NASA/TM—2010–216450

Ares V Utilization in Support of a Human Mission to Mars

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November 2010

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Acknowledgments

The work documented in this Technical Memorandum (TM) was performed during the summer of 2010 as a joint effort between Ares V team members at Marshall Space Flight Center, Glenn Research Center, Kennedy Space Center, Ames Research Center, Zero Point Frontiers Corporation, and Jacobs ESTS Group/Triumph Aerospace. The authors would like to particularly acknowledge Constellation Program support from the Johnson Space Center, persons who worked on the Mars DRA 5.0 team, and those involved with generating the further detailed "7-Launch NTR Space Transportation System for NASA's Mars DRA 5.0" AIAA paper, as these studies provided a baseline from which this TM was created.

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LIST OF ACRONYMS AND SYMBOLS

ACO	Advanced Concepts Office
ACS	attitude control system
AF	Air Force
AIAA	American Institute of Aeronautics and Astronautics
AR&D	automated rendezvous and docking
ARF	Assembly Refurbishment Facility
ATLAST	Advanced Technology Large-Aperture Space Telescope
CaLV	Cargo Launch Vehicle
CARD	Constellation Architecture Requirements Document
CEV	Crew Exploration Vehicle
CFM	Cryogenic Fluid Management
CG	center of gravity
CLV	Crew Launch Vehicle
CMG	Control Moment Gyroscope
CS	core stage
CVS	Concept Validation Study
CxP	Constellation Program
CY	calendar year
DAV	descent/ascent vehicle
DDT&E	design, development, test, and evaluation
DES	discrete event simulation
DRA	Design Reference Architecture
DRM	Design Reference Mission
ECO	engine cutoff
EDL	entry, descent, and landing
EDS	Earth Departure Stage
EPF	Eastern Processing Facility
EOI	Earth orbit insertion
ESAS	Exploration Systems Architecture Study
ESMD	Exploration Systems Mission Directorate
ETO	Earth-to-orbit

LIST OF ACRONYMS AND SYMBOLS (Continued)

FOM	figure of merit
GG	gravity gradient
GLONASS	Global Navigation Satellite System
GN&C	Guidance, Navigation, and Control
GOP	Ground Operations Project
GPS	Global Positioning System
GRAM	Global Reference Atmospheric Model
He	helium
HB	high bay
HLLV	heavy lift launch vehicle
IDAC-3	Integrated Design Analysis Cycle-3
IMLEO	initial mass in low-Earth orbit
IMU	inertial measurement units
ISRU	in situ resource utilization
ISS	International Space Station
KSC	Kennedy Space Center
LAT	Lunar Architecture Team
LCCR	Lunar Capabilities Concept Review
LEO	low-Earth orbit
LPE	Launch Pad Element
LH_2	liquid hydrogen
LOM	loss-of-mission
LOX	liquid oxygen
MAF	Michoud Assembly Facility
MAV	Mars ascent vehicle
MDAV	Mars descent/ascent vehicle
MECO	main engine cutoff
ML	mobile launcher
MLE	Mobile Launch Element
MMOD	micrometeoroid and orbital debris
MOI	Mars orbit insertion
MPS	main propulsion system
MSFC	Marshall Space Flight Center

LIST OF ACRONYMS AND SYMBOLS (Continued)

MTV	Mars transfer vehicle
NEO	near-Earth object
NEP	nuclear-electric propulsion
NERVA	Nuclear Engine for Rocket Vehicle Application
NPF	Nuclear Processing Facility
NSO	nuclear safe orbit
NSSP	Next Step in Strategic Partnership
NTP	nuclear thermal propulsion
NTR	nuclear thermal rocket
O&M	operations and maintenance
OML	outer mold line
OMS	orbital maneuvering system
OMV	orbiting maneuvering vehicle
Ops Con	Operational Concepts
OSF	Offline Stacking Facility
PA-C3	Phase A-Cycle 3
PA-C3′	Phase A-Cycle 3 Prime
PA-C3″	Phase A-Cycle 3 Double Prime
PA-C3D	Phase A-Cycle 3 point-of-departure concept
PBAN	polybutadiene acrylonite
PMAD	power management and distribution
PMBT	propellant mean bulk temperature
POD	point-of-departure
POST	program to optimize simulated trajectories
PS	payload shroud
QD	quantity distance
R&D	rendezvous and docking
RCS	reaction control system
RPOD	rendezvous, proximity operations, and docking
RPSF	Rotation Processing and Surge Facility
S&MA	Safety and Mission Assurance
SBU	Sensitive But Unclassified
SHAB	surface habitat

LIST OF ACRONYMS AND SYMBOLS (Continued)

SM	service module
SPE	Spacecraft Processing Element
SRB	solid rocket booster
SRPE	Solid Rocket Processing Element
SRR	Systems Requirements Review
SRRE	Spacecraft Recovery and Retrieval Element
SSC	Stennis Space Center
SSME	Space Shuttle Main Engine
SSPF	Space Station Processing Facility
ST	storage time
STS	Space Transportation System
TCS	thermal control system
TEI	trans-Earth injection
TIM	Technical Interface Meeting
TLI	trans-lunar injection
TM	Technical Memorandum
TMI	trans-Mars injection
TransHab	transit habitat
TRV3D	trajectory vehicle, Cycle 3, D-vehicle configuration
VAB	Vehicle Assembly Building
VI	Vehicle Integration
VIE	Vertical Integration Element
VIF	Vertical Integration Facility
ZBO	zero boiloff

TECHNICAL MEMORANDUM

ARES V UTILIZATION IN SUPPORT OF A HUMAN MISSION TO MARS

1. INTRODUCTION

Identified by numerous studies over the past 20+ yr as a necessary development that serves as the foundation upon which a human space flight exploration strategy must be built, a heavy lift vehicle like the Ares V has the potential to open the solar system to human exploration. Initial efforts by NASA's Exploration Systems Mission Directorate (ESMD) and the Constellation Program (CxP) to utilize the performance capability offered by the Ares V launch vehicle for a Mars mission strategy was undertaken in 2007 for what was to become Mars Design Reference Architecture 5.0 (DRA 5.0). This early assessment showed that the performance capability of the Ares V was necessary in order to reduce the number of launches required for the Mars architecture to a reasonable level (7 flights for nuclear and 11 or 12 flights for chemical).

With over 4 yr of increased knowledge of the Ares V concept, an evolved vehicle configuration, and established Ares V element teams in place for almost a year-and-a-half, the Ares V team assessed the capability offered by Ares V against the Mars Design Reference Mission (DRM) established in the Exploration Systems Architecture Study (ESAS)[1] and later the Constellation Architecture Requirements Document (CARD). This provided valuable insight into the available performance capability to low-Earth-orbit (LEO) and the available performance growth options available; the required functionality to launch, maintain, and assemble a Mars transfer vehicle (MTV) in LEO; and the capability of the Earth Departure Stage (EDS) to possibly perform the trans-Mars injection (TMI) maneuver if required. In addition, it allowed the Ares V team to assess the ability of the vehicle to be built, stored, transported, and launched at the required rate for the Mars architecture. The initial assessment in these focus areas will be discussed in this Technical Memorandum (TM), and future work will be identified that will provide confidence that the Ares V launch vehicle can help meet the challenges presented by an exploration strategy for human missions to Mars.

2. MARS ARCHITECTURE OVERVIEW: DESIGN REFERENCE ARCHITECTURE 5.0

While the trade space is quite large for potential scenarios for Mars exploration missions, a valuable resource that was leveraged by the Ares V team was Mars DRA 5.0. This architecture for sustained human exploration of Mars was worked throughout 2007 and 2008 and publically released in 2009.[2] As outlined in that study, a preferred approach for a human mission to Mars includes utilizing a conjunction class mission design (typically referred to as a 'long-stay' mission), predeploying cargo assets in both Mars orbit and on the surface before launching the crew, utilizing aerocapture for Mars orbit insertion (MOI) to some extent (in this case for the cargo assets), and utilizing in situ resource utilization (ISRU) to some extent in order to decrease the amount of required landed mass on the surface. These high-level trades were discussed in detail in DRA 5.0 with the remaining top-level trade explored being the in-space propulsion technique used for the TMI maneuver and the MOI and trans-Earth injection (TEI) maneuvers. This top-level trade tree can be seen in figure 1.

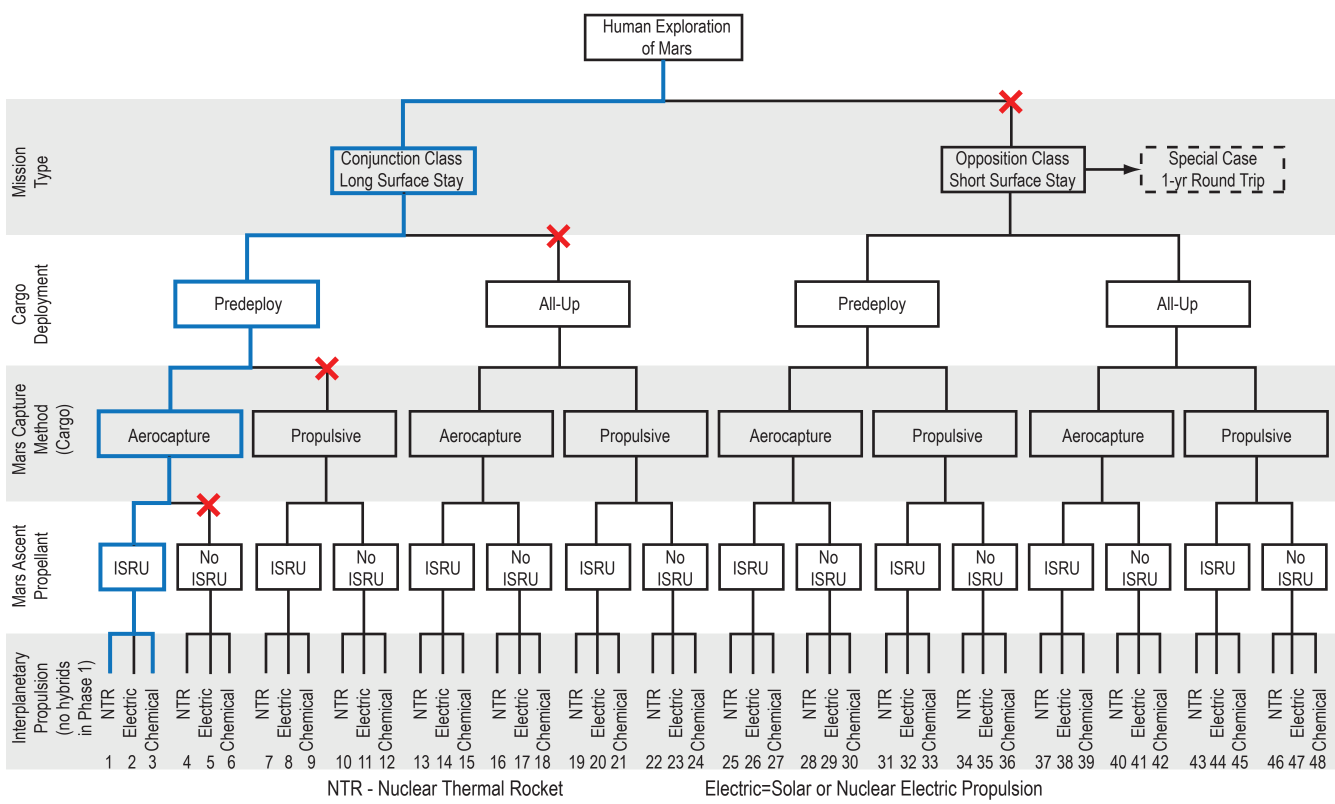

Figure 1. Mars DRA 5.0 top-level trade tree.

