Analysis of Launch and Earth Departure Architectures for Near-Term Human Mars Missions

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This paper presents a comparative analysis of launch and Earth departure strategies for human Mars missions. A variety of Earth departure architectures are analyzed with regard to their trans-Mars injection capabilities (performance surrogate metric) and equipment and operational requirements (cost surrogate metric); it is assumed that aerocapture and chemical propulsion are used for all maneuvers in Mars vicinity for all architectures. The architectures are based on chemical propulsion (custom stages or Ares V Earth Departure Stage) as well as nuclear thermal propulsion. Consideration is also given to the impact of different Earth departure options on Mars aerocapture and Mars entry, descent and landing. The comparative aspect of the analysis consists of an iso-TMI mass analysis for the different options. Results of the set of architectures indicate that while chemical Earth departure strategies results in a 30 – 50 % increase in the number if Ares V launches required per mission, the associated additional marginal cost may be outweighed by the cost of developing and maintaining a nuclear thermal propulsion capability, as well as the increased marginal cost of nuclear thermal propulsion stages. In addition, chemical Earth departure strategies side-step the sensitive issue of space nuclear applications that would be associated with nuclear thermal propulsion.

I. Introduction and Motivation

THe human exploration of the surface of Mars remains the expressed goal of human spaceflight for the foreseeable future[1]. Among the significant technical challenges associated with human Mars missions are the launch of the mission elements, Earth orbital operations, and subsequent injections into trans-Mars trajectories. Three major Earth departure propulsion concepts have emerged[2,3,4,5,6,7,8,9]:

- **Chemical propulsion**: usually a multi-stage tandem departure; more recent strategies based on LOX/LH_2 propellants for high specific impulse. Chemical propulsion is usually combined with Mars aerocapture to reduced overall mission mass.
- **Nuclear thermal propulsion**: used either only for Earth departure or for all of Earth departure, Mars capture, and trans-Earth injection (the latter strategy is enabled by the proposed bi-modal capability of the reactor system for both propulsion and electricity generation). For some mission architecture proposals the crew would rely on nuclear thermal propulsion only for all in-space maneuvers, while cargo would be injected towards Mars using nuclear thermal propulsion and then subsequently aerocaptured.
- **Electric propulsion**: when coupled with solar power generation this option is usually only used for raising the orbit of Mars payloads; final injection towards Mars is still carried out using chemical propulsion. When coupled with a nuclear fission power source, electric propulsion has been proposed for all in-space maneuvers during a human Mars mission.

In the literature, analyses for individual Earth departure strategies tend to be presented without comparison to other options; this usually means that each option is designed around the conditions which lead to optimal performance. Such previous studies have predominantly indicated that some form of advanced propulsion

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technology (i.e. nuclear thermal or electric propulsion) is highly desirable for human Mars missions. A comprehensive analysis of the entire Earth departure architecture space, however, has not been carried out.

In this paper, we present a systematic comparative analysis of chemical Earth departure options. The motivation for this analysis is two-fold: firstly, it represents the first step of a comprehensive analysis in a part of the Earth departure architecture space which has not been analyzed in as much detail as the parts involving advanced propulsion; secondly, chemical propulsion is interesting because it may allow for side-stepping significant technology development and political robustness issues associated with advanced propulsion. The chemical options are compared to a reference nuclear thermal propulsion option. Section II introduces the specific Earth departure architectures analyzed; it is assumed that aerocapture and chemical propulsion in Mars vicinity is used in all cases. In Section III, the results of performance analyses for each architecture option are presented. Section IV contains the comparative part of the analysis: the launch and ground processing requirements for each architecture are determined and compared for a range of mission mass requirements at TMI (iso-TMI mass analysis). Section V summarizes major findings and provides suggestions for future work.

II. Earth Departure Options for Human Mars Missions

As mentioned in the preceding section, the work in this paper is focused on chemical and nuclear thermal Earth departure architectures. Four options were specifically investigated:

- Option 1: chemical Earth departure using two 100 mt custom stages in tandem
- Option 2: chemical Earth departure using both a 100 mt custom stage and the Ares V EDS
- Option 3: chemical Earth departure using both a smaller custom stage and the EDS
- Option 4: chemical Earth departure strategy using only the Ares V EDS
- Option 5: reference nuclear thermal Earth departure strategy

For each of these options a detailed description of hardware elements and operations is provided in the following subsections. It is assumed that for human Mars missions the Ares V launch vehicle is available.

A. Description of Earth Departure Option 1

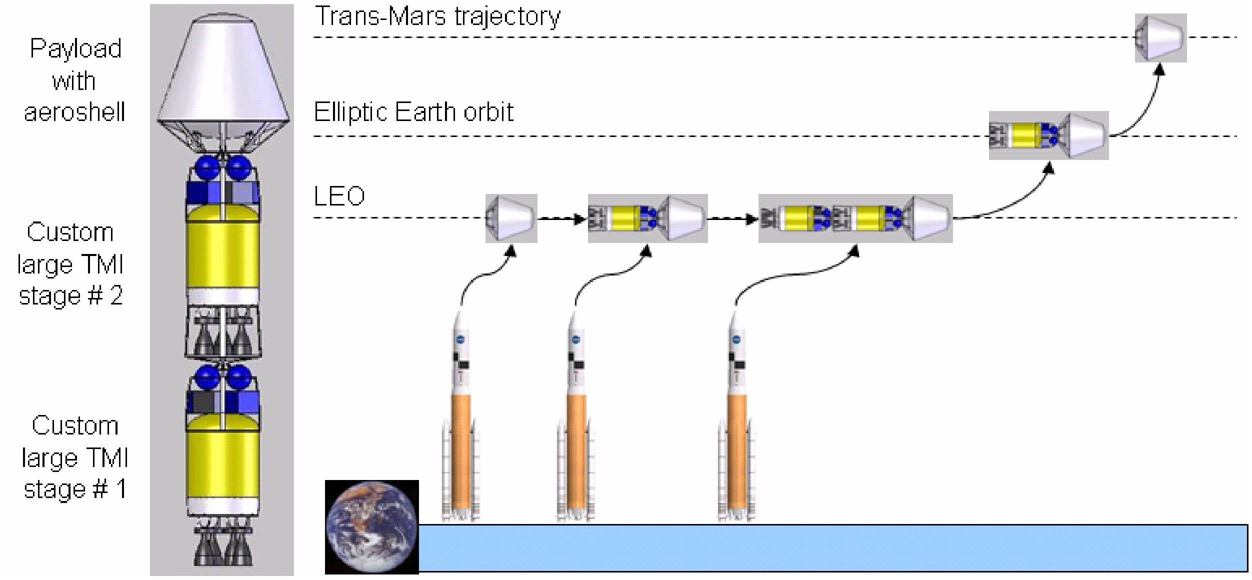

Figure 1: Configuration for Earth departure Option 1 based on 3 Ares V launches per payload and 2 custom TMI stages with RL-10-B2 engines used sequentially (Ares V image courtesy NASA)

Option 1 (see Figure 1) is based on the use of the Ares V and a custom large (i.e. designed to the LEO payload limit of a dedicated Ares V launch) chemical TMI propulsion stage. The TMI stage utilizes 4 or 6 RL-10B-2 LOX/LH_2 engines. Per payload injected towards Mars, three Ares V launches are required: the first launch delivers the payload in an aeroshell to a 400 km Low Earth Orbit (LEO). The payload loiters in this orbit until a second Ares V launch delivers one of the TMI stages, which is docked to the payload. This stack loiters until a third Ares V launch delivers a second TMI stage, which is also docked. After checkout of the integrated stack, this stage is burned, placing the stack into an elliptic Earth orbit which results in reduced thermal input to the remaining propulsion stage from the Earth. The other TMI stage, which is still attached, is burned to place the payload on a trans-Mars trajectory when the Earth-Mars transportation window opens.

