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On-Orbit Propellant Resupply Options for Mars Exploration Architectures

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ABSTRACT

A report detailing recommendations for a transportation architecture and a roadmap for U.S. exploration of the Moon and Mars was released by the NASA Exploration Systems Architecture Study (ESAS) in November 2005. In addition to defining launch vehicles and various aspects of a lunar exploration architecture, the report also elaborated on the extent of commercial involvement in future NASA activities, such as cargo transportation to the International Space Station. Another potential area of commercial involvement under investigation is the delivery of cryogenic propellants to low-Earth orbit (LEO) to refuel NASA assets as well as commercial assets on orbit. The ability to resupply propellant to various architecture elements on-orbit opens a host of new possibilities with respect to a Mars transportation architecture – first and foremost being the ability to conduct a Martian exploration campaign without the development of expensive propulsion systems such as nuclear thermal propulsion. In-space propellant transfer in the form of an orbiting propellant depot would affect the sizing and configuration of some currently proposed vehicles such as the Earth Departure Stage (EDS) and the Mars Transit Vehicle (MTV). In addition, it would influence the overall affordability and sustainability of a long-term Mars exploration campaign. To assess these consequences, these vehicles and their various stages are modeled to approximate the ESAS performance figures using a combination of analogous systems and physics-based simulation. Well established modeling tools -- such as POST for trajectory optimization, APAS for aerodynamics, NAFCOM for cost modeling, and Monte Carlo analysis for technology advancement uncertainty -- are used to perform these analyses. To gain a more complete view of the effects of an on-orbit propellant refueling capability, a reference Mars mission is developed and compared to an equivalent mission without refueling capability. Finally, the possibility of propellant resupply in Mars orbit is also discussed along with its implications on the sustainability of a long-term Mars exploration architecture.

NOMENCLATURE

C3	Excess hyperbolic energy squared (km^2/s^2)
DDT&E	Design, Development, Testing, and Evaluation
DRM	Design Reference Mission
EDL	Entry, Descent, and Landing
EDS	Earth Departure Vehicle

EOR	Earth-Orbit Rendezvous
ERV	Earth Return Vehicle
ESAS	Exploration Study Architecture Study
FY	Fiscal Year
IMLEO	Initial Mass in Low Earth Orbit
ISS	International Space Station
JSC	Johnson Space Center
KSC	Kennedy Space Center
LEO	Low Earth Orbit
LH2	Liquid Hydrogen
LOX	Liquid Oxygen
MOI	Mars Orbit Insertion
MPLM	Multi-Purpose Logistics Module
MT	Metric Tons
NAFCOM	NASA / Air Force Cost Model
NASA	National Aeronautics and Space Administration
NTR	Nuclear Thermal Rocket
O/F	Oxidizer-to-Fuel Ratio
POST	Program to Optimize Simulated Trajectories
PRM	Propellant Resupply Module
STS	Shuttle Transportation System
SVLCM	Spacecraft / Vehicle Level Cost Model
TCM	Trajectory Correction Maneuver
TFU	Theoretical First Unit
TMI	Trans-Mars Injection
VAB	Vehicle Assembly Building
VSE	Vision for Space Exploration

INTRODUCTION

The ambitious goal of the return of humans to the Moon and the eventual transportation of humans to Mars was put forth in the President's Vision for Space Exploration (VSE)[1]. To develop an idea as to how this could be accomplished, the Exploration Systems Architecture Study (ESAS) was performed, resulting in first-cuts at the recently labeled Orion crew vehicle and Ares I and V launch vehicles. The launch vehicles have undergone some changes since the ESAS Final Report[2], but the overall mission of using the vehicles to place humans and their associated cargo safely on the Moon has not changed. Since the development of two new launch vehicles will likely be an arduous and expensive task, one must look at how the proposed vehicles will fit into a human-Mars campaign.

FRAMING THE PROBLEM

Architecture Elements

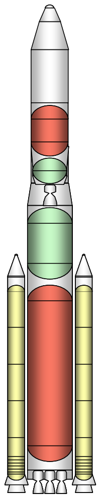
To establish a point of reference, the Design Reference Mission (DRM) v3.0[3] developed by NASA JSC is used as a baseline Martian architecture to manifest the various launches and trans-Mars injection (TMI) stages. This architecture is summarized in Table 1.

Payload	Mass	
	lbm	kg
Earth Return Vehicle (ERV)	163,301	74,072
Cargo Lander	145,600	66,043
Crew Lander	134,054	60,806
NTR TMI Stage (max)	168,874	76,600

Table 1: DRM payloads and nuclear thermal rocket (NTR) TMI stage[3].

Given the large payloads outlined by the DRM, the primary focus of this study will be on using the proposed heavy lift launch vehicle, the Ares V, and its Earth Departure

Stage (EDS). The assumed characteristics of the Ares V are detailed in Table 2.



Boosters	2x 5-seg RSRM
Booster Propellant	1,400 klbm each
Core Engines	5x RS-68
Isp (vac)	410 s
Thrust (vac)	745 klbf each
Area Raito	21.5
Weight	14.6 klbm each
Core Propellant	3,000 klbm
Core Diameter	33 ft
EDS Engine	1x J-2X
Isp (vac)	448 s
Thrust (vac)	274 klbf
Area Ratio	40.0
Weight	3.0 klbm
EDS Propellant	508 klbm
EDS Diameter	27.5 ft
Fairing Weight	13 klbm
Delivery Orbit	30 x 160 nmi

Table 2: Ares V configuration and performance specifications.

The DRM baselines a nuclear thermal rocket (NTR) TMI stage that is launched separately from each payload. However, for this study, *the EDS is used as the TMI stage in lieu of the DRM NTR stage***.** As designed, the Ares V uses the EDS as both an upper stage, to insert the payload into a parking orbit in LEO, and as an Earth departure stage, to inject the payload into the proper transfer orbit. However, a partially used EDS does not have the capability to inject all of the DRM payloads into a Mars transfer orbit. One way of remedying this problem is to refuel the EDS in LEO prior to the injection burn using a pre-positioned asset, such as a propellant depot similar to Figure 1. There are several unique advantages to the development and implementation of a propellant depot:

1. Commercial launch services could be utilized to re-supply the depot
2. The depot could be used to re-fuel other manned or unmanned spacecraft
3. Enabling depot technologies, such as low-boiloff and autonomous rendezvous & docking capabilities, could be used to enhance other planetary or Earth science spacecraft.