As shown in figure 2, the nuclear thermal propulsion (NTP) option was the preferred approach identified in DRA 5.0. The primary reason for this was the relatively large reduction in launches required using the NTP and chemical propulsion options explored in that study. While the NTP option required only 7 launches to meet the delta-velocity (dV) requirement for TMI with the payloads assumed, the chemical propulsion options required up to 12 launches. However, while the technology was recognized in DRA 5.0 for the benefit it would provide, these chemical options did not assess the potential of utilizing cryogenic propellant transfer in LEO. This option is the basis for the chemical options presented in this TM.

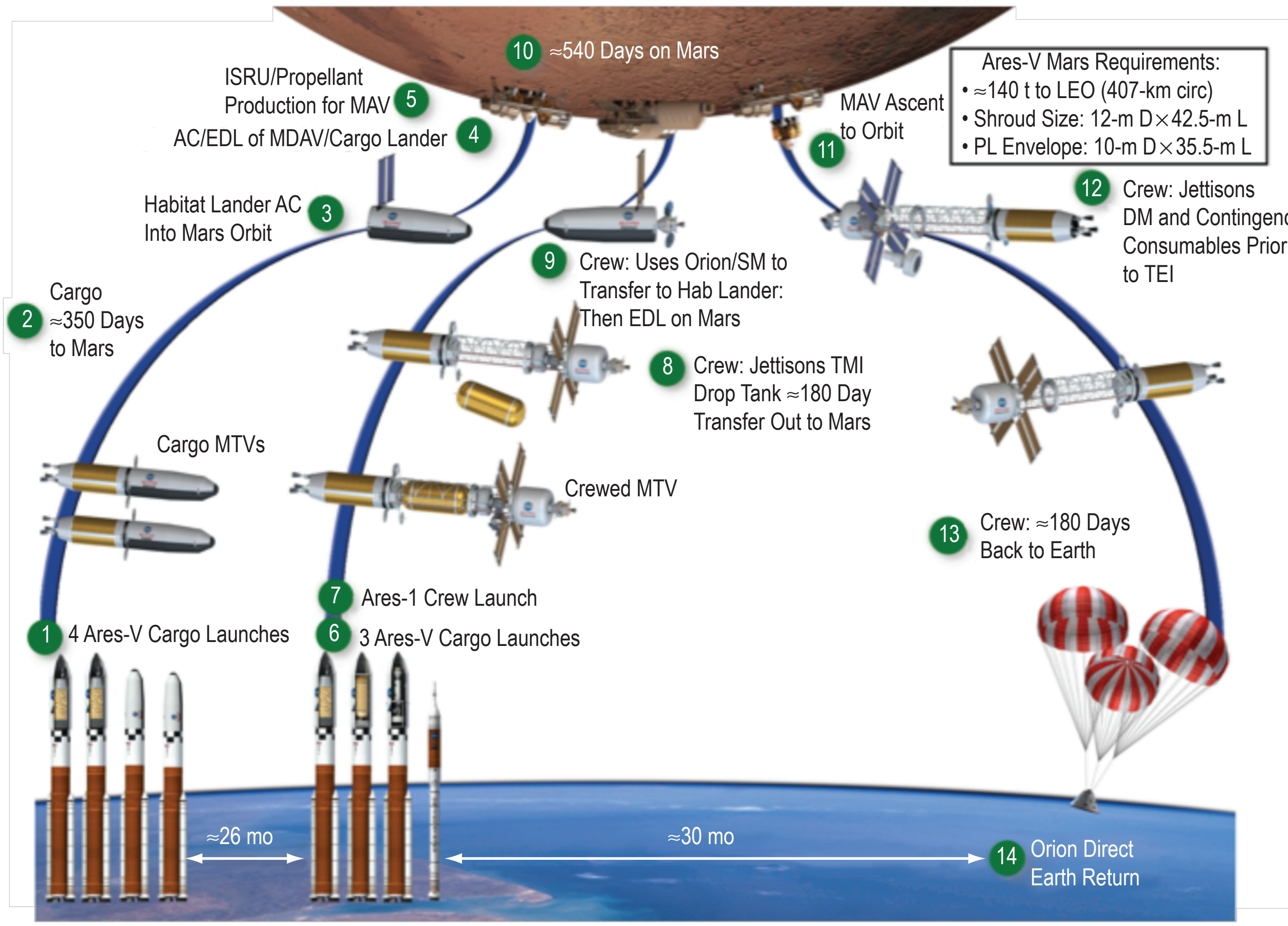

Figure 2. DRA 5.0 long-stay Mars mission overview (seven-launch NTP strategy).

The mission architecture presented in DRA 5.0 consists of the following 14 main characteristics:

(1) Ares V launches components of two cargo vehicles to LEO. The first launch for each of the cargo vehicles delivers a “dual-use” shroud with an encapsulated payload (an ascent vehicle/cargo lander for one MTV and a lander/surface habitat for the other MTV), and the remaining

launches deliver the propulsive element needed to provide the dV necessary to inject that shroud and encapsulated payload on an outbound trajectory to Mars.

(2) These two cargo MTVs, after being assembled in LEO, await the most energy efficient alignment of Earth and Mars for the given synodic opportunity, and then the propulsive element performs the TMI maneuver. This maneuver places the cargo MTVs on a low-energy transfer that takes on the order of 9–12 mo to transfer between the Earth and Mars.

(3)–(5) The dual-use shroud is used for an aerocapture maneuver to insert the shroud and payload into a Mars orbit. The ascent vehicle/cargo lander performs an entry, descent, and landing (EDL) maneuver and begins to make propellant for the Mars-to-Mars-orbit ascent. The lander and crew habitat within the dual-use shroud wait in Mars orbit for crew arrival and eventual EDL with the crew inside.

(6) About 2 yr later (on the order of 26 mo), a crewed MTV is being assembled in LEO. The Ares V is once again responsible for launching very large payloads to LEO in support of this MTV, which consists of a transit habitat (TransHab) module, contingency consumables, solar arrays, a docking mechanism for rendezvous with an Orion crew capsule (launched on a crew launch vehicle (CLV)), MOI and TEI propulsive components, and TMI propulsive elements.

(7)–(8) Once this vehicle is assembled, a CLV delivers the crew to the crewed MTV for transit to Mars. While the cargo MTVs used a low-energy transfer to Mars, the crewed MTV performs a more demanding TMI maneuver to place the crew on a shorter duration transfer to Mars—on the order of 6 mo as opposed to 9–12 mo. This reduces the amount of time that the crew is subject to the space environment (microgravity, radiation, etc.).

(9) Propulsive capture (rather than aerocapture) is used to place the crew in Mars orbit. They then rendezvous and transfer to the lander and crew habitat, which then performs the EDL maneuver to the surface of Mars. The crewed MTV stays in Mars orbit for the eventual trip back to Earth.

(10)–(13) The crew stays on the surface of Mars for a long duration, on the order of 540 days. Eventually, they board the ascent vehicle to transfer to Mars orbit, rendezvous, and transfer over to the waiting crewed MTV. The TEI maneuver is performed, which is a higher than minimum-energy transfer that places them on an approximately 6-mo-long trajectory back to Earth.

(14) The crew utilizes the Orion capsule for reentry back to the surface of Earth. The total crew trip duration is about 900–1,000 days.

3. ARES V OVERVIEW: LAUNCH VEHICLE OPTIONS

3.1 Ares V Concept History

Originally defined in ESAS as the cargo launch vehicle (CaLV), the Ares V concept was envisioned as a 27.5-ft-diameter/Space Shuttle Main Engine (SSME) based, heavy-lift launch vehicle with two solid rocket boosters (SRBs) (similar to the Space Transportation System (STS)). However, subsequent studies showed a single-engine upper stage, referred to as the EDS, increased payload injection to LEO and represented an optimized solution as a translunar injection (TLI) stage. As knowledge of functionality required, costs, schedule, and performance requirements matured, the vehicle underwent design changes leading up to the Lunar Capabilities Concept Review (LCCR) ("ESAS-LCCR Ares V; Ares V Pre-Phase-A Refinements," Powerpoint Presentation, 2009) in the summer of 2008. This served as the Mission Concept Review for Ares V and technically served as the transition from conceptual design studies to an Ares V Phase A concept. Currently, the Ares V is moving toward a Systems Requirements Review (SRR) in July 2011 that will formally begin the transition to a Systems Design Review (SDR) and into a Phase B project. Figure 3 shows the progress and revisions made on the Ares V vehicle since ESAS.

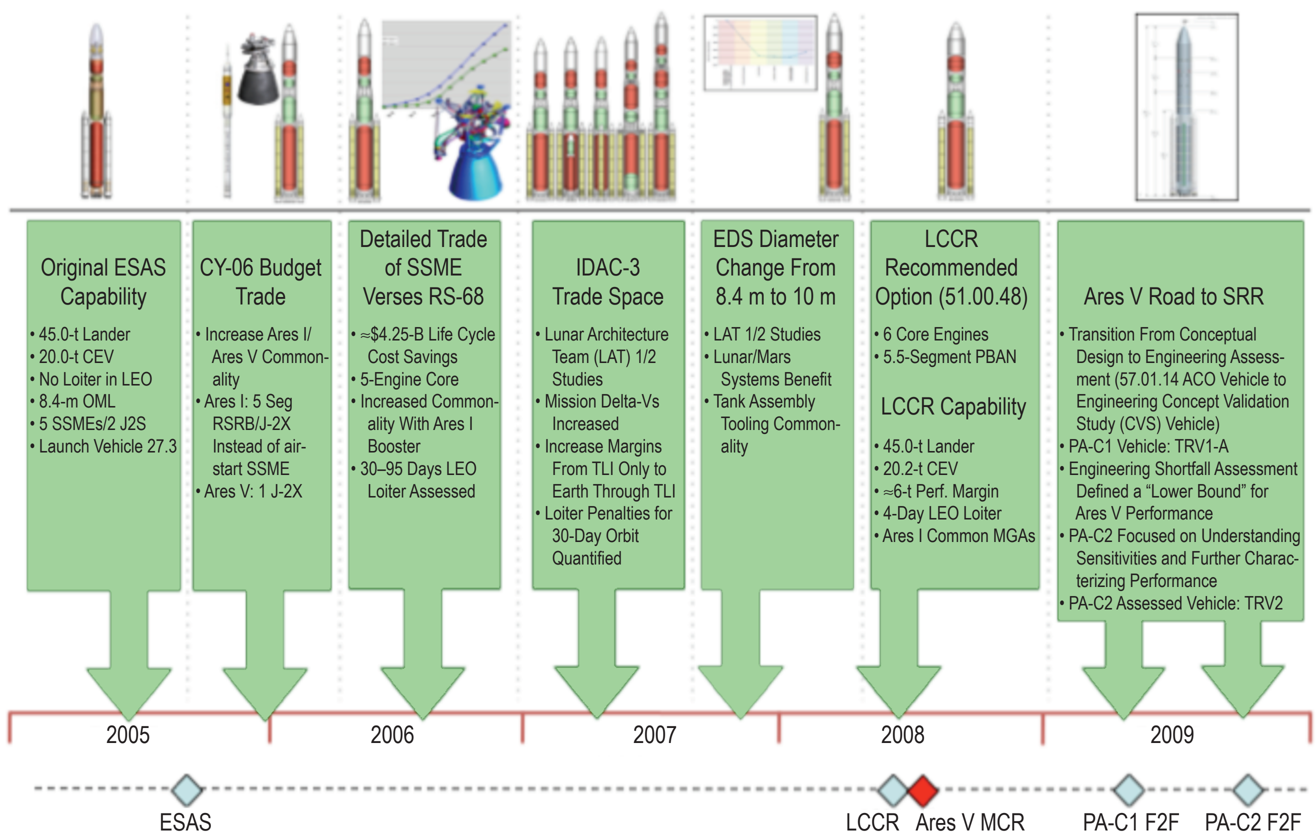

Figure 3. Timeline of major Ares V changes/reviews since ESAS.