The operations described above apply to cargo transportation only; for crew transportation the entire stack remains in LEO until the crew arrives (presumably launched on an Ares I). A 2-burn Earth departure is then carried out when the Earth-Mars transportation window opens.

B. Description of Earth Departure Option 2

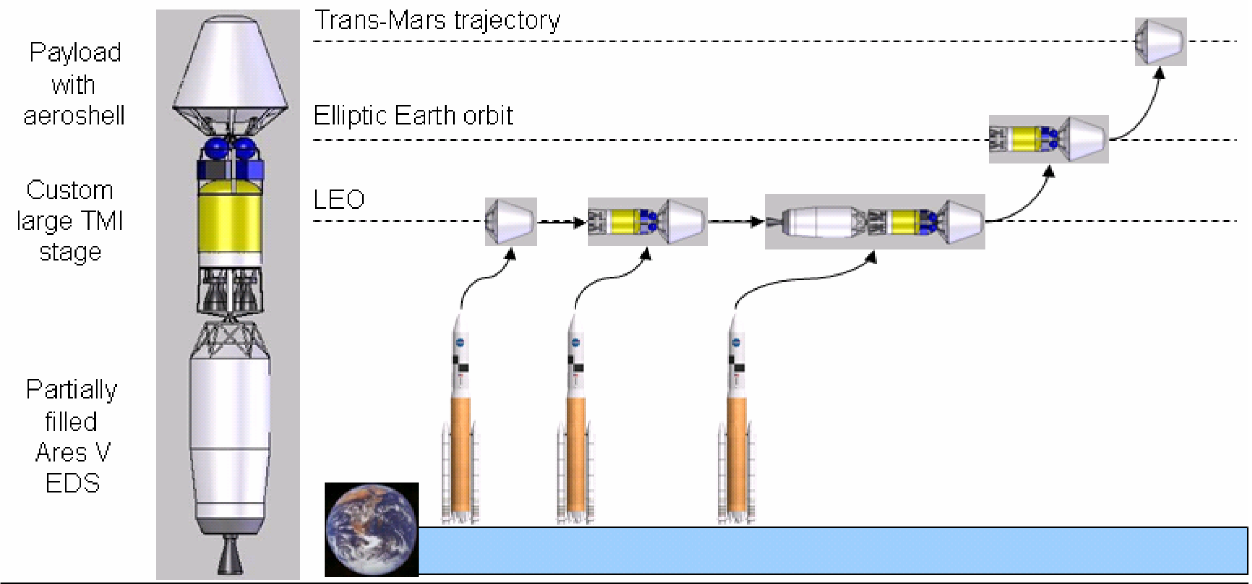

Figure 2: Configuration for Earth departure Option 2 based on 3 Ares V launches per payload, a single custom TMI stage with RL-10-B2 engines, and one partially filled Ares V EDS

Option 2 (see Figure 2) is similar to Option 1, but utilizes only one custom large TMI stage, and one Ares V EDS. This partially filled EDS is part of the Ares V launch vehicle (i.e. the EDS used for Earth departure is also the Ares V upper stage). Given that the EDS is only partially filled when used for the first Earth departure burn, the payload capability of Option 2 is lower than that of Option 1; however, there is a cost advantage because one less custom TMI stage is required than in Option 1 (the EDS which is used for Earth departure in Option 2 is also required for Option 1). As for Option 1, the stack used for crew transportation would remain in LEO until TMI.

C. Description of Earth Departure Option 3

Option 3 (see Figure 3) is based on the use of the Ares V EDS, as well as a small chemical propulsion TMI stage (i.e. the stage mass is significantly smaller than the Ares V LEO payload capability). For each payload injected towards Mars, two Ares V launches are required. The operations for cargo transportation are as follows: the first Ares V launch delivers the small TMI stage together with the payload and aeroshell into a 400 km LEO parking orbit. This stack loiters until the EDS or the second Ares V is docked. The EDS subsequently burns its remaining propellant to place the small TMI stage and payload into a highly elliptic Earth orbit. When the Earth-Mars transportation window opens, the small TMI stage is used to inject the payload towards Mars.

For crew transportation, the stack with the partially filled EDS remains in the LEO parking orbit until the crew arrives. The stack then loiters with the crew until the Earth-Mars transportation window opens; then the EDS stage is burned, placing the payload and TMI stage into a highly elliptic orbit. At the next pericenter, the TMI stage is then burned to place the payload on a trans-Mars trajectory.

For Option 3, there are to sub-options based on the propellant combination for the small TMI stage:

- Option 3.1 uses RL-10B-2 LOX/LH_2 engines for the small TMI stage
- Option 3.2 uses RL-10-derived LOX/LCH_4 engines for the small TMI stage

The motivation for including a LOX/LCH_4 propulsion option is the reduction of propellant / fuel boil-off due to the significantly higher boiling point of methane, albeit at reduced performance / specific impulse. The engine used for LOX/LCH_4 propulsion would not have to be a new development: assuming that the Mars vicinity propulsion system is based on LOX/LCH_4 propulsion, its engine design could be re-used for Earth departure propulsion.