Figure 1: OASIS cryogenic propellant depot[5].

To fully understand the usefulness of a refueled EDS, the Earth departure C3 from the ESAS reference 30 x 160 nmi orbit is plotted versus the injected payload in Figure 2. This plot assumes that the payload mass indicated along the abscissa is launched from KSC into a 28.5^{o} inclination and that the launched payload is the same as the injected payload (i.e. no Earth-orbit rendezvous to add additional payload to the departure stage). The minimum departure C3 indicated is the minimum Earth departure C3 that occurs within the 2030-2040 timespan.

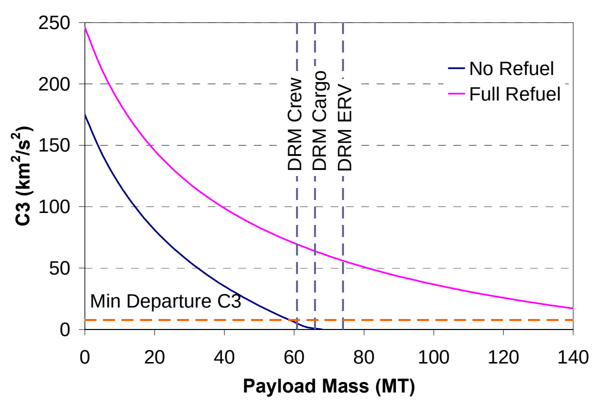


Figure 2: Available Earth departure C3 versus payload mass for a partially expended EDS and a completely fuelled EDS.

Clearly, a fully fuelled EDS has much higher capability – but that also means a propellant depot would have to sustain over 230 MT of propellant (a complete EDS refuel) on-orbit, which could be prohibitive depending on the propellant boiloff rate and capability of the commercial re-supply missions. Thus, this study focused on the *minimum* amount of re-fuel needed to accomplish a given mission – i.e. the DRM mission – with the goal of providing the minimum (and subsequently the most affordable) depot architecture needed to send humans to Mars.

Defining the Trade Studies

Three separate architectures are investigated in this study given the above architectural elements. These trades are compared against one another on a performance and cost basis.

1. Aerocaptured payloads – the DRM baselines aerocapture for all three payloads into an elliptical parking orbit at Mars. The only change in this trade is the replacement of the NTR stage with an EDS and a propellant depot in LEO.
2. Propulsively captured payloads – aerocapture could require large development resources. If the EDS can perform a propulsive capture at Mars after being refuelled in LEO, then aerocapture may not necessarily be required.
3. NTR stage – it is not entirely clear if the use of the EDS and a propellant depot is less expensive than using NTR, as the DRM originally suggests. Thus, the NTR is independently analyzed and compared against the first two architectures.

Interplanetary Trajectory Optimization

Slight modification of the DRM is necessary to put it in the scope of the VSE. The DRM assumes the first Mars departure of its Earth Return Vehicle (ERV) and Cargo Lander would occur on the 2011 conjunction-class opportunity, with the crew subsequently launched on the 2014 opportunity. However, given the timeline of the Space Shuttle retirement and lunar exploration program[2], a more realistic timeframe for the first Mars mission would range from 2030 to 2040. Optimized trajectories from the Earth to Mars within this timeframe were analyzed using JAQAR's Swing-by Calculator[4] resulting in three different trajectories per opportunity:

1. Minimum Earth Departure C3

Earth Departure Date	Departure C3 (km^2/s^2)	Mars Arrival Date	Arrival C3 (km^2/s^2)	Transit Time (days)
02/24/31	8.237	01/11/32	31.164	321
04/28/33	7.780	01/27/34	19.157	274
06/26/35	10.199	01/04/36	7.273	192
09/06/37	14.848	10/06/38	11.148	395
09/27/39	12.176	09/21/40	7.264	360
10/20/41	9.813	09/01/42	6.183	316
11/15/43	8.969	09/18/44	7.837	308

Table 3: Minimum Earth departure trajectories from 2030-2043.

2. Minimum Mars Arrival C3

Earth Departure Date	Departure C3 (km^2/s^2)	Mars Arrival Date	Arrival C3 (km^2/s^2)	Transit Time (days)
12/13/30	12.397	09/25/31	11.870	286
04/20/33	9.305	11/05/33	10.962	199
07/09/35	11.910	01/24/36	6.762	199
09/24/37	32.612	06/01/38	5.838	250
10/25/39	28.739	07/17/40	5.576	266
10/22/41	9.859	09/04/42	6.167	317
11/14/43	8.974	09/14/44	7.801	305

Table 4: Minimum Mars arrival trajectories from 2030-2043.

3. Minimum Total C3

Earth Departure Date	Departure C3 (km^2/s^2)	Mars Arrival Date	Arrival C3 (km^2/s^2)	Transit Time (days)
12/28/30	10.445	10/08/31	12.372	284
04/15/33	8.996	10/31/33	11.072	199
06/25/35	10.274	01/12/36	6.989	201
08/23/37	15.754	08/18/38	8.338	360
09/22/39	12.444	08/30/40	6.281	343
10/20/41	9.814	09/03/42	6.171	318
11/14/43	8.974	09/14/44	7.801	305

Table 5: Minimum total energy trajectories from 2030-2043.

Table 3 serves as a minimum propulsive requirement to get to Mars during the given opportunity. Table 4 is included as it places a lower bound on the Mars capture and entry, descent, and landing (EDL) requirements of the payload. Table 5 attempts to balance departure and arrival energy in order to provide a more practical solution than the previous two.

ANALYSIS AND RESULTS

Ares V Ascent Trajectory

The Program to Optimize Simulated Trajectories (POST) was used to simulate ascent of the DRM payloads given the Ares

V configuration detailed above. This simulation essentially provided the remaining EDS capability once the given payload achieved orbit. Figure 3 is an ascent trajectory obtained for the DRM Cargo launch with an 800 psf maximum dynamic pressure (max-q) constraint imposed on the trajectory. This constraint was used assuming at least some technology advancement above the current Space Shuttle max-q of 650 psf and the largest max-q witnessed by the Saturn V at 777 psf[6].