3.2 Concepts Assessed For Mars Design Reference Architecture 5.0

Performance analysis for the Ares V launch vehicle was assessed for DRA 5.0 during the summer and fall of 2007. During that timeframe (about 1.5 yr after ESAS), Ares V was involved in what was known as Integrated Design Analysis Cycle-3 (IDAC-3). The Ares V had recently transitioned from a 27.5-ft diameter/SSME based core stage (CS) to a 33-ft diameter/RS-68 engine based CS. At that time, many potential booster options, various CS lengths and diameters, etc. were being explored. For DRA 5.0, a reference vehicle was chosen that incorporated five RS-68B engines on a 33-ft diameter CS, an Ares I-like five-segment polybutadiene acrylonite (PBAN) SRB, a single J-2X on a 27.5-ft EDS, and a 27.5-ft diameter shroud (dubbed the 45.0.2 concept). While keeping that base configuration, several other vehicle options were assessed for "payload to LEO" sensitivity, as well as several orbit options for payload delivery.

A very important finding during the DRA 5.0 assessment was the characterization of the impact of the larger shrouds required for delivering 'Mars-sized' payloads to LEO. Several shroud options were assessed that varied both diameter and overall length. In addition, the dual-use shroud was assessed, which provided Earth-to-orbit (ETO) ascent protection, in-space thermal protection, MOI thermal protection, and thermal protection to the payload during the Mars entry, descent, and landing phase. Furthermore, a fuel-stage delivery 'nose cone only' type of shroud was assessed for LEO propellant delivery options. Figure 4 depicts a few of these shroud options, but they are discussed more fully in the DRA 5.0 addendum and follow-on technical papers listed.

3.3 Concept Assessed During Phase A-Cycle 3′

As analysis on the Ares V concept continued through to the more recent Ares V analysis cycle (Phase A-Cycle 3′ (PA-C3′)), the Mars mission architecture remained locked to that presented in Mars DRA 5.0, including two cargo MTVs that are predeployed (utilizing aerocapture for MOI) before a crewed MTV departs on the following synodic opportunity (utilizing propulsive capture for MOI). As such, MTV options were chosen within that architecture to include nuclear thermal propulsion (NTP) and chemical propulsion options intended to reduce the number of flights. The Ares V launch vehicle was employed to deliver components for these MTVs, such as payload and propellant. Each propulsion option required a different number of flights (e.g., NTP consisted of a seven-launch campaign as presented in DRA 5.0 and a competitive chemical option that was found during PA-C3′ consisted of a nine-launch campaign).

During PA-C3, a range of concepts was assessed that represented the best estimate for Ares V concepts that could potentially balance all of the performance, cost, and schedule figures of merit (FOMs) for the lunar DRMs. Among these, a concept emerged as a preferred point-of-departure (POD) concept—PA-C3D. This concept utilized five-segment 'fast-burn' PBAN SRBs, a 33-ft diameter CS with five RS-68B-E/O regeneratively cooled engines, a 33-ft diameter EDS with a single J-2X, and a 33-ft diameter composite tangent-ogive shroud. This concept was the chosen option to assess against the Mars DRM. The major attributes of this concept can be seen in figure 5.

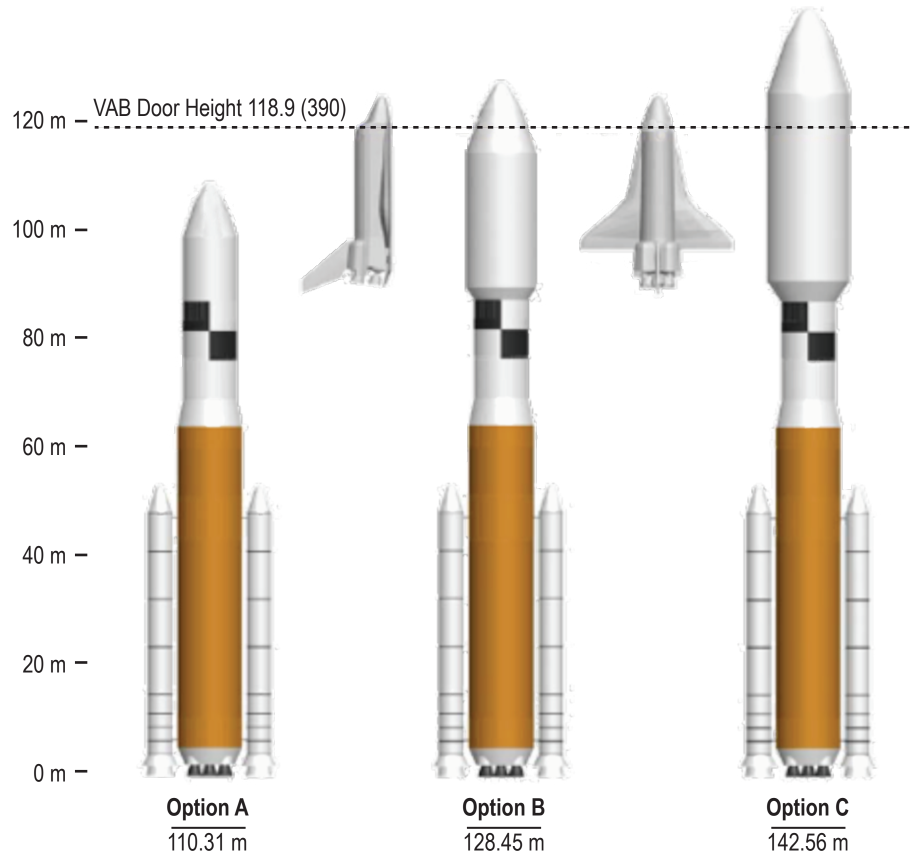

Figure 4. Potential shroud options assessed for DRA 5.0.[3]

In identifying the value of assessing the Mars DRM, four fundamental questions were posed. The answers to the following four questions aided in characterizing the synergy available between the lunar and Mars architectures, mainly the utilization of a common Ares V launch vehicle (with required modifications identified at a qualitative level):

(1) What is Ares V PA-C3D initial mass in LEO (IMLEO) capability as a function of altitude? Assess a range of orbit options (elliptical phasing, elliptical disposal, and direct inject to circular orbits) to determine the overall vehicle performance (with system estimates for circularization and disposal at the various options).

(2) What are the IMLEO growth potential/options for Ares V PA-C3D? Identify reasonable growth paths for PA-C3D if more performance is required.

(3) What is the impact of in-space pre-TMI burn functions on EDS design? Identify assumed functions required to insert and maintain the payload until the TMI maneuver.

(4) What are the extensibility options for Ares V EDS to the Mars TMI vehicle? Identify assumed functions required to perform the TMI maneuver.

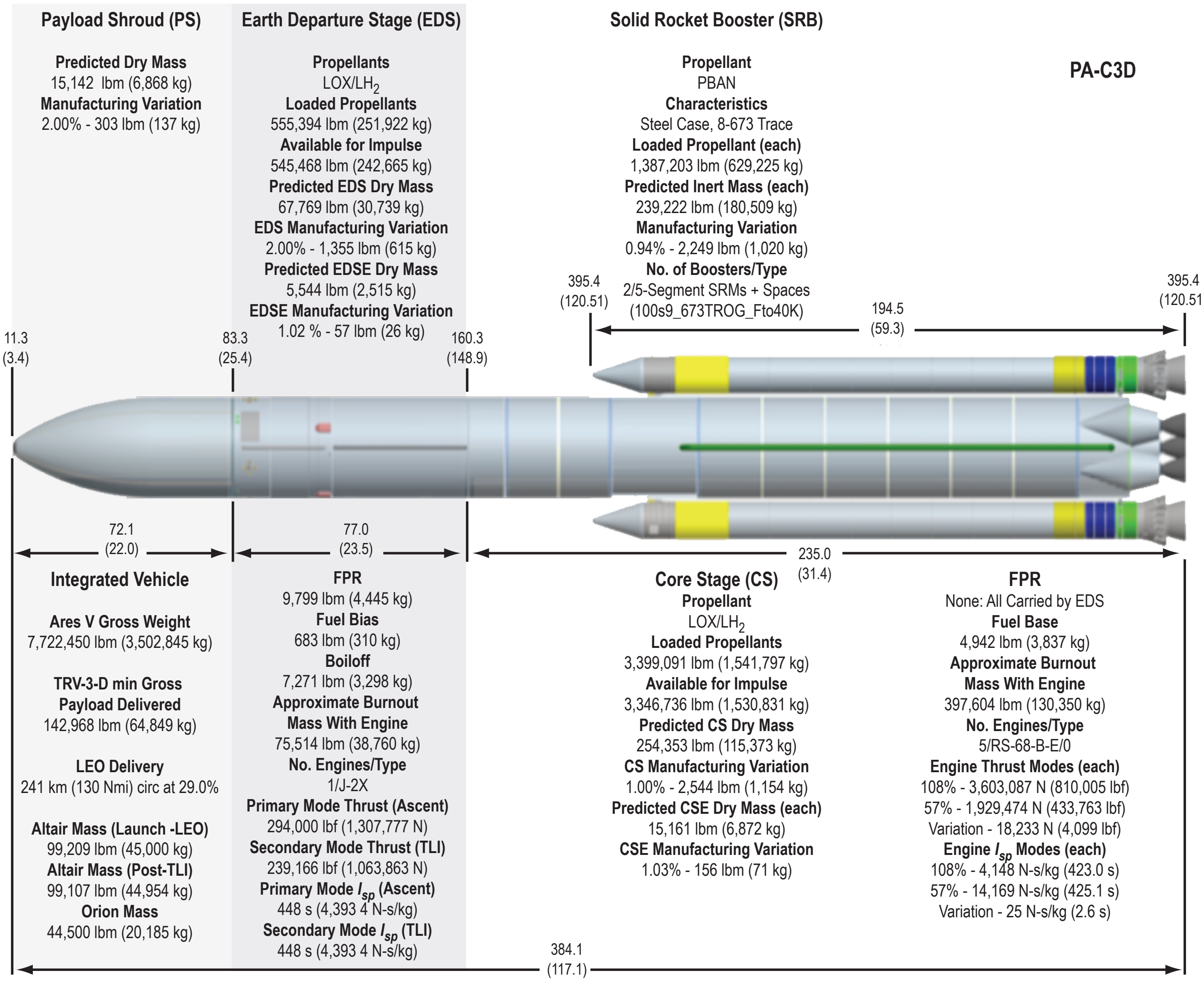

Figure 5. PA-C3D Ares V concept (baseball card).

3.4 Key Mass Assumptions For Phase A-Cycle 3′ Mars Design Reference Mission Assessment

Several key high-level assumptions were made to facilitate the analysis performed relative to the Mars DRM. First, it was determined that available performance would be decremented by the mass of the system required to circularize the payload to the final assembly altitude. This allowed the EDS to be decoupled from that particular system until a final delivery orbit and assembly orbit were determined. Furthermore, the mass allocation for the system required to jettison and dispose of the empty EDS was taken from the available performance for similar reasons. These systems will be further described in the following section.

At the EDS element level, the entire burnout mass of the stage was upsized by 18% (from 25.5 t to 30 t) to account for known system level impacts that could not be analyzed during the time constraints of PA-C3′. This 4.5 t was an initial estimate to account for structural growth to accommodate the larger payloads and payload shrouds (PSs), upsized reaction control system/attitude control system (RCS/ACS) components for increased loiter durations and larger payloads, and other hardware impacts.

Even after taking these performance penalties (and others as discussed in the next section) into consideration, the PA-C3D configuration delivered more than required in support of the Mars DRM as outlined in the CARD. This CARD requirement of 125 t of payload was used as the basis for constructing the mission scenarios presented herein. Of the 125 t of available payload, when on-orbit propellant transfer was utilized, an additional 5-t decrement was implemented to account for rendezvous/docking hardware, transfer lines, etc., both on the delivering tank side and the receiving EDS side. Obviously, more detailed analysis is required to support the mass allocations given to allow for these EDS capabilities.