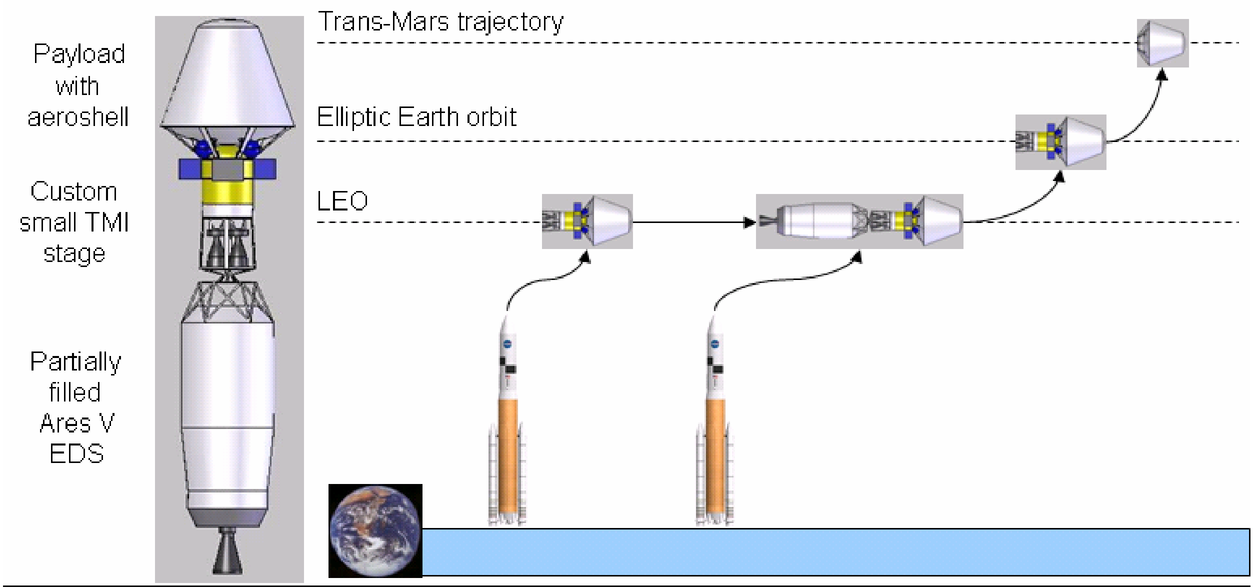

Figure 3: Configuration for Earth departure Option 3 based on 2 Ares V launches per payload, a single custom TMI stage which is launched with the payload, and a partially filled Ares V EDS

D. Description of Earth Departure Option 4

Option 4 is based on using only the Ares V EDS as the TMI propulsion stage. Two sub-options are considered: using one or two EDS stages per payload. The motivation for Option 4 is that neither development of a custom TMI stage is required nor the manufacturing of any stages dedicated to Earth departure: all EDS stages are also used as part of Earth launch.

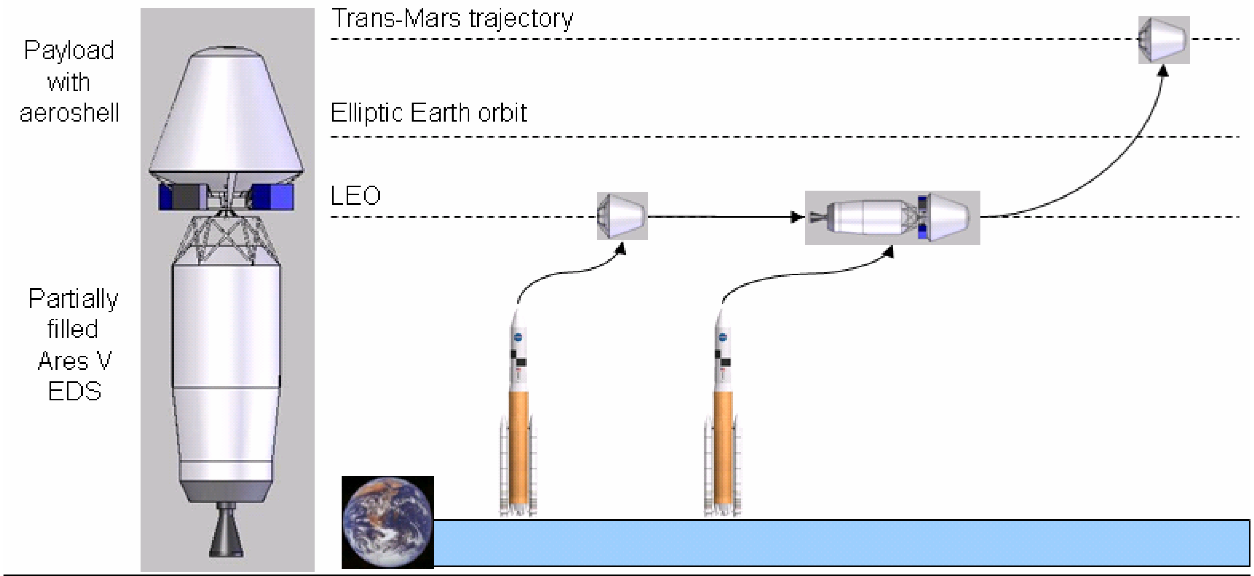

Figure 4: Configuration for Earth departure Option 4.1 based on 2 Ares V launches per payload and a single partially filled Ares V EDS

The operational concept for the single EDS strategy (Option 4.1) is shown in Figure 4. Two Ares V launches are required per payload injected towards Mars. The first delivers the payload and aeroshell to a 400 km LEO parking orbit; the second Ares V delivers its partially filled EDS into a 400 km orbit. The two elements are docked and loiter in the parking orbit until the Earth-Mars transportation window opens; then the remaining propellant in the EDS is burned and the payload id injected towards Mars. Crew and cargo operations are identical (except for the docking of the crew).

Option 4.2 is based on the use of two Ares V EDS for Earth departure propulsion (see Figure 5); the operational concept is similar to those of Option 1 and Option 2 above. The major difference is that no dedicated TMI stage is required (both development and launch) because the EDS stages used for Earth departure are also used as upper stages on the Ares V for Earth launch. As for Options 1 and 2, the cargo operations include burning one EDS to place the stack into an elliptic Earth orbit which has reduced thermal input from Earth for boil-off protection, where as for crew transportation the stack remains in the LEO parking orbit until the Earth-Mars transportation window opens. Because the EDS stages used for TMI are both only partially filled, the performance of Option 4.2 is reduced compared to both Options 1 and 2; the cost of Option 4.2 is, however, also expected to be lower.

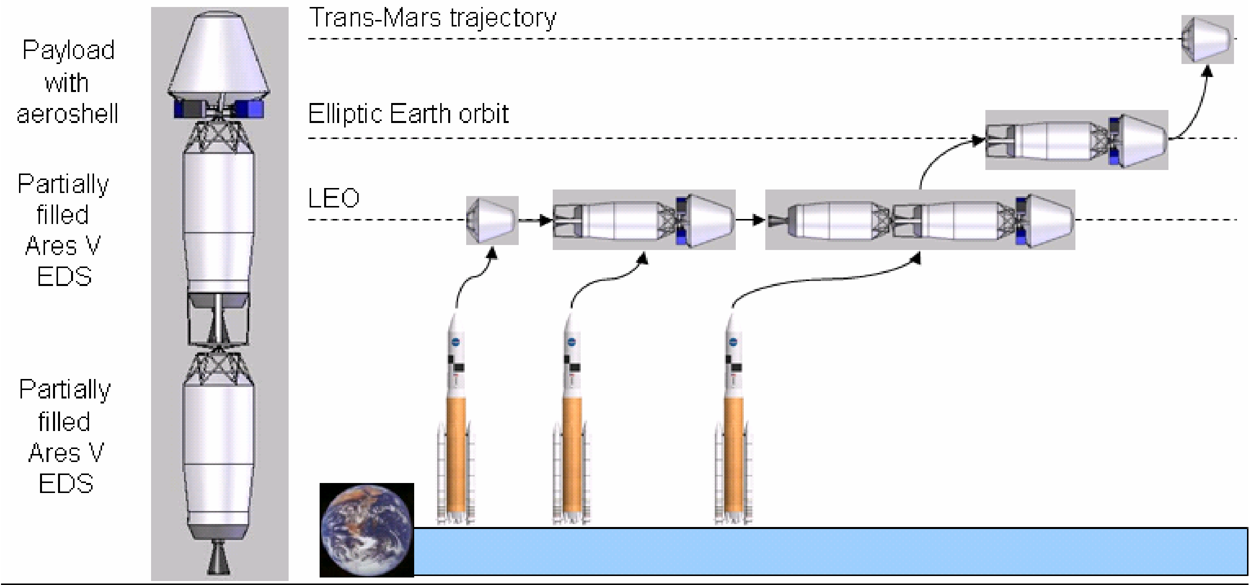

Figure 5: Configuration for Earth departure Option 4.2 based on 2 Ares V launches per payload and two sequentially burned partially filled Ares V EDS stages