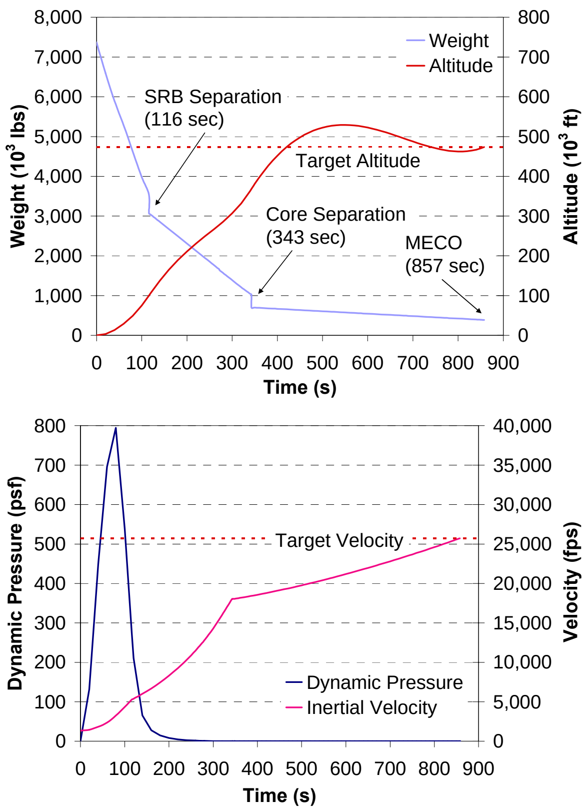


Figure 3: Ares V ascent trajectory for DRM Cargo payload to 30 x 160 nmi x 28.5^{o} orbit.

From the trajectory above, it is noticeable that the EDS lofts above the desired altitude as part of the optimized trajectory. This lofting action is caused by the particularly low thrust-to-weight ratio (0.38 to 0.40) of the modified EDS design which only uses one J-2X engine. The loft itself is a result of the optimization in which the EDS attempts to minimize drag losses in order to accelerate the payload to the desired velocity. Increasing the number of engines on the EDS would eliminate the loft and create a much smoother trajectory.

EDS Capability for Aerocapture Payload

The DRM v3.0 mission assumes that each payload uses aerocapture technology to decelerate into an elliptical Mars orbit upon arrival (Figure 4). Additionally, it assumes that each of the nuclear TMI stages is used only once and is discarded after the TMI maneuver is performed. All of the trajectory correction maneuvers (TCMs), orbit adjust and trim maneuvers after aerocapture, and de-orbit burns for payload entry are performed by the propulsive descent system onboard each payload. Finally, the nuclear TMI stages in the DRM are launched separately from the payloads, requiring six heavy launches per mission as well as three Earth-orbit rendezvous (EORs).

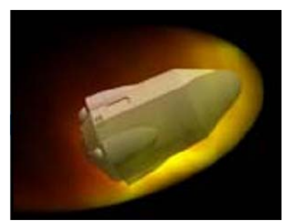


Figure 4: DRM triconic aerocapture vehicle with estimated L/D of 0.6[3].

In this study, each payload is launched on top of the Ares V / EDS stack, thus requiring three heavy launches per mission plus the necessary depot resupply launches (explained in detail later). Although stacking the DRM payloads on top of the EDS creates a very tall launch vehicle, current figures for the Ares V core and EDS heights coupled with the cited payload dimensions in the DRM v3.0 report do not violate the maximum door height of the Vehicle Assembly Building (VAB) at KSC, as shown in Table 3. As a reference point, the Saturn V measured approximately 364 ft in height and the Shuttle Transportation System (STS) measures 184 ft in height. It should be noted that a flight stability analysis was not performed on this configuration to ensure a stable ascent vehicle.

Element	Height	
	Element	Cumulative
Ares V Core	212.6 ft	212.6 ft
EDS	73.4 ft	286.0 ft
Payload + Shroud (max)	91.9 ft	377.9 ft
VAB Doors (max)[7]		456 ft
Clearance		78.1 ft

Table 6: Estimated height of the Ares V stack with DRM payload compared to the maximum height of the VAB high bay doors.

In addition to the analysis performed in the DRM concerning the feasibility of a large aerocaptured Mars payload, an additional source was used to confirm that the baseline DRM aerocapture vehicle had sufficient lift-to-drag (L/D) to aerocapture at the entry velocities witnessed throughout the 2030-2040 timeframe (5.47–7.45 km/s). According to Braun and Powell[8], a vehicle would require an L/D of 0.5 to capture within the velocity range of 6.0–8.5 km/s using atmospheric guidance control throughout the trajectory. Given that the DRM baseline aerocapture vehicle has an L/D of 0.6, it is assumed that the vehicle can aerocapture at velocities lower than 6.0 km/s as well as retain the capability to aerocapture at velocities up through 7.45 km/s with atmospheric guidance. The final capture orbit around Mars is assumed to be a highly elliptical 500 x 33,640 km elliptical orbit, resulting in a period of 25.67 hours – the same as the Martian day.

From the above C3 requirements and reference parking orbit, the required amount of propellant to initiate transfer to Mars is calculated. The ascent simulation performed in POST provided the amount of EDS propellant remaining after achieving orbit. The difference between the remaining EDS propellant and the amount required to initiate Mars transfer is the amount of propellant that needs to be replenished via the propellant depot. Figure 5 plots the amount of refuel propellant required for each DRM element at each opportunity. Three points for each payload at each opportunity translate into the minimum departure C3 (min refuel mass), minimum arrival C3 (max refuel mass), and minimum total C3 (mid refuel mass). When only one or two points are shown, refuel mass for the trajectories are approximately equal.