3.5 Performance of Phase A-Cycle 3D for Low-Earth Orbit Applications

Throughout the PA-C3 and PA-C3′ analysis, there have been several trajectory types that have been tailored based on the types of analysis being performed along with the available time to complete the analysis. Different studies use the trajectory type that is most applicable to the study's goals. The trajectory type chosen for the IMLEO study was the performance reference trajectory. While this is similar to the minimum performance trajectory, the main difference is that a knock-down is calculated for the minimum performance reference trajectory and not for the performance reference. Therefore, the performance reference is not applicable for comparing to requirement metrics and the CARD. The data produced in the IMLEO analysis is more suited for comparing between the different types of orbits and launch conditions than comparing directly to a target mass value.

The payload to LEO is maximized for a performance reference trajectory. The liquid engines, RS-68s and the J-2X, are run at the minimum guaranteed specific impulse (I_{sp}) and nominal thrust levels. The SRB trace is degraded following Shuttle methods and is based on the nominal burn rate at a 61-degree propellant mean bulk temperature (PMBT). Other characteristics of this trajectory type include February GRAM mean monthly winds and atmosphere, predicted mean mass, and the aerodynamic database in the nominal as delivered form. For all PA-C3 and PA-C3′ trajectories, the maximum dynamic pressure was limited to 800 lbf psf via throttling of the RS-68s.

For the IMLEO study, the performance reference trajectories consisted of a single ascent phase, generated in the program "Program to Optimize Simulated Trajectories" (POST). POST is a generalized point-mass, discrete-parameter, targeting and optimization program. It provides the capability to target and optimize point-mass trajectories for a powered or unpowered vehicle near an arbitrary rotating, oblate planet. POST has been used successfully to solve a wide variety of atmospheric ascent and reentry problems. Its generality is evidenced by its multiple phase simulation capability, which features generalized planet and vehicle models. This flexible simulation capability is augmented by an efficient discrete parameter optimization capability that includes equality and inequality constraints.

Mass to LEO was maximized for five different orbits, two elliptical orbits of 120×220 nautical miles (nmi) and 120×405 nmi, and three circular orbits with altitudes of 130 nmi, 220 nmi, and 405 nmi. The performance summaries for the two elliptical orbits are presented in table 1. The insertion altitude was allowed to vary and the optimal insertion for both elliptical orbits was near the perigee, though not exactly at perigee. Propellant was off-loaded from the fully loaded EDS tanks in order to optimize the performance to LEO. The optimal loading was around 70% for the two elliptical cases. The 120-×220-nmi orbit delivered 175 t, while the 120-×405-nmi orbit delivered 171 t. These performance values include everything taken to this orbit (the EDS burnout mass, circularization hardware and propellant, RCS/ACS propellant load, and all necessary disposal masses). It is the mass that arrives at these orbits and should be divided among the different parts as warranted.

Table 1. Performance summary for the elliptical orbits (all orbits to 29° inclination).

Insertion Orbit	120×220 nmi	120×405
Vehicle designation	TRV3D	TRV3D
Trajectory description	Performance Reference—PA-C3D Vehicle Configuration	Performance Reference—PA-C3D Vehicle Configuration
Trajectory date	4/9/2010	4/12/2010
Total mass at RS-68B ignition (lbm)	7,746,845.3	7,751,465.5
SRB loaded propellant	473,947	473,947
SRB inert mass	2,774,406	2,774,406
Core stage impulse propellant	3,351,666	3,351,666
Core stage jettison mass	375,718.4	375,718.4
Shroud jettison mass	14,268	14,268
EDS ascent impulse propellant	358,609.3	371,172.9
EDS ascent FPR propellant	13,434.7	13,306.6
Gross LEO mass is everything delivered to LEO: Includes EDS burnout mass, circularization hardware and propellant, RCS/ACS propellant load, and disposal mass. It is not the final payload of the vehicle and not the value to compare to requirements.		
Gross LEO mass (lbm)	**384,795.9**	**376,980.6**
Gross LEO mass (kg)	**174,540.5**	**170,995.5**
Total core usable propellant (impulse + start-up)	3,351,666	3,351,666
Total EDS usable propellant (impulse + FPR)	372,044	384,479.5
EDS usable propellant capacity (lbm)	544,784.8	544,784.8
Actual EDS propellant loading (%)	68.3	70.6
Delta EDS propellant (lbm)	–172,740.8	–160,305.4
J-2X burn time (s)	550.3	569.3
Insertion altitude (nmi)	121.5	122.1
Maximum acceleration (gs)	3.36	3.35
Maximum dynamic pressure (psf)	800	800

The performance summaries for the circular orbits are presented in table 2. The 130-nmi and 220-nmi orbits also used approximately 70% of the loaded EDS propellant, while the 405-nmi orbit used 83%. The mass delivered to the orbits was 176 t to the 130-nmi orbit, 166 t to the

220-nmi orbit, and 145 t to the 405-nmi orbit. The performance dropped off significantly with the higher circular orbit altitudes. Since the 120-×405-nmi orbit achieved 26 t more than a direct insertion into the 405-nmi orbit, it would be advantageous to examine the costs of circularizing from the 120-×405-nmi orbit to the desired final circular orbit.

Table 2. Performance summary for the circular orbits (all orbits to 29° inclination).

Insertion Orbit	130-nmi Circular	220-nmi Circular	405-nmi Circular
Vehicle designation	TRV3D	TRV3D	TRV3D
Trajectory description	Performance Reference—PA-C3D Vehicle Configuration	Performance Reference—PA-C3D Vehicle Configuration	Performance Reference—PA-C3D Vehicle Configuration
Trajectory date	4/22/2010	4/22/2010	4/22/2010
Total mass at RS-68B ignition (lbm)	7,745,468.5	7,753,871.7	7,759,173.6
SRB loaded propellant	473,947	473,947	473,947
SRB inert mass	2,774,406	2,774,406	2,774,406
Core stage impulse propellant	3,351,666	3,351,666	3,351,666
Core stage jettison mass	375,718.4	375,718.4	375,718.4
Shroud jettison mass	14,268	14,268	14,268
EDS ascent impulse propellant	354,786.4	384,108.7	438,016.5
EDS ascent FPR propellant	13,473	13,113.9	12,148.2
Gross LEO mass is everything delivered to LEO: Includes EDS burnout mass, circularization hardware and propellant, RCS/ACS propellant load, and disposal mass. It is not the final payload of the vehicle and not the value to compare to requirements.			
Gross LEO mass (lbm)	**387,203.8**	**366,643.7**	**319,003.5**
Gross LEO mass (kg)	**175,632.7**	**166,306.8**	**144,697.5**
Total core usable propellant (impulse + start-up)	3,351,666	3,351,666	3,351,666
Total EDS usable propellant (impulse + FPR)	368,259.4	397,222.6	450,164.7
EDS usable propellant capacity (lbm)	544,784.8	544,784.8	544,784.8
Actual EDS propellant loading (%)	67.6	72.9	82.6
Delta EDS prop (lbm)	–176,525.5	–147,562.2	–94,620.1
J-2X burn time (s)	544.5	589.1	671.4
Insertion altitude (nm)	130	220	405
Maximum acceleration (gs)	3.4	3.3	3.3
Maximum dynamic pressure (psf)	800	800	800

The final goal is to insert the payload into an ending circular orbit with the elliptical orbits considered as transfer orbits. This is accomplished by performing a single circularization burn at the apogee of the elliptical orbits. For preliminary analysis purposes, a dV budget was calculated for this burn for both of the elliptical orbits and is presented in table 3. In order to approximate the impact of gravity losses, the pure impulsive dV was increased by 5%. The circularization systems on the vehicle will need the capability to cover at least the amount of the listed dV.

Some of the equipment and mass that arrive in the circular orbit may not need to remain in the final orbit (i.e., the circularization system and EDS); hence, these items will need to be disposed

Table 3. Delta-V budget for circularization burn.

Desired Circular Altitude (nmi)	Elliptical Perigee Altitude (nmi)	Elliptical Apogee Altitude (nmi)	*dV* Budget (ft/s)
220	120	220	183
405	120	405	500

of properly. For this study, the disposal method consisted of a single deorbit burn that adjusts the circular orbit into an elliptical orbit with a 30-nmi altitude perigee. This perigee altitude is low enough to have the equipment reenter the atmosphere without any additional burns. The dV budgets recommended for these deorbit burns are presented in table 4. As with the circularization dV budget, a 5% increase was included to approximate the impact of gravity losses.

Table 4. Delta-V budget for disposal burn.

Starting Circular Altitude (nmi)	Deorbit Elliptical Perigee (nmi)	*dV* Budget (ft/s)
130	30	190
220	30	354
405	30	668

3.6 Payload Performance to Final Circular Orbit

As already discussed, the ascent performance results shown in tables 1 and 2 represent the total weight injected into each insertion orbit analyzed. The EDS dry weight along with the propellant and system weights necessary to circularize the payload into its final orbit must come out of this total weight. Additionally, the system must be sized to deorbit the EDS, provided it is not being used to support TMI. Figure 6 illustrates the conceptual mission profile assumed for the calculation of payload to the final circular orbit altitude.

The large number of possible on-orbit engine configurations led to a parametric sweep of the I_{sp} and mass fraction of this system. The sweep included an I_{sp} range of 280 through 450 s along with propellant mass fractions between 0.5 through 0.8. To evaluate the circularization requirements for the 220-nmi and 405-nmi apogee elliptical orbits, the dVs from table 3 were assumed. Using the I_{sp} and required dV, the ideal rocket equation is used to find the propellant necessary to perform circularization. That propellant amount is then used with the assumed mass fraction to calculate the system's inert weight. Two sets of numbers were generated for each insertion orbit, one representing the case where the entire EDS is circularized with the other sizing the system when the EDS is not circularized with the rest of the vehicle. For all cases, including direct circular insertion, the EDS is assumed to have an inert mass of 66,138 lbm, (≈30 t) not including

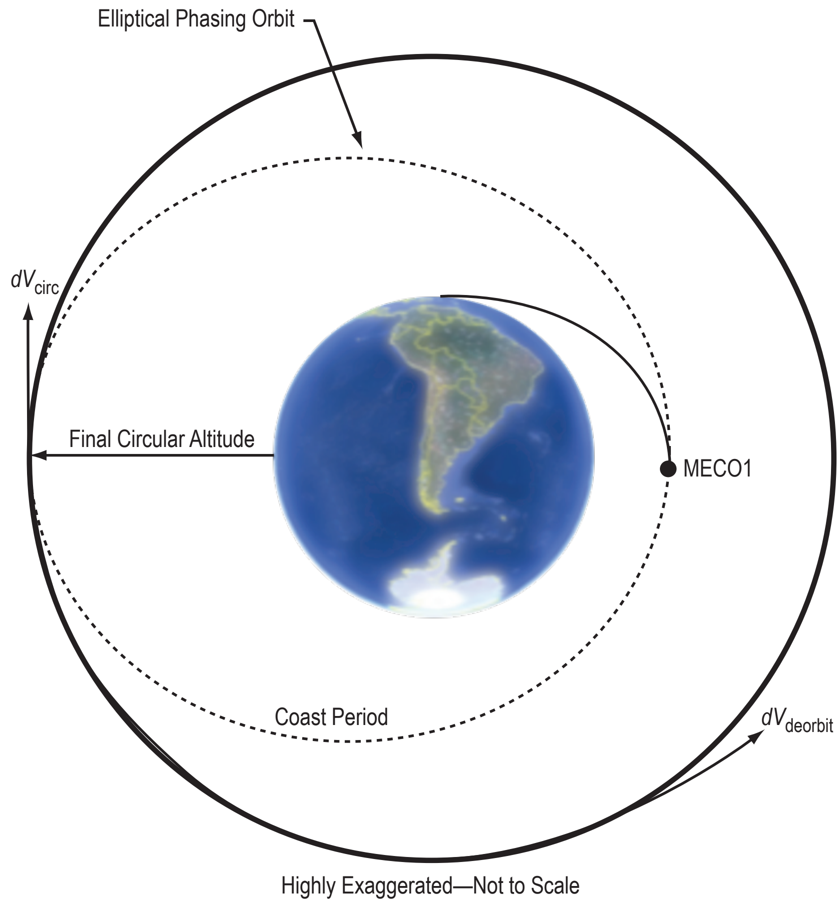

Figure 6. Conceptual LEO mission profile.

the on-orbit propulsion system. Results of the circularization system sizing to the 220-nmi altitude circular orbit from the 120-×220-nmi orbit are presented in table 5. The sizing results from the 120-×405-nmi orbit to the 405-nmi circular orbit are presented in table 6. Note: The payloads listed in tables 5 and 6 do not include the disposal system mass.