E. Description of Earth Departure Option 5

As mentioned in the introduction, the motivation for the analysis presented in this paper was to carry out a comparative assessment of the chemical Earth departure propulsion architecture space. For this comparison to be relevant it is necessary to include a reference concept for an advanced propulsion Earth departure strategy; Option 5 serves as this reference concept.

Option 5 (see Figure 6) is based on the use of nuclear thermal propulsion systems, as well as aerocapture of crew and cargo at Mars; it is therefore similar to the Earth departure architectures for NASA Mars Design Reference Missions 1.0 and 3.0[3,4]. Two Ares V launches are required per payload injected towards Mars: the first delivers the payload and aeroshell to a 400 km LEO parking orbit, the second delivers a nuclear thermal propulsion stage which is docked to the loitering payload. The nuclear thermal stage is burned to depletion to place the payload on a trans-Mars trajectory once the Earth-Mars transportation window opens. It is assumed that the nuclear thermal propulsion stage can be burned safely at 400 km altitude and does not have to be delivered to a higher "nuclear safe" orbit. Operations are identical for crew and cargo transportation.

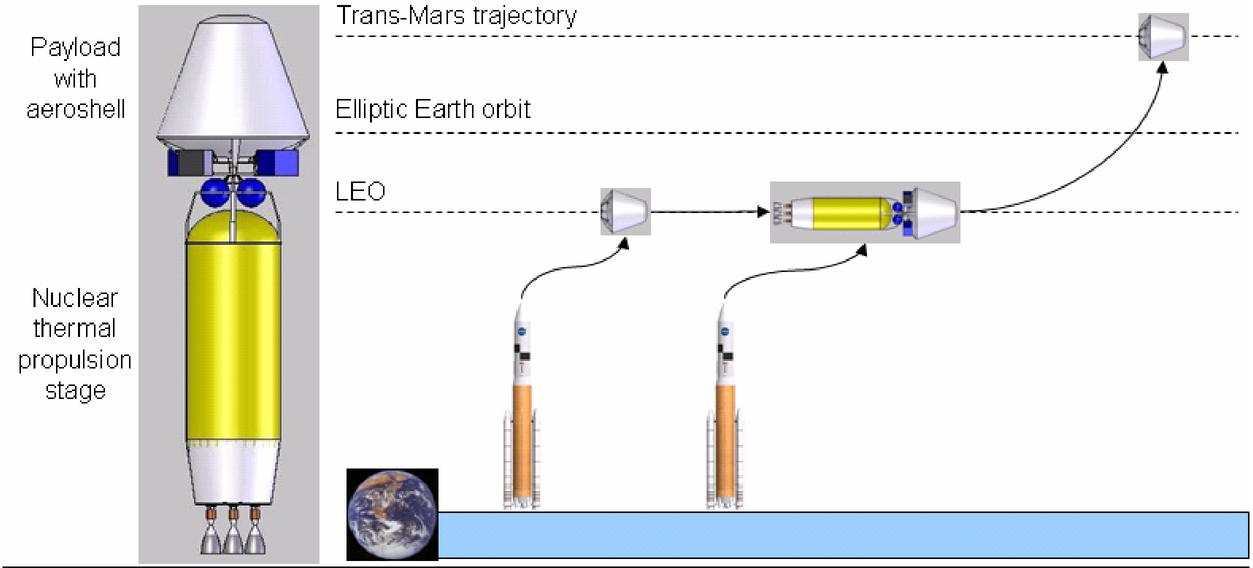

Figure 6: Configuration for Earth departure Option 5 based on 2 Ares V launches per payload and a single nuclear thermal propulsion stage

III. Quantitative Analysis Results

For each of the options described above, two types of analyses were carried out: an assessment of the payload injection capability of each option including sensitivities to delta-v changes and propellant boil-off, and an analysis of the launch requirements for a given trans-Mars injection mass requirement using a particular option. The analysis was based on estimated Ares V performance characteristics shown in Figure 7 as propellant remaining in the Ares V EDS as a function of Ares V LEO payload (values are for a 400 km LEO). If the EDS is burned to depletion, the resulting net payload in LEO is assumed to be 110 mt.

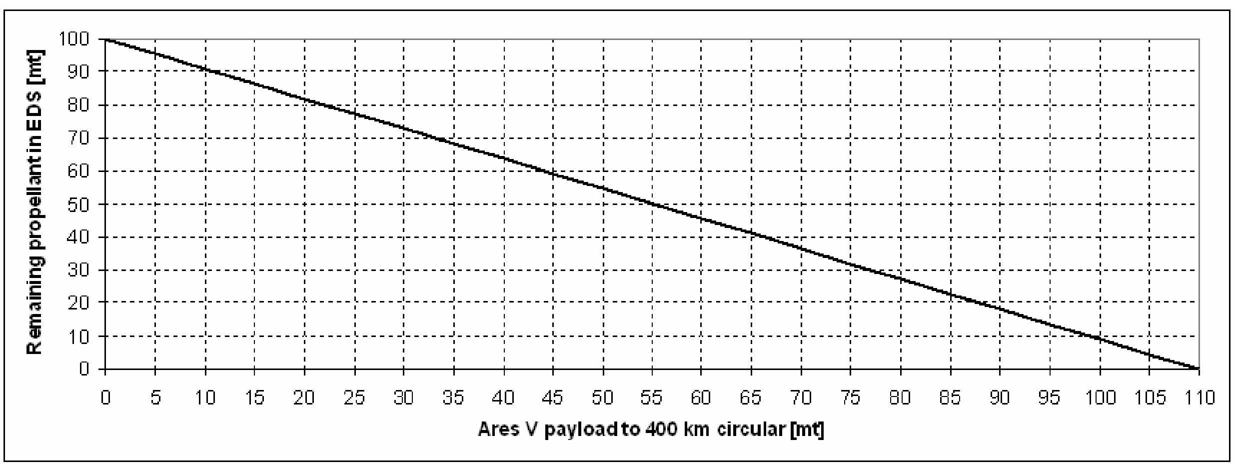

Figure 7: Ares V performance assumptions for a 400 km LEO staging orbit

All custom TMI propulsion stages (used in Options 1, 2, and 3) were modeled in Microsoft Excel following the methodology presented in reference 10 as outlined in Table 1. The total wet mass of the stage was set; either at 110 mt or at 40 mt, and the Goal Seek function in Microsoft Excel was used to determine the sizes of the various components based on the maximum amount of usable propellant. The following sections outline the steps shown in Table 1 using the 110 mt 6 LOX/LH_2 engine stage as an example. The other three stages were modeled in a similar manner while changing the total wet mass, number of engine, and type of engine/propellant as required.