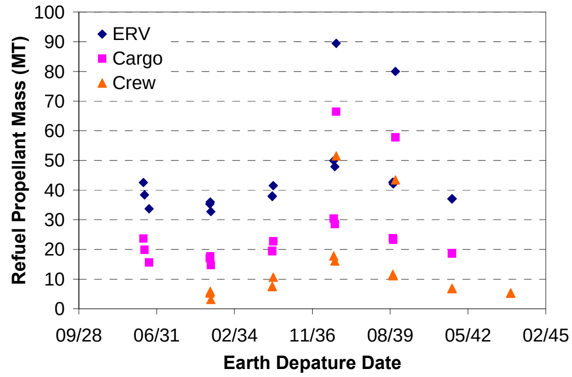


Figure 5: Required propellant resupply mass given payload aerocapture. The Crew payload is offset from its corresponding ERV and Cargo payload by one opportunity.

Figure 5 illustrates that a propellant depot can vary widely in capacity depending on the departure date and the payload mass. However, given the DRM aerocapture vehicle, only the minimum departure C3 case (minimum refuel propellant mass) need be considered.

Figure 6 illustrates the total refuel propellant required per mission (one mission equals one ERV and Cargo departure on one opportunity with the Crew departure one opportunity later) considering only the lowest Earth departure energy cases. The plot shows that, even for the least energetic mission to Mars, approximately 52.5 MT of propellant must be boosted into orbit over the course of two years. These figures do not include propellant boiled off from the depot while waiting for the EDS to launch and refuel.

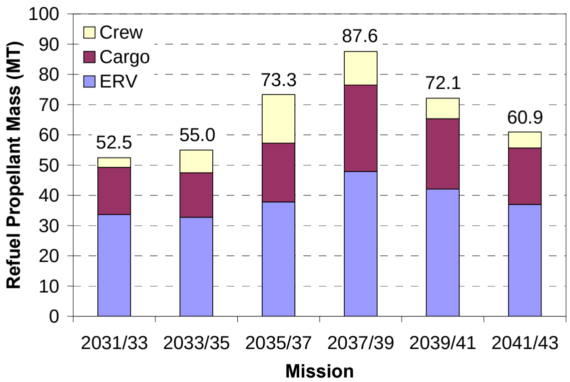


Figure 6: Total refuel propellant required for aerocaptured payloads assuming a minimum departure C3 trajectory.

EDS Capability for Propulsive Capture Payload

It should be noted that the refuel masses for the aerocapture cases do not completely fill the EDS – i.e. the EDS has additional capability if provided more propellant. More EDS propellant could be used as an alternative to aerocapture allowing the payload to be propulsively captured into a Mars parking orbit. This approach holds a set of distinct advantages:

1. Aerocapture has never been demonstrated and a more risky maneuver than propulsive capture.
2. Aerocapture requires additional hardware, increasing the mass of the vehicle. Additionally, it requires the entry vehicle to have a prescribed amount of L/D, which could lead to packaging constraints both inside the aeroshell and on top of the launch vehicle.
3. NASA may not have enough resources to support the simultaneous development of Mars surface hardware and descent capability along with aerocapture technology.

For these reasons, the concept of a propulsive capture should be investigated. To simulate payloads without aerocapture hardware, each vehicle mass was reduced by about 8%, resulting in the payload masses in Table 7. The DRM found that the heatshield required for aerocapture and entry of these large payloads resulted in about 16% of the total vehicle mass. It is assumed in this study that eliminating aerocapture would reduce the heatshield mass by about half.

Payload	Mass	
	lbm	kg
Earth Return Vehicle (ERV)	150,237	68,146
Cargo Lander	133,952	60,760
Crew Lander	123,330	55,942

Table 7: DRM payloads less aerocapture hardware.

The lighter payloads result in slightly more EDS capability once the launch stack reached LEO. However, since the EDS is not being discarded after the TMI burn, the descent stage cannot perform the TCMs during transit. Thus, for this case, the EDS performs the TMI burn, all TCMs, and the Mars-orbit insertion (MOI) burn before being discarded.

Since transit times to Mars can take anywhere from 190 to 400 days within the timespan investigated, propellant boiloff must be accounted for. Propellant boiloff rates are modelled as a constant percent of the current propellant mass lost per day. The boiloff values for liquid oxygen and liquid hydrogen were backed out of the ESAS final report and slightly improved to account for some technological advancement in cryogenic propellant storage, resulting in an LH2 boiloff rate of about 0.20% per day and a LOX boiloff rate of about 0.02% per day. Additionally, TCM propellant is estimated at 2% of the total EDS propellant mass.

Compiling the TMI, TCM, boiloff, and MOI propellant, a total required propellant for the mission is obtained. Using the ascent data produced by POST for the lighter payloads, the total amount of refuel propellant shown in Figure 7.

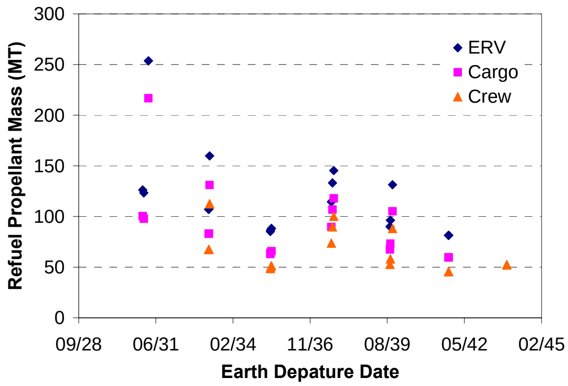


Figure 7: Required propellant resupply mass given propulsive payload capture. The Crew payload is offset from its corresponding ERV and Cargo payload by one opportunity.

Figure 7 is distinctively different from Figure 5 as the amount of refuel propellant is now strongly dependent on the Mars arrival C3 – higher arrival C3 not only means more MOI propellant, but it also means more propellant boiled off during transit. The minimum amount of refuel propellant, providing the smallest propellant depot, generally coincides with minimum total C3 trajectory. The refuel requirements for this trajectory are shown in Figure 8. These values also do not include propellant boiloff while loitering in the depot.