System requirements for the disposal of an empty EDS were then calculated separately using the dV information in table 7 using the same methodology, I_{sp}, and mass fraction ranges from the circularization parametric sweep. The propellant and system mass results for disposal of an empty EDS from the three circular obits can be found in table 7.

Table 5. Circularization system for 220-nmi circular orbit from the 120- × 220-nmi orbit.

Circularization performed with Total stack (Payload + Empty EDS) (lbm)						
		I_{sp} (s)				
	pmf	280	300	350	400	450
Circular system burnout mass	0.5	7,738	7,227	6,203	5,433	4,833
Circular system propellant mass		7,738	7,227	6,203	5,433	4,833
Payload (subtotal)		303,170	304,192	306,240	307,780	308,980
Circular system burnout mass	0.6	5,159	4,818	4,135	3,622	3,222
Circular system propellant mass		7,738	7,227	6,203	5,433	4,833
Payload		305,750	306,601	308,308	309,591	310,591
Circular system burnout mass	0.7	3,316	3,097	2,658	2,328	2,071
Circular system propellant mass		7,738	7,227	6,203	5,433	4,833
Payload		307,592	308,322	309,785	310,885	311,742
Circular system burnout mass	0.8	1,934	1,807	1,551	1,358	1,208
Circular system propellant mass		7,738	7,227	6,203	5,433	4,833
Payload		308,974	309,613	310,893	311,855	312,605
Circularization performed with Payload ONLY (lbm)						
		I_{sp} (s)				
	pmf	280	300	350	400	450
Circular system burnout mass	0.5	6,408	5,984	5,136	4,499	4,002
Circular system propellant mass		6,408	5,984	5,136	4,499	4,002
Payload		305,831	306,677	308,373	309,648	310,642
Circular system burnout mass	0.6	4,272	3,990	3,424	2,999	2,668
Circular system propellant mass		6,408	5,984	5,136	4,499	4,002
Payload		307,967	308,672	310,085	311,148	311,976
Circular system burnout mass	0.7	2,746	2,565	2,201	1,928	1,715
Circular system propellant mass		6,408	5,984	5,136	4,499	4,002
Payload		309,492	310,097	311,308	312,219	312,929
Circular system burnout mass	0.8	1,602	1,496	1,284	1,125	1,001
Circular system propellant mass		6,408	5,984	5,136	4,499	4,002
Payload		310,637	311,165	312,225	313,022	313,643

Table 6. Circularization system for 405-nmi circular orbit from the 120- × 405-nmi orbit.

Circularization Performed With Total Stack (Payload + Empty EDS) (lbm)						
		I_{sp} (s)				
	pmf	**280**	**300**	**350**	**400**	**450**
Circular system burnout mass	0.5	20,353	19,031	16,372	14,365	12,797
Circular system propellant mass		20,353	19,031	16,372	14,365	12,797
Payload		270,125	272,769	278,086	282,100	285,238
Circular system burnout mass	0.6	13,569	12,687	10,915	9,577	8,531
Circular system propellant mass		20,353	19,031	16,372	14,365	12,797
Payload		276,909	279,113	283,544	286,889	289,503
Circular system burnout mass	0.7	8,723	8,156	7,017	6,157	5,484
Circular system propellant mass		20,353	19,031	16,372	14,365	12,797
Payload		281,755	283,644	287,442	290,309	292,550
Circular system burnout mass	0.8	5,088	4,758	4,093	3,591	3,199
Circular system propellant mass		20,353	19,031	16,372	14,365	12,797
Payload		285,390	287,042	290,366	292,874	294,835
Circularization Performed With Payload Only (lbm)						
		I_{sp} (s)				
	pmf	**280**	**300**	**350**	**400**	**450**
Circular system burnout mass	0.5	16,782	15,692	13,499	11,845	10,551
Circular system propellant mass		16,782	15,692	13,499	11,845	10,551
Payload		277,268	279,448	283,832	287,142	289,729
Circular system burnout mass	0.6	11,188	10,461	9,000	7,896	7,034
Circular system propellant mass		16,782	15,692	13,499	11,845	10,551
Payload		282,862	284,678	288,332	291,090	293,246
Circular system burnout mass	0.7	7,192	6,725	5,785	5,076	4,522
Circular system propellant mass		16,782	15,692	13,499	11,845	10,551
Payload		286,857	288,414	291,546	293,910	295,758
Circular system burnout mass	0.8	4,195	3,923	3,375	2,961	2,638
Circular system propellant mass		16,782	15,692	13,499	11,845	10,551
Payload		289,854	291,216	293,957	296,025	297,642

Table 7. Disposal propellant and system mass sizing.

Disposal Performed With Empty EDS From 130-nmi Circular (lbm)						
		I_{sp} (s)				
	pmf	**280**	**300**	**350**	**400**	**450**
Circular system burnout mass	0.5	1,381	1,289	1,107	969	862
Circular system propellant mass		1,381	1,289	1,107	969	862
Circular system burnout mass	0.6	920	860	738	646	575
Circular system propellant mass		1,381	1,289	1,107	969	862
Circular system burnout mass	0.7	592	553	474	415	370
Circular system propellant mass		1,381	1,289	1,107	969	862
Circular system burnout mass	0.8	345	322	277	242	216
Circular system propellant mass		1,381	1,289	1,107	969	862

Disposal Performed With Empty EDS From 220-nmi Circular (lbm)						
		I_{sp} (s)				
	pmf	**280**	**300**	**350**	**400**	**450**
Circular system burnout mass	0.5	2,549	2,382	2,047	1,795	1,598
Circular system propellant mass		2,549	2,382	2,047	1,795	1,598
Circular system burnout mass	0.6	1,699	1,588	1,365	1,197	1,065
Circular system propellant mass		2,549	2,382	2,047	1,795	1,598
Circular system burnout mass	0.7	1,092	1,021	877	769	685
Circular system propellant mass		2,549	2,382	2,047	1,795	1,598
Circular system burnout mass	0.8	637	596	512	449	399
Circular system propellant mass		2,549	2,382	2,047	1,795	1,598

Disposal Performed With Empty EDS From 405-nmi Circular (lbm)						
		I_{sp} (s)				
	pmf	**280**	**300**	**350**	**400**	**450**
Circular system burnout mass	0.5	4,728	4,423	3,810	3,346	2,983
Circular system propellant mass		4,728	4,423	3,810	3,346	2,983
Circular system burnout mass	0.6	3,152	2,949	2,540	2,231	1,988
Circular system propellant mass		4,728	4,423	3,810	3,346	2,983
Circular system burnout mass	0.7	2,026	1,896	1,633	1,434	1,278
Circular system propellant mass		4,728	4,423	3,810	3,346	2,983
Circular system burnout mass	0.8	1,182	1,106	952	836	746
Circular system propellant mass		4,728	4,423	3,810	3,346	2,983

To provide a baseline estimate of the usable payload delivered to each circular orbit, the impacts of both the circularization and disposal systems were combined assuming a propellant mass fraction of 0.8 and an I_{sp} of 300 s.[4] These values are similar to the Shuttle orbital maneuvering system and provide a reasonable first guess for this type of system. The payload curves in figure 7 include a couple of important assumptions. First is the assumption of an EDS inert mass of 66,138 lbm (30 t), which may not include all necessary weight changes necessary to support the Mars DRM. Second, due to the use of the performance reference trajectory instead of a minimum performance trajectory, an estimated LEO performance knockdown factor of 2,645 lbm (1.2 t) is included to approximate the minimum performance trajectory. The net result is that this early estimate provides confidence that the PA-C3D configuration can deliver >125 t to LEO in support of the Mars DRM.

As part of the PA-C3′ Mars DRM assessment activity, reasonable vehicle performance growth options were captured in trade tree format for future assessment if payload mass grows beyond the performance capability of the PA-C3D configuration's capability. These growth options were not explored in detail during this study, but in most cases the thousands of conceptual designs that have been performed on the Ares V concept since ESAS have characterized the general performance increase that they could potentially provide. Figure 8 is a representative trade tree of the growth options that would be considered for increased LEO performance capability.

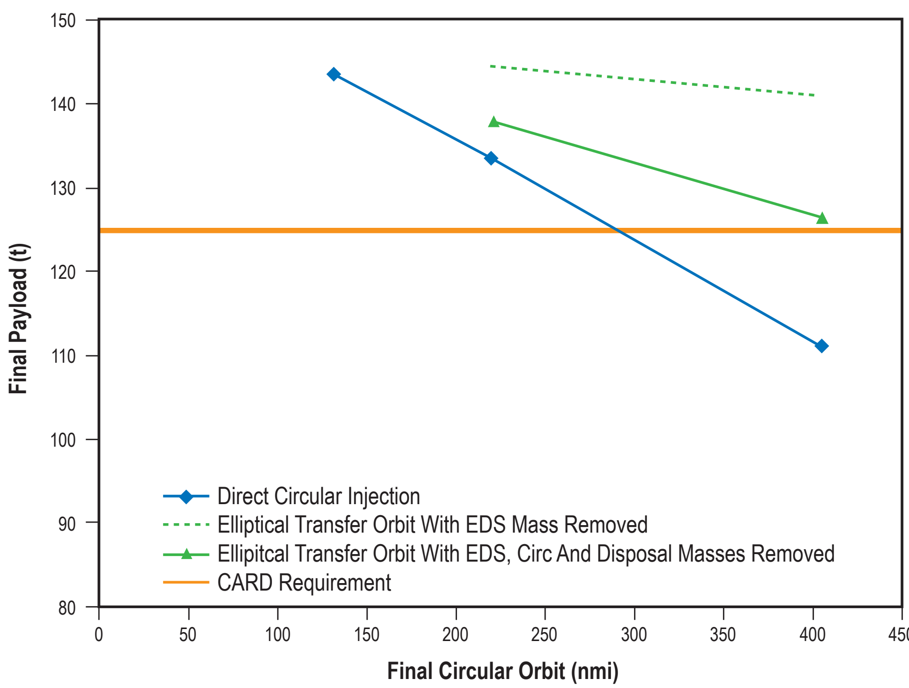

Figure 7. Representative LEO payload using Shuttle OMS assumptions. A 1.2-t knockdown assumption is included in the data.