For the LOX/LH_2 TMI stage, the RL10B-2 expander cycle engine was chosen, which is currently used on the second stage of the Delta IV heavy launch vehicle. Each RL10B-2 engine provides approximately 110 kN of thrust and is capable of a specific impulse of 462 seconds using a fuel-oxidizer mixture ratio of 5.88. Each engine has a mass of approximately 300 kg. The exact engine parameters used for this study are shown in Table 2.

Table 1: Procedure for modeling custom TMI stages

Step #	Description	Comment
Step 1.	Choose the number and type of engines	Choice based on desired thrust-to-weight ratio
Step 2.	Estimate the mass of usable propellant available	Value will be iterated to determine amount of required propellant for a desired total stage wet mass
Step 3.	Size oxidizer and fuel tanks based on propellant required	Single oxidizer and single fuel tank with common end cap assumed
Step 4.	Size pressurization system based on propellant tank characteristics	Sized using the ideal gas law
Step 5.	Add over head for feed system and margin	Added to all calculated values to determine total stage wet mass
Step 6.	Use the Goal Seek function to iteratively change the propellant required until the desired total stage wet mass is reached	Propellant tanks and pressurization system values must both be iterated

Table 2: RL-10B-2 engine parameters used for this study

Parameter	Value
Thrust per engine, N	110100
Mass per engine, kg	301
Specific impulse, s	462
Mixture ratio, -	5.88

The number of engines used on each propulsion stage was determined by selecting a minimal thrust-to-weight ratio for the vehicle stack, which would occur at the beginning of each stage's burn. It was assumed that a minimal value of 0.2 was required to counter gravity losses and other effects. Figure 8 shows the thrust-to-weight ratio of the vehicle stacks at the beginning of the burns for stages with various numbers of engines. The green vertical line shows the vehicle stack of one launch vehicle (110 mt), the orange vertical line shows the vehicle stack of two launch vehicles (220 mt), and the red vertical line shows the vehicle stage for three launch vehicles (330 mt). The black horizontal line shows that for a thrust-to-weight ratio of 0.2, the number of engines required for a stage pushing each stack is 2, 4, or 6, respectively.

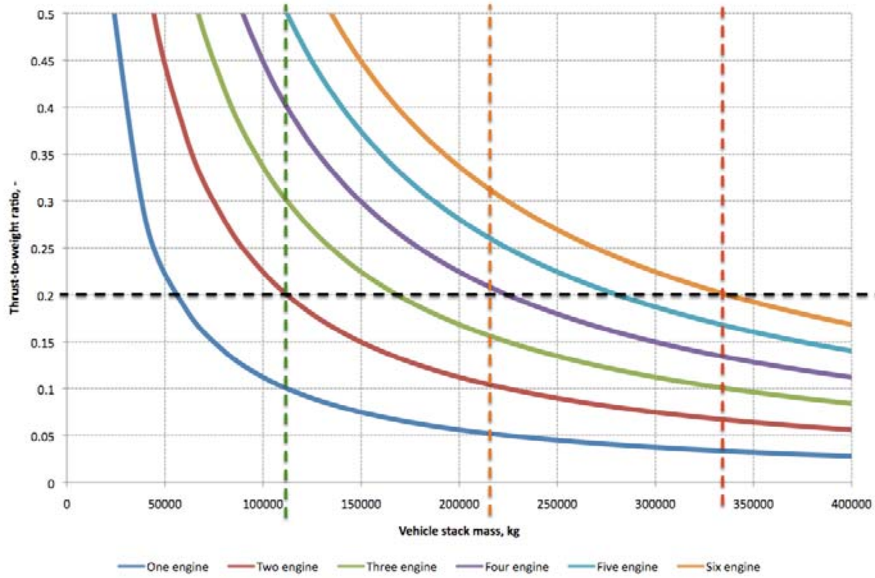

Figure 8: Thrust-to-weight versus vehicle stack mass for various numbers of RL-10B-2 engines

The first step to sizing the propellant tanks was to estimate the amount of usable propellant. This was done using a structural mass fraction, as shown in Equation 1, to determine the amount of propellant for a stage with a total mass of 110 mt. A structural mass fraction of 0.2 was assumed for the initial estimation, which equates to 91.7 mt of usable propellant.

$$\text{Structural mass fraction} = \frac{\text{Stage inert mass}}{\text{Usable propellant}} = \frac{\text{Total stage mass - usable propellant}}{\text{Usable propellant}} \quad \textbf{Equation (1)}$$

To account for the propellant trapped in the propellant tanks at the end of the burn, three percent was added to the usable propellant to calculate the total propellant required. Using the mixture ratio of the RL10B-2 engine and the densities for liquid hydrogen and liquid oxygen, the masses and volumes of the total amount of oxidizer and fuel were calculated. These volumes were increased by three percent to account for unusable volume inside the respective tanks to produce the total volume that the fuel and oxidizer tanks required.

A single propellant tank was assumed for the oxidizer and for the fuel. The tanks were modeled as cylinders with elliptical end caps and it was assumed that the tanks would share a common bulkhead to reduce the stage's overall structural mass. The propellant tanks were assumed to be an integral part of the overall stage structure and were given a diameter of 6.6 m based on launch vehicle dimensions. Using equations from Ref. 10, and assuming a maximum operating pressure of 300 kPa based on the use of RL-10B-2 engines, and assuming the tank walls were made up of a Kevlar material, the size, wall thickness and overall mass of the oxidizer tank were calculated. Table 3 shows the specifications used for the wall material. The mass of 10 mm of MLI blanket, wrapped around the outer part of the tank, was added to the mass of the tank. The fuel tank was sized using the same methodology, but the mass of only one end cap was included in the final mass budget to avoid double bookkeeping of the common bulkhead.

Table 3: Assumed specifications for Kevlar

Parameter	**Value**
Density, kg/m^3	1470
Ultimate strength, Pa	3.45E+09
Factor of safety, -	1.5
Maximum allowable operational stress, Pa	2.30E+09
Modulus of elasticity, Pa	1.79E+11
Poisson's ratio, -	0.36

The pressurization system for the TMI stage was based on a helium-gas blow down system. Using the ideal gas law (shown in Equation 2), the properties of helium (shown in Table 4) and the maximum operating pressure of the propellant tanks (300 kPa), the required initial volume and mass of helium was determined. The initial pressure and temperature of the helium was assumed to be 21 MPa and 300 K, respectively. The final pressure of the system was assumed to be the maximum operating pressure of the propellant tanks. In order to calculate the initial volume using the ideal gas law, the final volume had to be estimated, which was done as the volume of the empty propellant tanks. This value was then iterated to determine the initial volume.

$$\frac{p_i \times V_i}{T_i} = \frac{p_f \times V_f}{T_f} \quad \textbf{Equation 2}$$

The helium was stored in four spherical tanks made of the same Kevlar material as the propellant tanks. A factor of safety of 1.5 was assumed in sizing the wall thickness for the tanks.