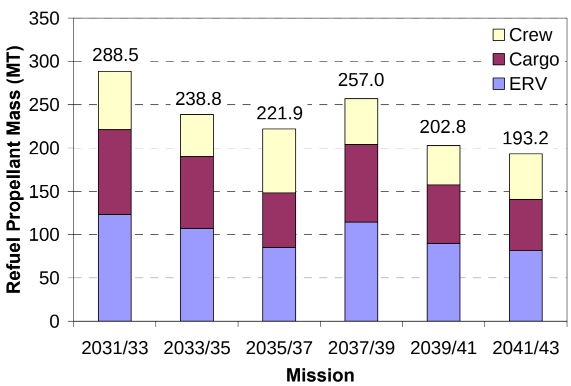


Figure 8: Total refuel propellant required for propulsively caputred payloads assuming a minimum total C3 trajectory.

For the propulsive option, the first mission opportunity in the 2031/33 timeframe is now the worst option, which is exactly opposite of the aerocapture results in Figure 6. However, even the most attractive propulsive mission requires over 193 MT of refuel propellant.

Based purely on these results, aerocapture requires a smaller depot and less total mass in LEO. Initial mass in low-Earth orbit (IMLEO) is generally a good indication as to how much a given architecture will cost. This rationale favors aerocapture above propulsive capture.

Propellant Depot Sizing

Since the amount of refuel propellant can vary quite widely depending on the opportunity and the associated payload mass, the worst-case propellant resupply from each scenario is used to size an appropriate depot. Each depot is sized using the scalable depot sizer developed by Street[9] assuming aluminium storage tanks with insulation but no cryocoolers (i.e. propellant will boiloff while the depot is in orbit). The propellant boiloff rates were not modified in the scalable depot sizer as these rates apply to LEO whereas the EDS boiloff rate applies to deep space.

Since the ERV and Cargo mission both leave Earth on the same opportunity, both must be launched and refuelled within a small window of time. This means that the depot must be sized to provide enough propellant for both departures. A 90-day window between the last commercial depot resupply and EDS refuel is assumed in sizing the depot.

For the aerocapture mission, the maximum amount of refuel propellant is 76.5 MT, which occurs during the 2037 departure of the ERV and Cargo payloads. In order to have that much propellant in the proper oxidizer-to-fuel (O/F) ratio after the assumed 90-day wait period, the orbiting propellant depot would be required to store 94.5 MT of propellant. Table 8 provides the pertinent details about the propellant depot. The depot's outer diameter was chosen such that the depot would be able to fit inside the Ares V payload fairing. However, the fairing would have to be lengthened slightly in order to contain the depot.

Depot Diameter	7.8 m
Depot Length	17.59 m
Dry Mass	13.6 MT
LH2 Storage Capacity	11.9 MT
LOX Storage Capacity	66.3 MT
O/F Ratio @ Capacity	5.6
Total Wet Mass	94.0 MT

Table 8: Propellant depot metrics for the aerocapture architecture.

For the propulsive capture mission, the maximum amount of refuel propellant is 221 MT, which occurs during the 2031 departure of the ERV and Cargo payloads. Table 9 outlines the depot required to support this mission. This depot is also sized to fit within the Ares V payload shroud. However, due to its length, the depot would have to be launched using two Ares Vs and assembled on orbit.

Depot Diameter	7.8 m
Depot Length	36.22 m
Dry Mass	28.9 MT
LH2 Storage Capacity	32.5 MT
LOX Storage Capacity	190.2 MT
O/F Ratio @ Capacity	5.8
Total Wet Mass	265.5 MT

Table 9: Propellant depot metrics for the propulsive capture architecture.

As stated previously, a propellant depot would likely use commercial launch services to replenish its stores when necessary. For extremely large amounts of propellant, the Ares V could be used to boost a larger resupply module. A Propellant Resupply Module (PRM) could be quite similar to the Russian Progress M spacecraft or the Italian Multi-Purpose Logistic Module (MPLM), both of which are used to ferry supplies to the International Space Station (ISS). Assuming a conservative structural mass fraction similar to the MPLM (m_{loaded}/m_{empty} = 0.31), a rough mass estimate for a PRM can be obtained.

Figure 9: Multi-Purpose Logistics Module (MPLM) is analogous to a Propellant Resupply Module (PRM) for depot resupply.

Using this sizing strategy, Table 10 details the propellant resupply capability of each of the Delta IV launch vehicles as well as their assumed launch costs.

Delta IV Variant	Capacity to LEO[10]	Refuel Capability	Launch Cost[11] FY06
Medium	9.1 MT	6.3 MT	$140M
Medium+ (4,2)	12.3 MT	8.5 MT	$155M
Medium+ (5,2)	10.6 MT	7.3 MT	$155M
Medium+ (5,4)	13.9 MT	9.6 MT	$171M
Heavy	21.9 MT	15.1 MT	$264M
Ares V	140.6 MT	97.0 MT	$563M

Table 10: Delta IV propellant resupply capability and assumed cost.

The cost of each PRM can be roughly obtained based purely on the estimated dry mass by using the Spacecraft/Vehicle Level Cost Model (SVLCM)[12] developed by the NASA JSC Cost Estimation group. The design, development, testing, and engineering (DDT&E) and theoretical first unit (TFU) can be obtained using an unmanned Earth orbital spacecraft analogy in the cost model. These costs are detailed in Table 11.

Module Size	Dry Mass	DDT&E FY06	TFU FY06
Medium	2.8 MT	$304.9M	$84.3M
Medium+ (4,2)	3.8 MT	$359.7M	$102.9M
Medium+ (5,2)	3.3 MT	$331.7M	$93.3M
Medium+ (5,4)	4.3 MT	$384.2M	$111.4M
Heavy	6.8 MT	$493.9M	$150.7M
Ares V	43.6 MT	$1,374M	$516.2M

Table 11: Development and unit cost estimates for the PRM associated with each launch vehicle.

Realistically, only one, potentially two, sizes of PRM would be fabricated and used to supply the depot. Since both options require a significant amount of propellant, the Delta

IV Heavy PRM and the Ares V PRM are traded in the following section.