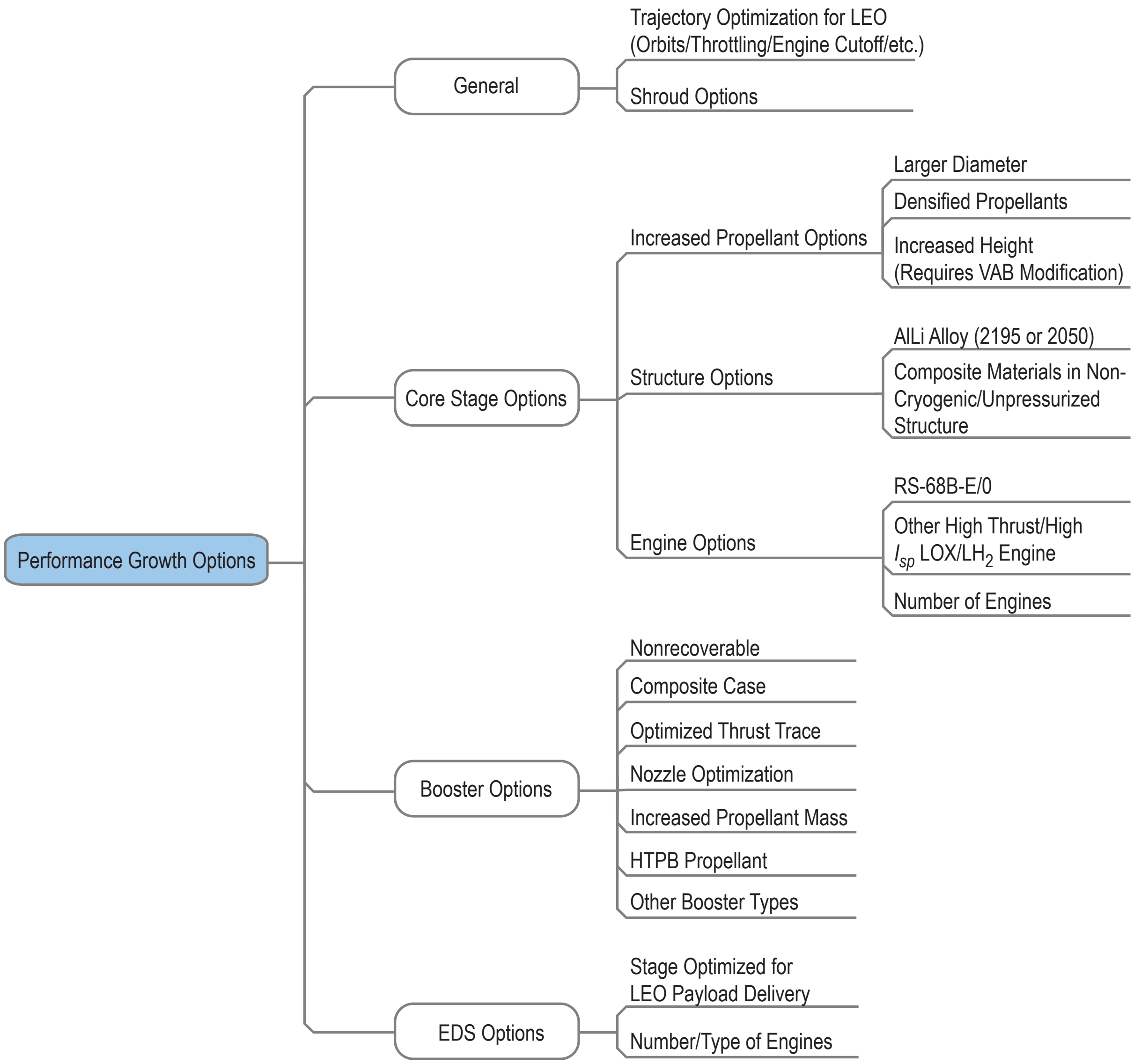

Figure 8. LEO performance growth options.

4. MARS MISSION ANALYSIS: MARS TRANSFER VEHICLE OPTIONS

As part of the initial ground rules and assumptions set at the beginning of the PA-C3′ cycle for the Mars DRM activity, it was determined that all post-TMI payloads would be set to their assessed values during the DRA 5.0 study and follow-on activities. This locked the general parameters for the main activity during the 7-wk assessment period of PA-C3′, which dealt with understanding the requirements of the Ares V vehicle to launch and possibly maintain the main propulsion system for the TMI maneuver (and perform the TMI maneuver if necessary). Two main propulsion system types were assessed for applicability to the TMI maneuver for the architecture presented in DRA 5.0: NTP and chemical propulsion (liquid oxygen/liquid hydrogen (LOX/LH_2)). Figure 9 shows a general breakdown of the options considered throughout PA-C3′.

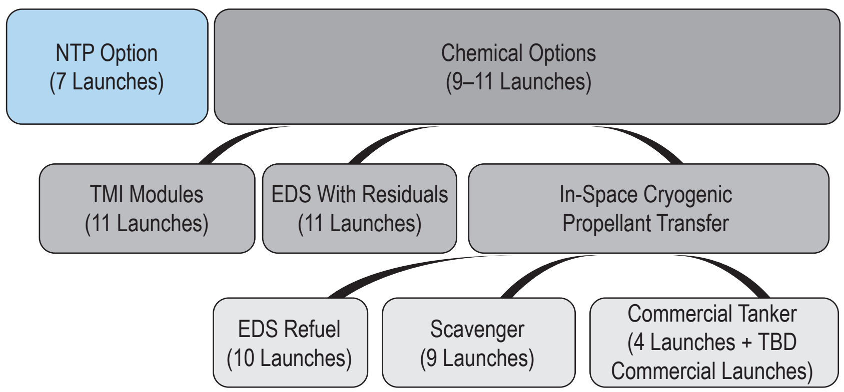

Figure 9. TMI stage propulsion options assessed during PA-C3′.

As the launch count required for the various options was assessed, the “carry-forward” options were further condensed to depict the most attractive chemical propulsion options from a launch count and lunar-EDS evolution perspective. As shown in figure 10, these include both “minimum Ares V participation” and “maximum Ares V participation” NTP options and a chemical propulsion option that utilized in-space cryogenic propellant transfer and its derivative that relied on commercial vendors to supply the needed propellant. These four options will be the focus of this TM.

4.1 Nuclear Thermal Propulsion Option

The NTP option is considered the reference approach for the DRA 5.0 architecture, and it will be referenced as a specific implementation of that technology based on DRA 5.0 in this TM (engine type, engine performance parameters, propellant/stage mass assumptions, subsystem assumptions, etc.).[5] The general launch sequencing is depicted in figure 11.

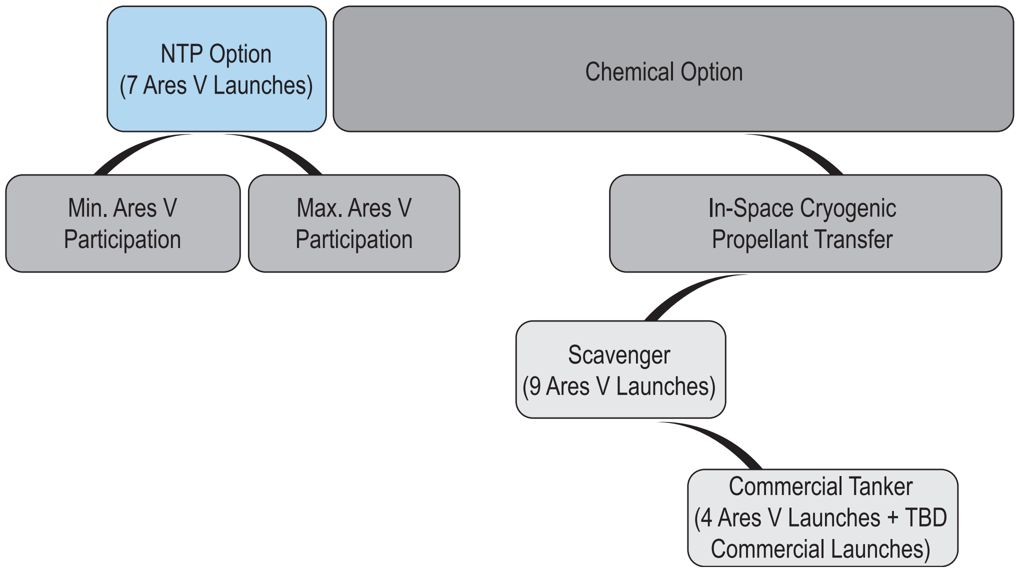

Figure 10. TMI propulsion options discussed.

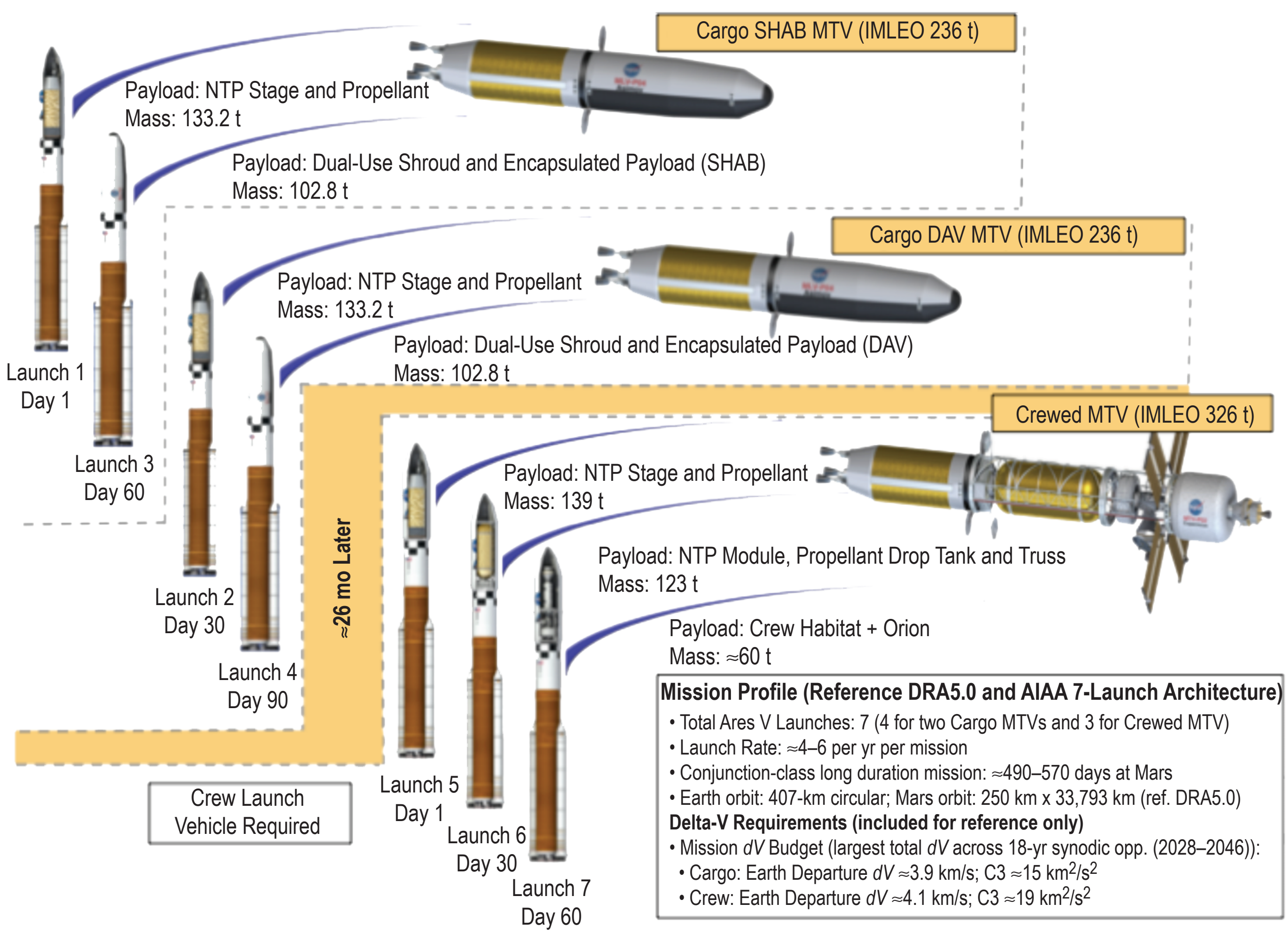

Figure 11. NTP launch sequence and payload mass requirements.

Delivering twice the I_{sp} of its chemical propulsion counterparts, nuclear propulsion systems significantly reduce the mass injected to LEO requirements of the Ares V launch system. This increase in operating efficiency of the system and the resulting reduction in mass required allows for the same payload to be injected to Mars with a reduced number of Ares V launches because a larger percentage of the in-space mass is payload rather than the propellant required to send that payload to its destination. The general requirements of the Ares V launch system were divided into two main categories for this MTV option: (1) Minimum Ares V participation and (2) maximum Ares V participation.