Table 4: Properties of the propulsion stage pressurization system

Parameter	**Value**
Pressurant Gas Type	Helium
Pressurant Isentropic Parameter	1.66
Pressurant Molecular Mass, kg/mol	4.003
Pressurant Gas Constant, J/(kg*K)	2077.07
Pressurant Initial Pressure, Pa	21,000,000
Pressurant Initial Temperature, K	300

The proceeding steps provided mass estimates for the engines, propellant tanks, and pressurant tanks, as well as for the usable and unusable propellant, and the pressurant. The mass of the feed system was estimated to be 10% of the mass of the propellant and pressurant tanks. It was assumed that each stage would require a docking adapter, which was estimated to be 5,000 kg. For margin, 20% was added to all dry masses, including the feed system and docking adapter. Finally, another 2,500 kg was added to account for the mass of the launch vehicle adapter.

All of these masses were summed to determine the gross lift-off mass of the stage before the process went through several iterations to match the gross lift-off mass to the desired amount (either 110 mt or 40 mt). Each iteration involved using Microsoft Excel's Goal Seek function to alter the amount of usable propellant estimated, which is turn required another Goal Seek function to be performed to calculate the required amount of helium pressurant, this process was repeated until the desired total mass was achieved. Table 5 shows a mass breakdown for the 6 engine LOX/LH_2 TMI stage.

Table 5: Mass breakdown for the 6-engine LOX/LH_2 propulsion stage

Parameter	**Mass, kg**
Engines	1806
Propellant tanks	552
Pressurant tanks	482
Feed system	103
Docking adapter	5000
Dry mass	7943
20% dry mass growth allowance	1589
Dry mass with growth allowance	9532
Pressurant mass	791
Unusable propellant mass	2830
Inert mass at time of burn	13154
Usable propellant mass	94346
Total mass in orbit	107500
Launch vehicle adapter	2500
Gross lift-off mass	110000

For the Ares V EDS, a burnout mass of 22000 kg and a specific impulse in vacuum of 448 s was assumed. As mentioned above, the remaining propellant as a function of Ares V payload mass to a 400 km LEO was calculated according to the relationship shown in Figure 7. For the cases in which the EDS was used during Earth launch and subsequently as a TMI stage, a 2000 kg docking adapter was assumed to be transported to the 400 km LEO on top of the EDS.

The TMI mass injection capabilities of the options described in Section II are shown in Figure 9 and Figure 10 as a function of percentage of propellant boiled off during Earth-orbit loitering. Each diagram represents capabilities for a different TMI delta-v requirement (4000 m/s and 4100 m/s). From the diagrams it is apparent that Option 4.1 is uninteresting because it offer significantly degraded performance compared to Option 3.1 / 3.2 which also use 2 Ares V launches per injected payload; this indicates that the development of a small custom TMI stage is worthwhile from a performance perspective. It is also apparent that Option 4.2 is uninteresting when compared to Option 1 and Option 2 due to significantly degraded performance. This indicates that from a performance perspective the development of a custom large TMI stage (as alternative to the small custom TMI stage) is worthwhile. Option 1, however, does not much offer significant performance benefit compared to Option 2 despite being more costly due to the need to produce a 2nd custom large TMI stage; it is therefore also regarded as uninteresting. This leaves us with Option 2, Option 3.1 / 3.2 as the interesting chemical propulsion concepts, in addition to Option 5 (the nuclear thermal reference concept).

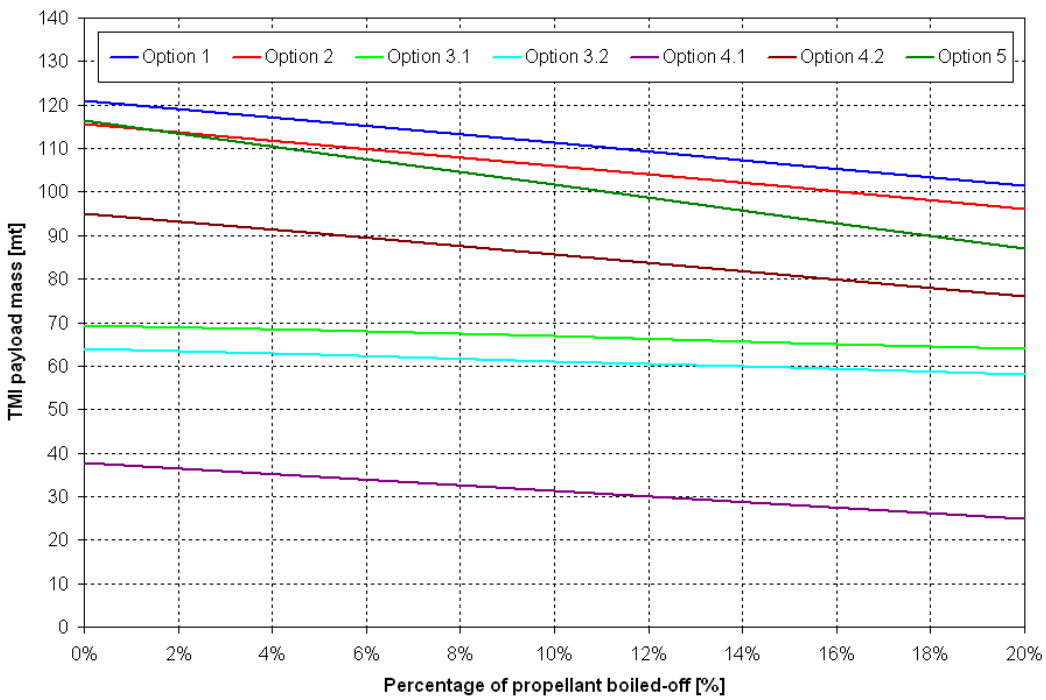

Figure 9: Overview of trans-Mars injected payload capabilities for the Earth departure options considered as a function of propellant boil-off, 4000 m/s TMI delta-v

