface.

This requires the burning of storable propellants NTO andUDMH with an I_{sp} of 310 seconds (V_{ex} = 3km/sec). A propulsion stage of approximately 2000kg is required to burn this propellant.

This final burn further reduces the payload to be delivered to Mars orbit.

Assuming the same amount of propellant, and a SIV-B, we could send a total mass on a Mars transfer orbit of with $\Delta V = 4.3$km/sec we have ($\Delta V/V_{ex} = 0.37$)

$$e^{(V/V_{ex})} = M_o/M_f = 1.45 \quad (6)$$

$$\frac{M_{prop}}{M_f} = 0.45 \quad (7)$$

This means useful payload to Mars space is 24,053kg of which only 16,614 kg will end up in Mars orbit, of which 2000kg is the storable propellant upper stage empty mass and is not reusable. This leaves 14,614 kg of useful payload in Mars orbit. Thus, a Saturn V, which was a real operational system, can deliver approximately 14.6metric tons into a useful Mars orbit from LEO. This is considered a baseline calculation. We can now compare this to the MET–DSR architecture.

THEMET DSR SEP ARCHITECTURE

The MET (2) is a new EP device that uses a vortex stabilized, electrodeless, microwave discharge to superheat propellant to create thrust. It is is the only thruster system demonstrated to use water vapor as propellant, and can use 915MHz microwaves, which can be generated with 95% efficiency.

The low thrust of electric propulsion presents two problems relative to chemical propulsion. First, it cannot take advantage of the Oberth effect in Earth orbit to achieve a lowering of the required ΔV to reach Mars Transfer orbit injection. Secondly, the low thrust of the electric propulsion suffers severe gravity losses duringthe required 'spiral out' phase of the mission required to leave Earth's gravity well. These gravity losses are due to the inherently non-optimal pointing of the thrust along the velocity vector that must occur during an escape-trajectory, which means it must develop a component to oppose gravity directly, like at lift-off, which means wasted thrust. This gravity loss effect doubles the actual ΔV = to 6km/sec from the 3km/sec required to achieve escape from LEO by a chemical burn. However, the higher I_{sp} of the SEP system gives considerable advantage at the Mars capture and orbit phaseover chemical propulsion. Also, of considerable importance, is the demonstrated capability to transfer water from one spacecraft to another. Therefore, we will use the Saturn V system to boost the MET SEP system out of the Earth's gravity well, taking advantage of the Oberth effect in LEO to reach escape. We will then rendezvous with a water tanker in deep space placed there by a previously launched smaller, low-cost booster, such as a Falcon 9.

The Mars mission vehicle will take on water from the tanker to accomplish a 3km/sec burn for Mars transfer orbit injection, plus 1.1 km/sec burn at Mars to achieve capture into a usefulMars orbit. This will require a 4.1 km/ sec to deliver the full payload to Mars orbit ($\Delta V/V_{ex} = 0.46$)

$$e^{(V/V_{ex})} = M_o/M_f = 1.57 \quad (8)$$

$$\frac{M_{prop}}{M_f} = 0.57 \quad (9)$$

Thus we begin with an assumption of the Saturn V system delivering a useful payload into escape of 45,960kg the including a 2000kg dry mass MET SEP unit. We are going to take this payload to Mars orbitusing the MET SEP unit and the water taken on from the tanker. We will take on 26,197 kg of water propellant to accomplish this taskto be transferred from the tanker in deep space. This can be readily and safely accomplished, as demonstrated by the ISS resupply missions, since water has a low vapor pressure at ambient space temperatures and is not toxic or explosive.

We assume here a package of lightweight water tank, solar panels, and engines of approximately 2000kg (approximately 0.5kg/kW) with a power of 1 Megawatt at Earth orbit. We will assume a thrust per unit power of 0.2N/kw, efficiencies of 97% and an I_{sp} of 900sec, or V_{ex} = 9km/sec. Thus with full sunlight at Earth orbit, the MET SEP propulsion will generate approximately 200N of thrust. We include this with the manned spacecraft payload since this payload can function as an abort propulsion system using on-board water for a return to Earth orbit, if MTI is canceled.

We can work backwards from Mars orbit where 45,960 kg has been delivered after ΔV of 1.1 km/sec.

The Mars vehicle must perform a $\Delta V = 1.1$ km/ second burn at Mars to assume a useful Mars orbit ($\Delta V/V_{ex} = 0.11$).

$$e^{(V/V_{ex})} = M_o/M_f = 1.12 \quad (10)$$

$$\frac{M_{prop}}{M_f} = 0.12 \quad (11)$$

Thus, the final payload mass 45,960kg delivered into useful Mars orbit using the MET SEP willburn 5,515 kg of water of the original load of 26,197 kg. The final payload will be accomplished at a solar flux of