Fixed Cost of Depot Architectures

Each of the propellant depot configurations were costed using NAFCOM 2004 and inflated to 2006 costs. Analogous systems used to generate the cost data included the Skylab Orbital Workshop, STS External Tank, Centaur-D upper stage, and Saturn IV-B upper stage. This resulted in the costs listed in Table 12. All costs are $M FY2006 USD. Launch costs assume one Ares V for the aerocapture depot and two Ares Vs for the propulsive depot. The launch cost for the Ares V was estimated at about 25% more than the average Shuttle launch cost ($450M). This cost increase from Shuttle is derived from the fact that the Ares V will be a much larger vehicle, requiring more processing and manpower to prepare and launch.

	Aerocapture	Propulsive	Difference
DDT&E	$1,418.8	$3,530.4	$2,111.6
TFU	$110.3	$240.5	$130.2
Launch	$563	$1,126	$563
Total	$2,029	$4,897	$2,868

Table 12: Development and unit costs for propellant depot options.

DDT&E includes the cost associate to mature the technologies associated with depot operations; Chato[13] describes some of these technologies in his paper.

The cost to develop, test, and reliably implement aerocapture technology for crewed vehicles also needs to be captured in order to accurately compare the two options. This sort of technology program would likely involve high-fidelity computer simulations, high-altitude testing, and potentially subscale and full-scale testing at Mars. Extensive testing and verification to “man-rate” an aerocapture vehicle would undoubtedly be an expensive venture. As a point of comparison, total costs to develop the X-38 crew return vehicle were estimated around $2.17B FY05[14]. According to the depot cost analysis above, there would be about a $2.87B difference between the hardware costs of the larger, propulsive capture depot and the smaller, aerocapture depot. Once again the SVLCM is used obtain a cost estimate for a man-rated aerocapture vehicle. Using manned spacecraft as the analogy and the DRM-cited dry mass of the aeroshell at 9.9 MT, a DDT&E cost of about $3B and a unit cost of $282M are obtained. The depot cost difference is virtually equivalent to the cost of developing and testing a new man-rated aerocapture vehicle.

Table 13 summarizes the total fixed costs for both depot options. The PRM cost is the DDT&E cost associated with a Delta IV Heavy-sized PRM; the aerocapture cost represents its associated DDT&E cost. All costs are in $M FY2006 USD.

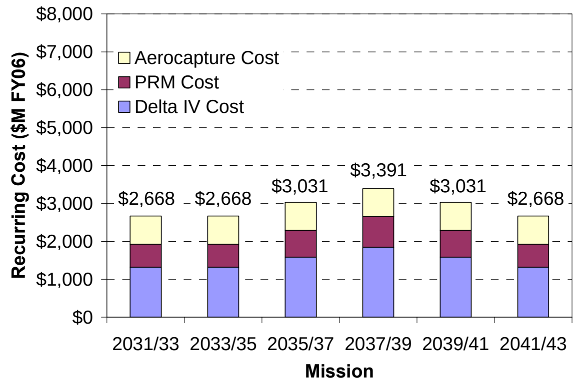
	Aerocapture	Propulsive	Difference
Depot	$2,092	$4,897	$2,868
PRM	$494	$494	$0
Aerocapture	$3,195	N/A	$3,195
Total	$5,781	$5,391	$327

Table 13: Total fixed costs for both depot architecture options.

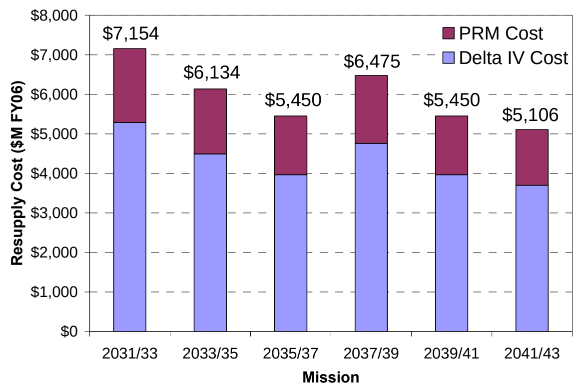
Recurring Cost of Depot Architectures

Since the two depot architectures are about equivalent in up-front development cost, the recurring cost of each architecture will be the deciding factor. In order to assess each mission on equal grounds, it is assumed that the depot must be resupplied with the required departure propellant prior to an Earth departure. Additionally, the supplied propellant must be sufficient to satisfy the refuelling requirements of the departure after a 90-day loiter time.

Figure 10a and b compare the recurring for the aerocapture and propulsive capture options. These both assume that a Delta IV Heavy is used exclusively to deliver all depot resupply propellant. The three recurring Ares V launches are not shown as they are the same for both options. As a large number of PRMs need to be built, an 85% learning curve is applied to the PRM unit cost.



(a) Aerocapture



(b) Propulsive Capture

Figure 10: Recurring cost (less Ares V) of both depot options assuming Delta IV Heavy depot resupply.

From the above graphs, it is apparent that the propulsive capture option has considerably higher resupply costs: between $2.4B and $4.5B more than the aerocapture option depending on the opportunity. With fixed costs between the two depot options approximately equal and the recurring costs decidedly in favor of aerocapture, it appears that it is worth the investment to pursue aerocapture.

Boiloff Sensitivity

There is currently no standardized method to model cryogenic propellant boiloff while a vehicle is in space. Additionally, the choice of the propellant boiloff rates in this study may seem somewhat arbitrary despite their traceability from the ESAS Final Report. In an effort to assess the effect that boiloff has on the propulsive capture option, a sensitivity analysis was conducted assuming a departure on the 2031 opportunity for minimum total C3. The Mars Cargo Lander payload (less aerocapture) was used as the reference payload. Since this is a large payload with a relatively large arrival C3, the effect of boiloff will be readily apparent from the change in refuel mass. The baseline amount of propellant resupply is 97.8 MT.

A general range of LH2 and LOX boiloff rates can be obtained by examining other documents which cite propellant boiloff. These documents and their assumed boiloff rates are presented in Table 14.

Reference	LH2 Rate	LOX Rate
Baseline	0.200 %/day	0.020 %/day
Ref. [3]*	~0.363 %/day	N/A
Ref. [15]*	~0.260 %/day	~0.047 %/day
Ref. [16]*	~0.150 %/day	~0.020 %/day
Ref. [17]	0.127 %/day	0.016 %/day

*Calculated based on surface area of EDS LOX and LH2 tanks

Table 14: Representative boiloff rates from various sources.