While either the minimum participation or maximum participation would have a significantly different implication for the requirements on the Ares V launch vehicle, either option would fit within the mission architecture depicted in figure 11.

4.1.1 Minimum Ares V Participation

The first scenario for delivering the required payloads and propulsion elements for the NTP option to LEO involves utilizing the Ares V EDS in much the same way as the Ares I upper stage is utilized for the lunar DRM. The EDS would only be required to deliver the large payloads to an elliptical orbit (exact orbit to-be-determined) and require the payload and/or auxiliary systems added to the payload to provide all of the on-orbit functionality required to deliver the payload to the ultimate loiter/assembly orbit (circularization maneuvers, rendezvous/proximity operations maneuvers, etc.), maintain the payload for the entire loiter/assembly duration (power, ACS, etc.), perform any functionality required prior to the TMI maneuver (micrometeoroid and orbital debris (MMOD) avoidance, reboost, control moment gyroscope (CMG) desaturation if applicable, etc.), and ultimately perform the TMI maneuver and the ensuing NTP stage disposal maneuver. In this scenario, the Ares V EDS is assumed to function much like a traditional payload-to-LEO delivery service. In addition, it would be an option for the EDS to perform some, if not all, of the circularization maneuver and its own jettison and disposal maneuvers, but that assumption was not included in the minimum participation scenario.

A representative LEO delivery sequence is shown in figure 12. In this notional assembly sequence, the Ares V EDS (or second stage) is shown delivering the NTP element of the cargo MTV to an elliptical orbit on the first launch followed by the dual-use shroud with encapsulated payload on the second launch. In this depiction, the orbit is assumed to be a stable phasing elliptical orbit so the payload will have several orbits for potential contingency operations should they be needed. The EDS is jettisoned at the earliest possible convenience, and it performs a disposal maneuver for reentry, eventually breaking up and splashing down over an open body of water. The expected EDS lifetime for both cases is ≤5 days.

Where the cargo MTVs consist of two Ares V launches each, the crewed MTV requires three Ares V launches followed by an Ares I (CLV) to deliver the crew to the assembled crewed MTV. A notional assembly sequence for this mission is depicted in figure 13. The NTP element is delivered first, then the truss structure/LH_2 “drop tank” and about 94 t of LH_2, and then the TransHab with contingency consumables, associated canister, and docking mechanism.[5] Again, the EDSs that are responsible for delivering the crewed MTV components into a stable phasing elliptical orbit have an expected lifetime of ≤5 days.

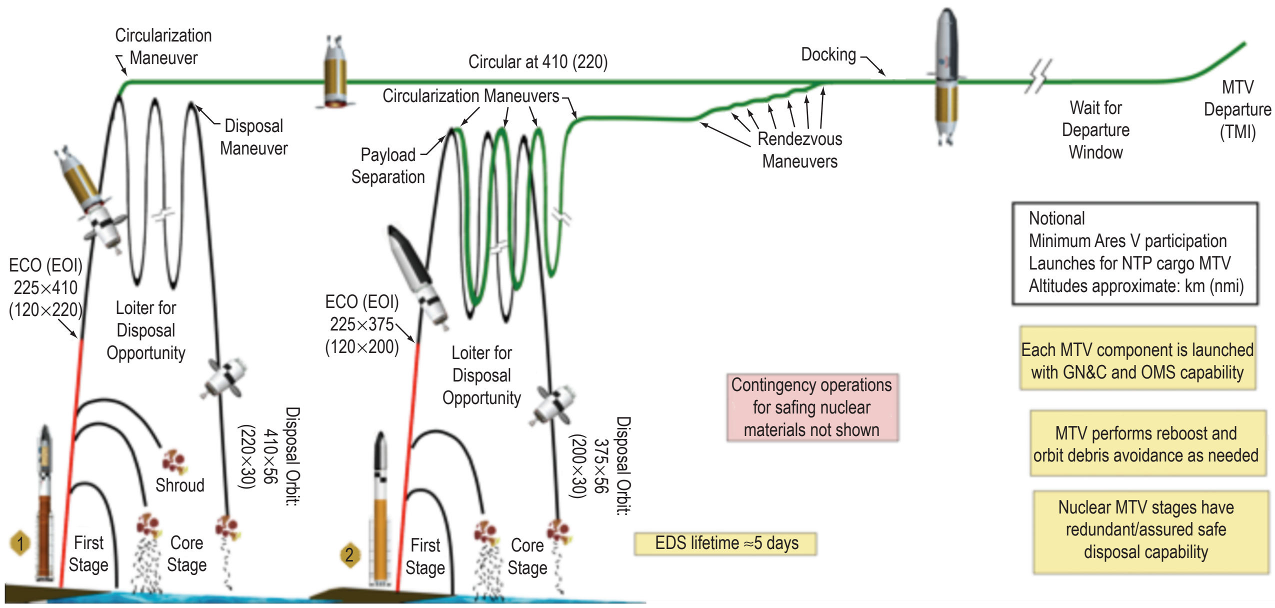

Figure 12. Ares V payload to LEO delivery sequence (minimum participation, NTP cargo).

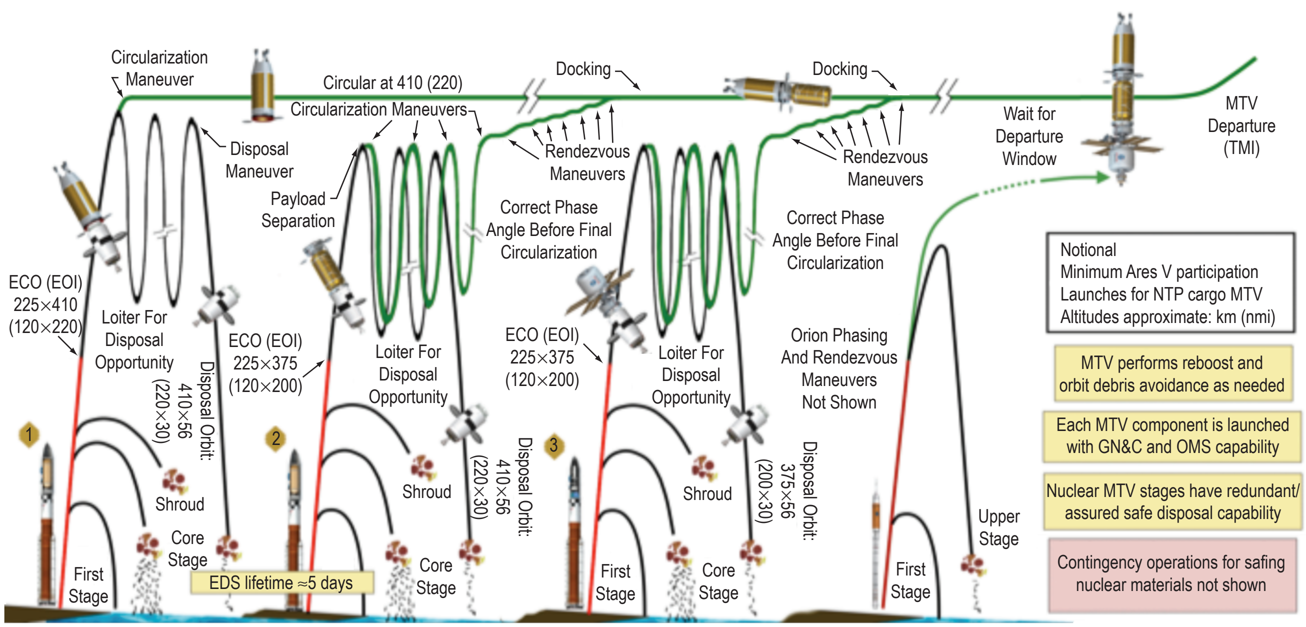

Figure 13. Ares V payload to LEO delivery sequence (minimum participation, NTP crew).

There would be additional opportunities to further optimize the delivery orbit for this scenario; however, the stable phasing elliptical chosen for this case was selected because of the following reasons:

- The MTV is a high-value National asset. Inserting into an orbit with a 45-minute lifetime (negative perigee for ensured EDS reentry on first pass) allows little time to debug and repair a failure (separation problems, failure of the MTV's orbital maneuvering system (OMS) system, failure of the MTV's avionics, etc.). Inserting into a stable orbit allows time to repair many of the possible failures.
- Inserting into a 130-×220-nmi orbit versus a 30-×220-nmi orbit saves significant dV on the behalf of the auxiliary propulsion systems of each of the MTV components.
- EDS is a fully capable stage. No new capability is required above those required for the lunar DRMs and deorbit from LEO is a lunar requirement for contingencies.

It is also fairly evident from looking at these depictions that it may be in the best interest of the total system architecture for the EDS or some other auxiliary propulsion stage to perform some, if not all, of the circularization maneuver; rendezvous, proximity operations, and docking (RPOD) maneuvers; and perhaps some on-orbit functionality in order to decrease the requirements levied on the payload elements (and decrease mass at TMI). This thought process led to the development of the maximum Ares V participation approach.

4.1.2 Maximum Ares V Participation

The other scenario explored for delivering NTP components to LEO was termed the maximum Ares V participation option. Under this scenario, the Ares V EDS and added auxiliary propulsion systems would be responsible for delivering the payloads to their final loiter/assembly orbit. In addition, the EDS was assessed for its ability to add some on-orbit functionality that only applied during the LEO assembly operation phase of the mission architecture and was not needed after the TMI maneuver. These types of functions (and some contingency capabilities) seemed best placed on the EDS so they could be provided by that stage, but the mass associated with providing those functions could be jettisoned along with the EDS before the TMI maneuver was performed—increasing overall mission efficiency. These functions include final orbit insertion/circularization, RPOD functionality, ACS/RCS during rendezvous if applicable, MMOD avoidance maneuvers if applicable, CMG desaturation, and its own jettison/disposal functionality.

A general assembly sequence for the cargo case of this option is depicted in figure 14. In this case, the first Ares V is responsible for delivering the NTP TMI propulsion stage for the cargo mission to LEO. Two of these are delivered for the two cargo MTVs (as shown in fig. 11). The second Ares V (for this cargo MTV) is responsible for delivering the dual-use shroud with encapsulated payload to the final assembly orbit. Once assembled, the TMI maneuver is performed to inject the payload on a transfer to Mars.