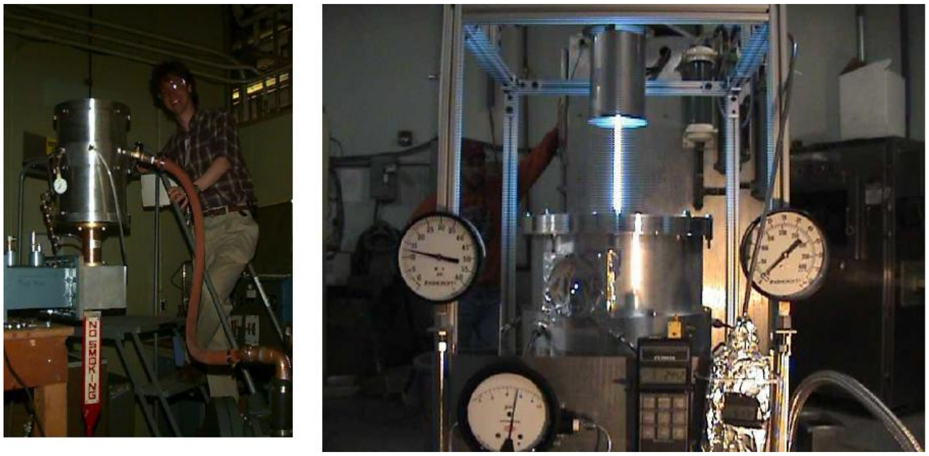

Figure 2 : The MET 75kW rocket engine operating with water propellant in air

approximately 0.43 of Earth given the average Mars distance from the Sun of 1.52 AU. This means the power and thus thrust will be only 43% of that at Earth orbit. Thus the rate of water consumption will be reduced by the same factor. The 5,515 kg of water will be consumed at a rate of 0.00946kg/sec and thus take 582,981 seconds or approximately a week (5.9x 10^5 sec /week). This propellant water would be normally separated from that required for use by the human crew during the mission.

To perform a ΔV =3km/ second burn to assume Mars transfer orbit will burn 20,682kg of water. We assume here a package of lightweight water tank, solar panels,and engines of approximately 2000kg (approximately 0.5kg/kW) with a power of 1 Megawatt at Earth orbit. We will assume a thrust per unit power of 0.2N/kw, efficiencies of 97% and an I_{sp} of 900sec, or V_{ex} = 9km/sec. Thus is full sunlight at Earth orbit the MET SEP propulsion will generate approximately 200N of thrust.

This means the amount of water to boost the entire payload to Mars transfer orbit, including the 2000kg MET propulsion unit and 4,506 kg of water for Mars orbit capture is 72,157 kg of water of the propellant to be transferred from the tanker in deep space. The total vehicle mass of payload and propellant will thus be 72,157 kg at the beginning of the Mars Transfer orbit injection burn. At thrust of 200N, the water will be consumed at the rate of

$$\frac{dM_{prop}}{dt}=\frac{T}{V_{ex}}=\frac{.022kg}{\sec} \quad (12)$$

or 45 sec per kg of propellant consumed. The ΔV of 3km/sec will be accomplished in a time period of 9.3x 10^5 seconds or approximately 1.5 weeks.

Thus 45, 960 kg is delivered to useful Mars orbit. This versus 14,614 kg for the purely chemical system. Therefore the MET SEP can deliver 3.1 times as much payload to Mars. MET DSR architecture thus delivers roughly 3times the payload to useful Mars orbit.

SUMMARY AND DISCUSSION

Thus, based on this analysis, a mixed chemical MET SEP architecture for Mars using a DSR with propellant transfer can deliver considerably more payload to useful Mars orbit. This occurs because of the high I_{sp} of water burned in the MET but also because of the ease and safety of transfer of water propellant between spacecraft, which is a feat previously demonstrated on every ISS resupply mission. Possible architectures for Lunar exploration, using propellant transfers were studied during the Apollo era and produced advantages in Lunar payload,but they were not pursued due to the volatileand cryogenic nature of liquid hydrogen andLOX propellants withsafety concerns for the manned mission. Now with water fuel, propellant transfer can be accomplished easily and safely and the merits of propellant transfer in a DSR scenario can be realized.

The demonstrated ability to transfer propellant is largely confined to the MET system, since it is the only operational electric thruster to use water as propellant. Other EP systems use xenon gas which is a mild cryogen and thus quite volatile.

REFERENCES

[1] J.E.Brandenburg, John F.Kline, Ronald Cohen, Kevin Diamante; "Solaris: A low cost human mars mission architecture based on solar electric propulsion" Proceedings of the 2001 Mars Society Symposium at Stanford University, **(2001)**.

[2] J.E.Brandenburg, J.F.Kline, D.F.Sullivan; "The microwave electro-thermal (MET) thruster using water as propellant "IEEE Transactions On Plasma Science, **33(2)**, 776 **(2005)**.

[3] Bilstein, E.Roger; *Stages to saturn: a technological history of the apollo/saturn launch vehicle*, DIANE Publishing, ISBN 9780788181863, 59–61 **(1999)**.

[4] R.R.Bate, D.d.Mueller, J.E.White; "Fundamentals of Astrodynamics" Dover Publications NY, 365, **(1971)**.