Given the data above, the LH2 boiloff rate is widely varied from 0% per day (zero-boiloff case) to 1.0% per day. Similarly, the LOX boiloff rate is widely varied from 0% per day to 0.1% per day. The results of the sensitivity are shown in Figure 11, with the "Most Likely Region" being bounded by the extreme values in the Table 14.

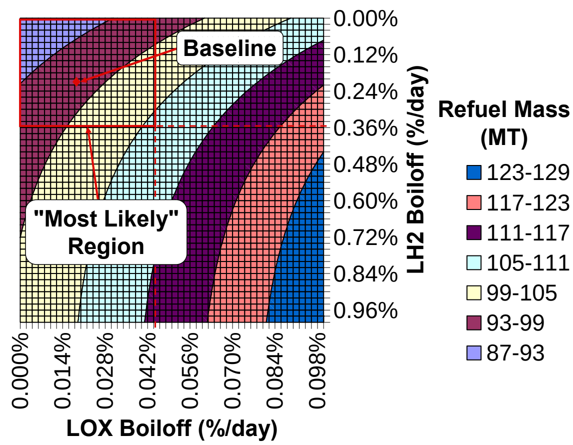


Figure 11: Refuel propellant mass sensitivity to boiloff rates.

According to Figure 11, the amount of refuel mass can vary by up to 42 MT for this wide range of propellant boiloff rates. However, within the Most Likely Region, the refuel mass varies by 17 MT. The advantage of implementing a low-boiloff system over a long storage time (approximately 300 days in this case) becomes readily apparent from the sort of sensitivity.

To bring this sensitivity analysis back into the architecture study at hand – would a zero-boiloff system make the propulsive capture option desirable over the aerocapture option? This question is easily answered as "No." Given the same departure conditions, the aerocapture option would still result in a 67 MT propellant mass savings over the propulsive option. This suggests that the least expensive Mars transportation architecture is not dependent on boiloff rate but rather on the mode of arrival.

Comparison with NTR

In this study, the idea of using a nuclear thermal departure stage was discarded from the start. This elimination does need some justification before it can be readily accepted. To frame this trade, the EDS-Propellant Depot combo with an aerocaptured payload will be compared against a dedicated nuclear thermal TMI stage with a similar aerocapture system, shown in Figure 12.

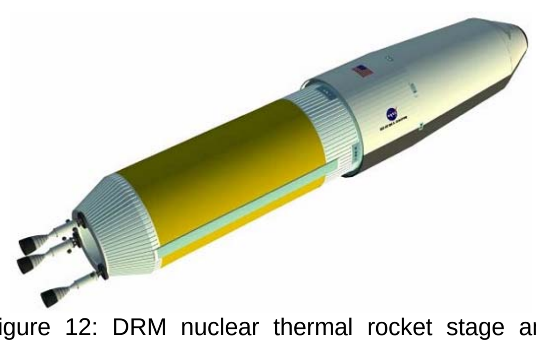


Figure 12: DRM nuclear thermal rocket stage and aerocapture payload.

The DRM v3.0 is leveraged to obtain the nuclear thermal rocket (NTR) departure stage. Earlier analysis has shown that the amount of propellant needed for departure is dependent on the opportunity, so the required propellant to accomplish each mission is once again calculated to ensure that the DRM NTR stage can be used without major design changes. The DRM sizes the NTR for the fast-transfer opportunity in 2009, resulting in a maximum propellant capacity of 54 MT of LH2. Applying losses for ullage, start-up losses, residuals, and 90 days of boiloff, each NTR would have just over 47 MT of usable propellant.

Analysis for an aerocaptured payload demonstrates propellant requirements ranging from 42 MT to 49 MT using an NTR departure stage. The assumed NTR is unable to push the ERV on three of the opportunities investigated. The NTR could possibly achieve these three burns if the boiloff time was reduced to only 41 days. Alternatively, the NTR stage could be lengthened such that its maximum propellant capacity was around 56 MT. As seen in the boiloff sensitivity, the propellant mass variability due to boiloff is quite large. This kind of propellant mass uncertainty precludes a detailed sizing of the NTR – so the nuclear stage remains as sized in the DRM, which is detailed Table 15. The shadow shield mentioned in the table is used to absorb and deflect radiation from the nuclear reactor on the crewed flight only. The concept of operations involving the NTR also remains the same – i.e. each NTR stage is expended after the TMI burn is executed.

Propellant	LH2
Thrust	45,000 lbf
Isp	960 s
Dry Mass	23,400 kg
Max Propellant	53,729 kg
Shadow Shield Mass	3,200 kg
Stage Length	28 m
Stage Diameter	8.4 m

Table 15: DRM v3.0 baseline NTR departure stage.

Unfortunately, do to the radiation emitted by the nuclear reactor, the system cannot be started until it reaches orbit, meaning the

NTR stage cannot function as an upper stage similar to the EDS. Additionally, the VAB cannot support an Ares V which stacks the Mars payload and the NTR stage on top of an upper stage used to achieve the correct Earth parking orbit. This immediately forces the departure stage to be launched separately from the payload, creating an additional three launches per mission. An ascent trajectory was analyzed for an Ares V launch of just the TMI stage. The Ares V demonstrated enough capability to achieve the reference 30 x 160 nmi orbit using only the solid rocket boosters and the Ares V core. Additionally, each of the individual Mars payloads can also achieve the reference orbit without the use of an additional upper stage.

The particularly prohibitive aspect about space nuclear systems is their high development costs. Most recently, this was demonstrated with the Prometheus Project, whose projected budget was $430M FY05**Error! Reference source not found.** prior to cancellation. The Nuclear Engine for Rocket Vehicle Applications (NERVA) program was a program executed from 1961 to 1973 to build nuclear powered rockets and missiles. Before it was cancelled, the program consumed $3.9B FY96[19], which translates into over $4.73B FY05. Since the cost of developing and implementing a man-rated nuclear thermal rocket cannot be estimated with any reasonable accuracy, an attempt is made to determine *how much it would have to cost in order to be competitive* with the EDS-Depot architecture. Table 16 and 18 outlines the fixed and recurring costs for each system.