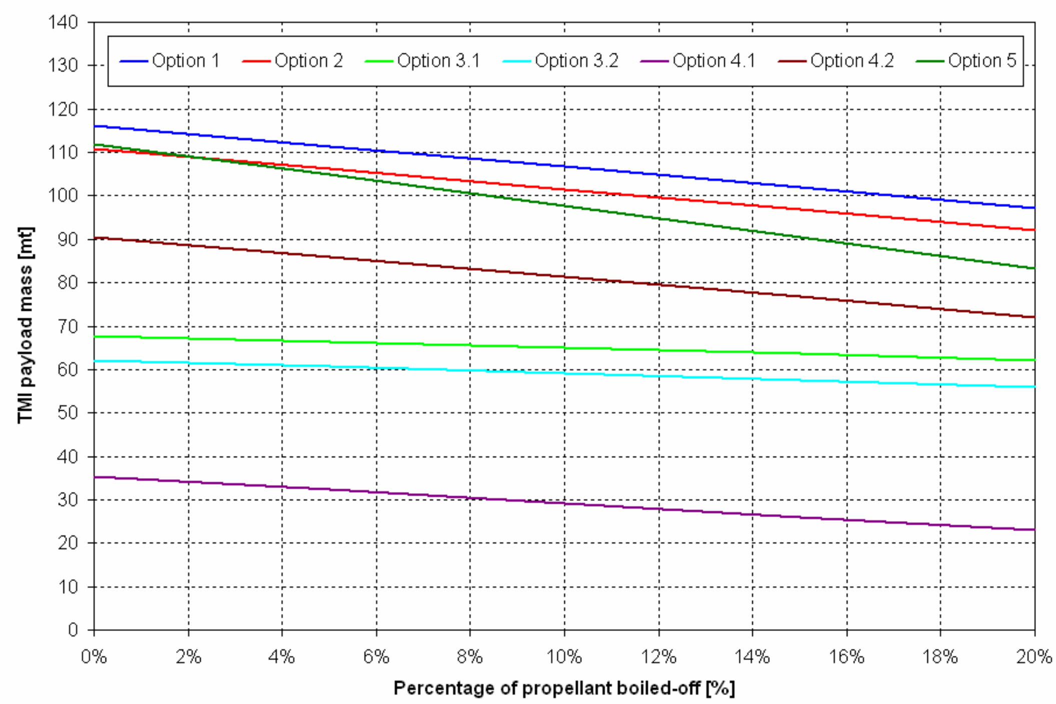

Figure 10: Overview of trans-Mars injected payload capabilities for the Earth departure options considered as a function of propellant boil-off, 4100 m/s TMI delta-v

IV. Discussion of Results

In the previous section, results from a quantitative analysis of the different Earth departure architecture options were presented for each architecture individually. In this section, we carry out a comparative analysis of the different options. To that end, it is necessary to choose a base for the comparison; in this case we choose mass injected on a trans-Mars trajectory (or TMI mass) as the metric for comparison. To establish a range of possible TMI mass requirements, recent human Mars mission designs were reviewed; Table 6 shows associated mass breakdowns.

Table 6: Overview of TMI mass requirements for selected human Mars mission designs[3,4,8,9]

Mission architecture	Vehicle 1	Vehicle 2	Vehicle 3	Total TMI mass [mt]
NASA Mars DRM 1.0	101.8	98.4	89.9	290.1
NASA Mars DRM 3.0	74.1	66.1	60.9	201.1
Three-Magnum split mission	72.2	39.0	0.0	111.2
Mars Direct (Zubrin)	40.0	40.0	0.0	80.0
Draper/MIT CE&R	125.0	109.0	116.0	350.0

Based on the TMI injection capabilities calculated in Section III, we can calculate how many Ares V launches would be required for a mission architecture as a function of TMI mass requirement. Figure 11 shows the results from this calculation for the interesting architectures identified in Section III over a range of likely TMI mass requirements. The mission masses from Table 6 are shown as well.

In addition to the interesting architectures identified above, a hybrid between Option 2 and Option 3 is shown as well: this approach would require the development of two dedicated TMI stages, one sized for a full Ares V launch, and one sized to be launched with an aeroshell. In addition, it may also be necessary to design two custom aeroshells. The advantage of this hybrid would be that, in certain cases, it could reduce the number of launches required compared to Option 2 and Option 3.

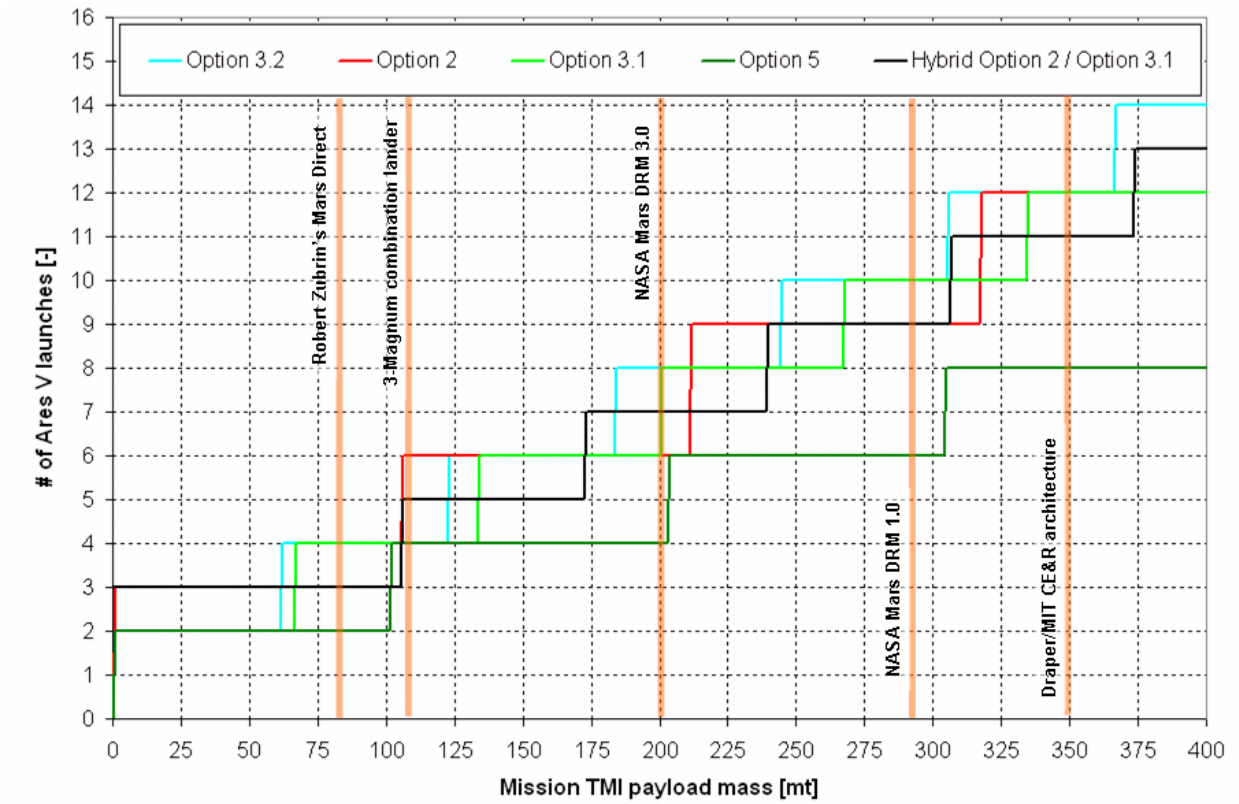

Figure 11: Overview of Ares V launch requirements as a function of TMI payload requirement; proposed designs shown for reference. Assumptions: TMI delta-v of 4000 m/s and 10% propellant boil-off.