Depot DDT&E	$1,419
Depot Unit	$110
Depot Launch	$563
PRM DDT&E	$494
Aerocapture DDT&E	$3,195
Total Fixed Cost	$5,781
Aerocapture Unit (3x)	$739
PRM Hardware (max)	$802
Delta IV Heavy Launches (max)	$1,850
Ares V Launches (3x)	$1,688
Total Recurring Cost	$5,079
Overall Cost (through 1st mission)	$10,860

Table 16: Depot architecture cost through first mission.

NTR DDT&E	N/A
Total Fixed Cost	N/A
NTR Unit (3x)	N/A
Ares V Launches (6x)	$3,375
Total Recurring Cost	≥ $3,375
Overall Cost (through 1st mission)	≥ $3,375

Table 17: NTR architecture cost through first mission.

Although the data in the table above is incomplete, it is possible to derive some conclusions about an NTR system versus the EDS-Depot architecture.

First, considering recurring costs, the NTR system is at a disadvantage as it requires six Ares V launches per mission. This alone compensates for nearly 3/5 of the depot recurring cost. The remaining unknown is the NTR departure stage unit cost. Intuition says that three nuclear reactors and their associated vehicles will cost more than $1.7B to fabricate. Looking at the fixed cost, an NTR system would have to be developed, flight tested, and certified for under $6B. Considering that NERVA consumed $4.7B pursing the NTR engine only, it might be difficult to achieve a human-rated nuclear vehicle for under $6B. However, regardless of the fixed costs associated with the development of either architecture, the NTR system will be more expensive in the long run if each stage is discarded as the DRM suggests.

Next, considering the near-simultaneous departures of the ERV and Cargo payloads each mission, the NTR architecture represents a more complicated concept of operations (conops) than the depot alternative. Four Ares V launches and two automated EORs must be performed within a short amount of time (to prevent excessive boiloff of the TMI propellant) as well as two remote nuclear reactor start-ups. The propellant depot conops involves only two Ares V launches and two automated EORs (to refuel at the depot).

Additionally, there will be a host of environmental concerns regarding launching three nuclear reactors (four including the Mars surface power system) for each mission to Mars. These concerns could likely delay the verification process and drive up cost and development time. Both the propellant depot and an aerocapture vehicle are not subject to this kind of scrutiny, eliminating some possible bureaucratic hang-ups during development and implementation.

Overall, it appears that the EDS-Depot option is superior to the development of a nuclear rocket. Despite the cost ambiguity of developing such technologies as NTR, intuition and operational considerations both indicate that an NTR stage in the architecture would be disadvantageous when compared to a depot architecture.

Launch and IMLEO Comparison

Cost, thus far, has been the primary driver in deciding which architecture is superior to the other. However, cost estimation at this level can have a lot of variability. It may be more beneficial to look at the three investigated architectures purely from a launch and IMLEO standpoint. The total number of launches for the three architectures is shown in Figure 13 assuming the Delta IV Heavy as the exclusive depot resupply vehicle.

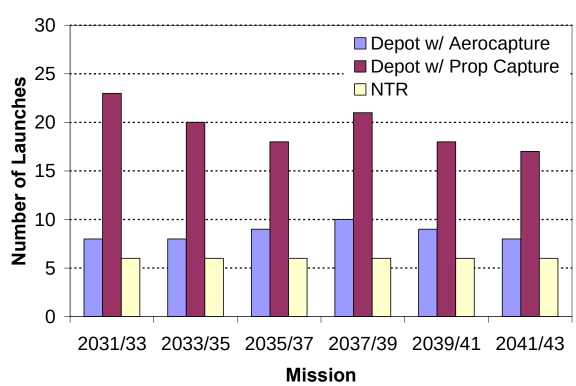


Figure 13: Total number of launches per mission for each architecture.

The total IMLEO is shown below in Figure 14. For the depot options, this IMLEO includes:

- three Mars payloads
- three EDS stages with propellant remaining from ascent
- all necessary propellant required to resupply the depot (including boiloff) for the mission
- all PRM hardware mass associated with all resupply missions
- the propellant depot dry mass

For the NTR option, the IMLEO includes the three Mars payloads and three of the nuclear TMI stages.

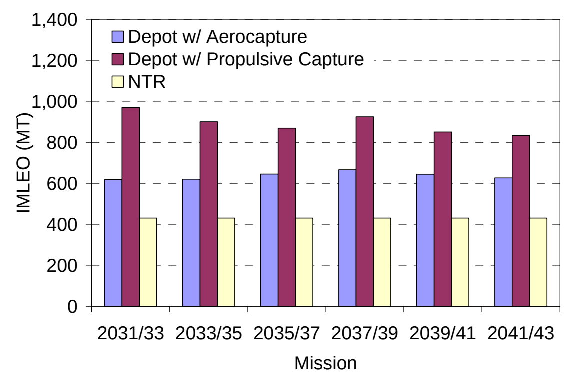


Figure 14: IMLEO for all three architectures.

The NTR system consistently has the least number of launches and IMLEO as compared to both depot options. However, the uncertainty associated with its development cost and the large cost incurred when each nuclear stage is discarded prevent a nuclear solution from being viable in this study.

CONCLUSION

In order to make human excursions to Mars more feasible, the use of existing hardware would be desirable. Assuming that the NASA-proposed Ares V launch vehicle is operational during the lunar campaign, it can be used to send significant payloads to Mars given a pre-deployed propellant depot. Three different levels of technology development were also considered – propulsive Mars capture (low technology development), aerocapture at Mars (moderate to extensive), and nuclear propulsion (highly extensive) – to help gauge which technology would be most economic for a Mars campaign. It was shown that propulsively capturing at Mars using the ESAS-proposed EDS provides the lowest life-cycle cost and the lowest risk. It was also shown that an NTR solution would likely be more operationally complex and expensive than both depot options in the long run. Overall, the use of a propellant depot can be a relatively inexpensive, enabling system for human-Mars exploration.

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