The results in Figure 11 indicate that the chemical Earth departure architectures result in a 0-2 Ares V launch overhead for TMI masses below 100 mt (0-100%), a 0-3 Ares V launch overhead for TMI masses between 100 – 200 mt (0-75%), a 1-4 Ares V launch overhead for TMI masses between 100 – 200 mt (16-66%), and a 1-4 Ares V launch overhead for TMI masses between 100 – 200 mt (12.5-50%). While the hybrid option can offer a launch reduction of one Ares V launch, this advantage only applies to a relatively narrow mass range and therefore does not appear to be worth the significant added cost of developing two TMI stages and aeroshells.

For the likely conservative mass range of 200 – 300 mt, the likely overhead will be 2-3 Ares V launches, i.e. 33–50% more launches, as well as an added marginal cost for the production of 2-3 Ares V launch vehicles and an additional TMI stage. The nuclear thermal option, however, will have a higher fixed recurring cost for maintaining the capability to build and launch nuclear thermal propulsion stages than for the chemical TMI stages, and these stages will also be more expensive in marginal cost. It is therefore questionable whether the nuclear thermal option will actually be cheaper from a recurring cost perspective. It is safe to assume that nuclear thermal propulsion would be more costly from a DDT&E perspective.

Another important factor for comparing nuclear thermal and chemical propulsion for Earth departure is the ground operations impact. To that end, a simple high-level characterization of the ground operations requirements for the different options has been carried out for a 220 mt TMI mass and a 300 mt TMI mass. Also included in the analysis was a space shuttle operations scenario which featured 6 (or more) flights per year; this was achieved for a sustained period of time between 1990–1997[11]. The metrics considered are the overall number of launches for cargo and crew, the number of SRB segments that need to be processed, the number of major propulsion stages that need to be processed (this includes the external tank for shuttle missions), as well as the number of major payloads that need to be processed (this includes aeroshells with payloads, as well as shuttle orbiters with payloads), all over a period of 26 months (i.e. the time interval between Earth-Mars transportation windows).

Table 7 shows the results of this high-level analysis for 220 mt of TMI mass requirement. The number of launches is lower in all cases than for the 6-flight per year shuttle program, as is the number of SRB segments that need to be processed. The processing requirements for major propulsion stages are somewhat increased compared to the shuttle scenario; however, the number of major payloads is significantly reduced. As payloads would be expected to be more difficult to process than propulsion stages (with the possible exception of nuclear thermal propulsion stages), the overall ground processing requirements for a human Mars mission with 220 mt TMI mass would appear to be lower than those for a 6-flight per year shuttle scenario.

Table 8 shows the results from a comparable analysis for a 300 mt TMI mass requirements. The Option 3 values are the only that change: the number of SRB segments to process now exceeds the processing requirements for the 6-flight shuttle scenario. The number of launches is still lower, as is the number of major payloads to process. The number of major propulsion stages to process is significantly higher than for the shuttle scenario; however, the combined number of major payloads and major propulsion stages to process is still comparable to those for the shuttle scenario (31 vs. 26). If Option 2 is used, the ground processing requirements are identical to those for a 200 mt TMI mass scenario.

Table 7: High-level comparison of ground operations for a 220 mt TMI mass human Mars mission for different Earth departure options; also included: a Space Shuttle operations scenario with 6 flights per year.

Scenario	Option 2	Option 3.1 / Option 3.2	Option 5	Shuttle, 6 flights per year
# of launches	10 (9 Ares V + 1 Ares I)	9 (8 Ares V and 1 Ares I)	7 (6 Ares V + 1 Ares I)	13
# of SRB segments processed	95	85	65	104
# of major propulsion stages processed	19 (9 core stages, 9 EDS, 3 TMI stages, 1 Ares I upper stage)	21 (8 core stages, 8 EDS, 4 TMI stages, 1 Ares I upper stage)	16 (6 core stages, 6 EDS, 3 nuclear TMI stages, 1 Ares I upper stage)	13 external tanks
# of payloads processed	4 (3 aeroshells and 1 CEV)	5 (4 aeroshells and 1 CEV)	4 (3 aeroshells and 1 CEV)	13 shuttles + payloads

Table 8: High-level comparison of ground operations for a 300 mt TMI mass human Mars mission for different Earth departure options; also included: a Space Shuttle operations scenario with 6 flights per year.

Scenario	Option 2	Option 3.1 / Option 3.2	Option 5	Shuttle, 6 flights per year
# of launches	10 (9 Ares V + 1 Ares I)	11 (10 Ares V and 1 Ares I)	7 (6 Ares V + 1 Ares I)	13
# of SRB segments processed	95	105	65	104
# of major propulsion stages processed	19 (9 core stages, 9 EDS, 3 TMI stages, 1 Ares I upper stage)	26 (10 core stages, 10 EDS, 5 TMI stages, 1 Ares I upper stage)	16 (6 core stages, 6 EDS, 3 nuclear TMI stages, 1 Ares I upper stage)	13 external tanks
# of payloads processed	4 (3 aeroshells and 1 CEV)	5 (4 aeroshells and 1 CEV)	4 (3 aeroshells and 1 CEV)	13 shuttles + payloads

It should be noted that this analysis of ground processing requirements is not intended as a detailed assessment of ground operations, but rather an assessment of potential show-stoppers. At this point, it appears that a human Mars mission architecture with a TMI mass between 200 – 300 mt using chemical propulsion would be feasible from a ground operations perspective.

V. Conclusion and Future Work

The discussion of the analysis above yielded a number of interesting insights into Earth departure strategies for human Mars missions which are summarized here in the form of key findings:

- A number of interesting and feasible chemical Earth departure concepts have been characterized including ones with 2 and ones with 3 Ares V launches per payload inserted towards Mars. For the likely TMI mass requirements range of 200–300 mt per human Mars mission opportunity, these Earth departure strategies would require 8-9 Ares V launches.

- It is worthwhile to reuse an EDS upper stage outfitted with a docking adapter as a chemical Earth departure stage (this would use the remaining propellant in the EDS for the first TMI burn).
- For the likely range of TMI mass requirements (200–300 mt), chemical Earth departure architectures exhibit a launch overhead of 2-3 Ares V launches per opportunity (26 months); they also require up to 1 additional TMI stage and 1 additional aeroshell (for Option 3).
- These overheads needs to be compared to the increased cost of developing and maintaining a nuclear thermal propulsion capability as well as the increased unit cost of nuclear thermal propulsion stages. This analysis should also include an assessment of the policy robustness and operational risk of a nuclear thermal propulsion enterprise.
- Based on a high-level assessment, ground operations requirements for a human Mars mission based on chemical Earth departure propulsion appear to be comparable to those of a shuttle operations scenario with 6 launches per year. A mission frequency of 6 flights per year has been achieved or exceeded between 1990-1997.

Opportunities for future work include expansion of the analysis to other architectural options such as advanced propulsion in Mars vicinity, increase in modeling fidelity and design resolution for the propulsion stages, explicit consideration of Mars aerocapture and entry in the analysis, a more detailed assessment of ground operations in the KSC infrastructure for each option, and a more detailed assessment of the impact boil-off on the different options.

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