It can be seen in this notional case, that the EDS is assumed to inject the NTP stage into a stable phasing elliptical orbit and then auxiliary propulsion systems added to the EDS perform the circularization maneuver at the desired altitude (currently assumed to be 20 nmi less than the final

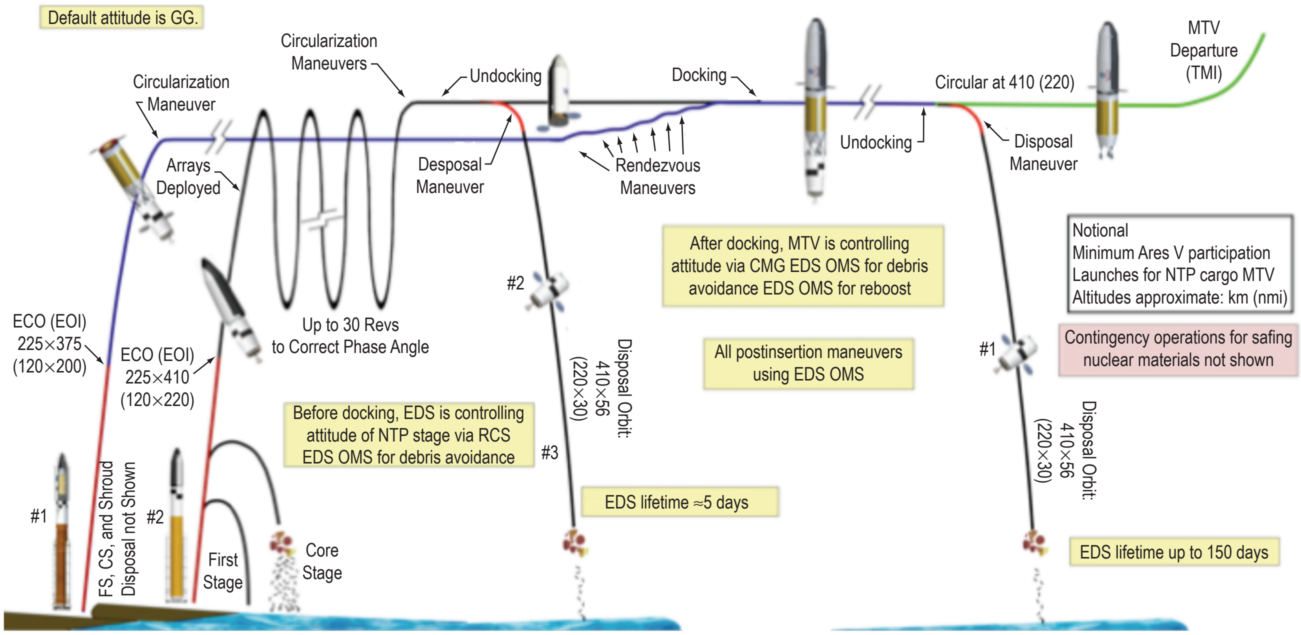

Figure 14. Ares V payload to LEO delivery sequence (maximum participation, NTP cargo).

assembly altitude). Furthermore, the second Ares V launch delivers the dual-use shroud and encapsulated payload to a stable phasing elliptical orbit. This payload and EDS are maintained in the orbit until the appropriate phase error reduction is achieved between it and the NTP stage delivered on the first launch. When that reduction in phase angle is realized, the auxiliary propulsion system performs the circularization maneuver at the desired loiter/assembly altitude (currently assumed to be a 220-nmi circular orbit), at which time the EDS is discarded.

The auxiliary propulsion systems on EDS1 (delivered with the NTP stage) are responsible for performing the gradual bumps in altitude required to rendezvous with the payload at the final assembly altitude. Finally, after docking is achieved between the NTP stage and the dual-use shroud/payload, the EDS is responsible for maintaining the loiter altitude (pre-TMI), any MMOD avoidance maneuvers that may be needed, and perhaps CMG desaturation if used to maintain attitude on orbit. Overall, these types of LEO functions are best placed on a system that has the ability to be jettisoned and disposed of before the TMI maneuver is performed (the EDS in this case). In that manner, the mass required to perform those functions is discarded to increase overall mission efficiency.

The crewed MTV assembly sequence is shown in figures 15–17. The same types of functions are also provided for that particular assembly sequence.

4.2 Chemical Propulsion Options

While benefits in mass requirements in LEO because of increased operational efficiency have been shown for the NTP option, some chemical propulsion options also offer advantages.

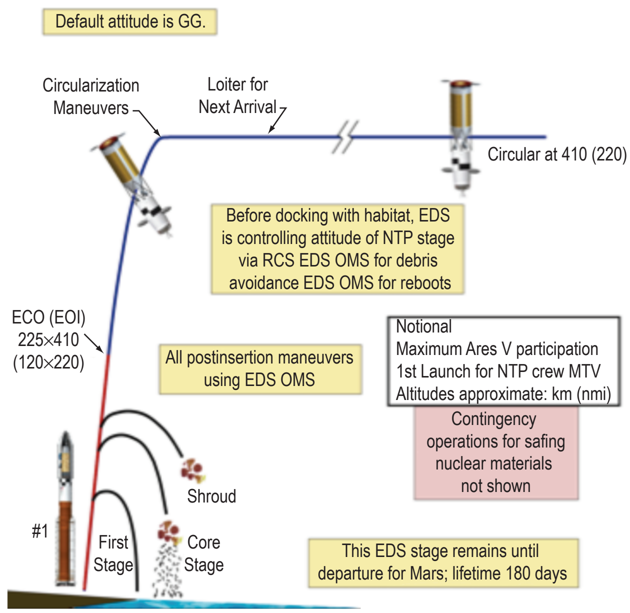

Figure 15. Ares V payload to LEO launch No. 1 (maximum participation, NTP crew).

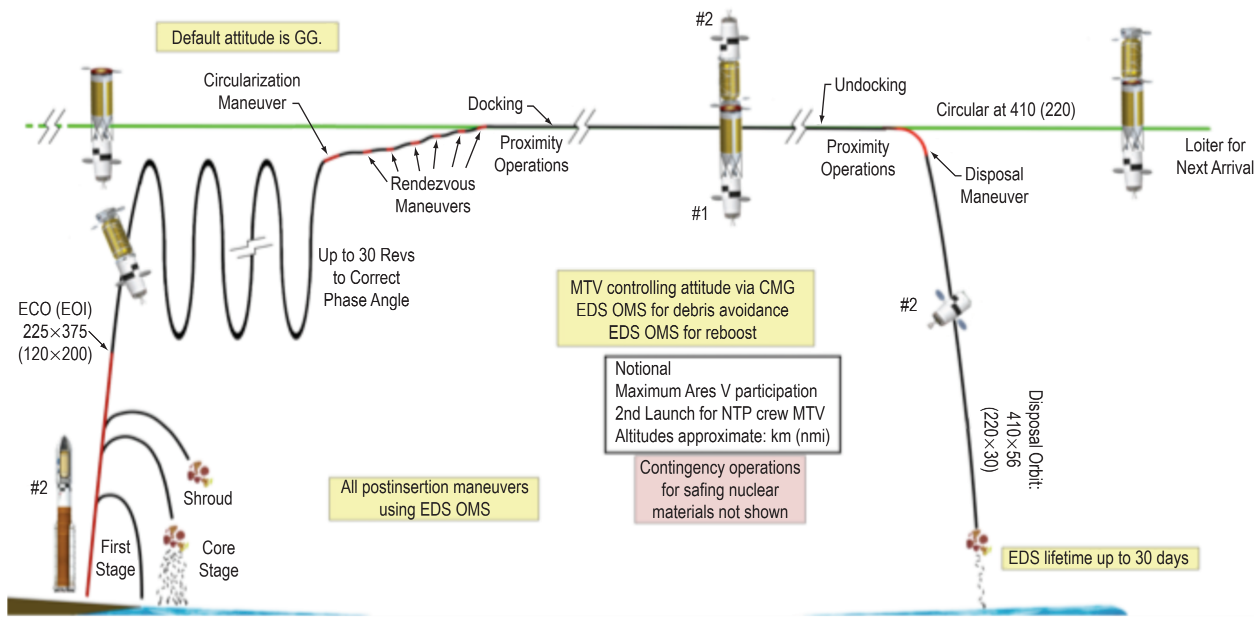

Figure 16. Ares V payload to LEO launch No. 2 (maximum participation, NTP crew).

The two that were studied in detail during PA-C3′ and carried forward for further analysis both involve cryogenic propellant transfer in LEO but provide maximum utilization of a lunar-EDS derived, in-space transportation system architecture. However, another option was also assessed

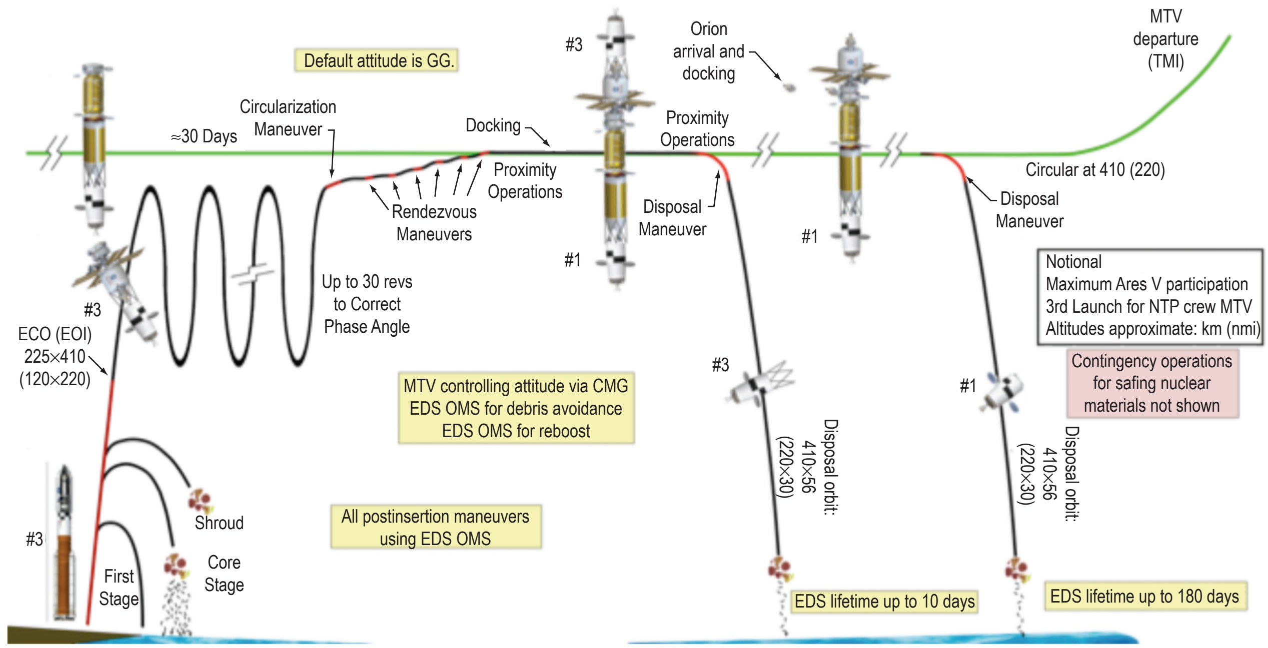

Figure 17. Ares V payload to LEO launch No. 3 (maximum participation, NTP crew).

for comparison purposes, which was presented in much more detail in the DRA 5.0 final report. This was the chemical propulsion module option discussed below.

4.2.1 Chemical Propulsion Modules Assessed for Mars Design Reference Architecture 5.0

In developing the cargo and crewed MTVs, separate propulsive elements were identified for each major mission maneuver: TMI, MOI, and TEI. All TMI modules are meant to retain commonality among the cargo and crewed MTVs, and each module utilizes five RL10-B2 engines and is jettisoned after performing its burn. The MOI modules use two RL10-B2 engines for the all-propulsive crew and cargo missions. TEI modules similarly use two RL10-B2 engines and also perform plane changes while in Mars orbit.

The two cargo MTVs require two TMI modules each (20 total RL-10s), as specified in DRA 5.0. The crewed MTV employs multiple propulsive stages, namely three TMI modules (15 RL-10s), one TEI module (2 RL-10s), and one MOI module (2 RL-10s). With three TMI modules required for each crewed MTV, a two-burn TMI maneuver is necessary; therefore, the two outer stages perform the first portion of the TMI burn (and then are jettisoned), followed by the TMI module located in the center providing the remaining *dV* necessary to place the crewed MTV elements into a trans-Mars coast.

This approach and the associated number of propulsion modules that were assumed to be delivered to LEO requires an 11-launch architecture with an additional crew launch vehicle to deliver the crew into LEO once the crewed MTV is assembled and the TMI window is open.

While this option is detailed more fully in the Mars DRA 5.0 final report and addendum, analysis by the Ares V team of this mission architecture led to alternative in-space transportation system approaches (while still using traditional chemical propulsion system approaches) that were found to reduce the number of launches, total mass to orbit requirements, and overall cost and complexity of developing dedicated in-space transportation systems. The two most competitive options were deemed the propellant tanker option (also called the scavenger option) and the derivative commercial tanker option.

For comparative purposes, the launch and assembly sequence for an approximate DRA 5.0 chemical propulsion module case is depicted in figure 18.

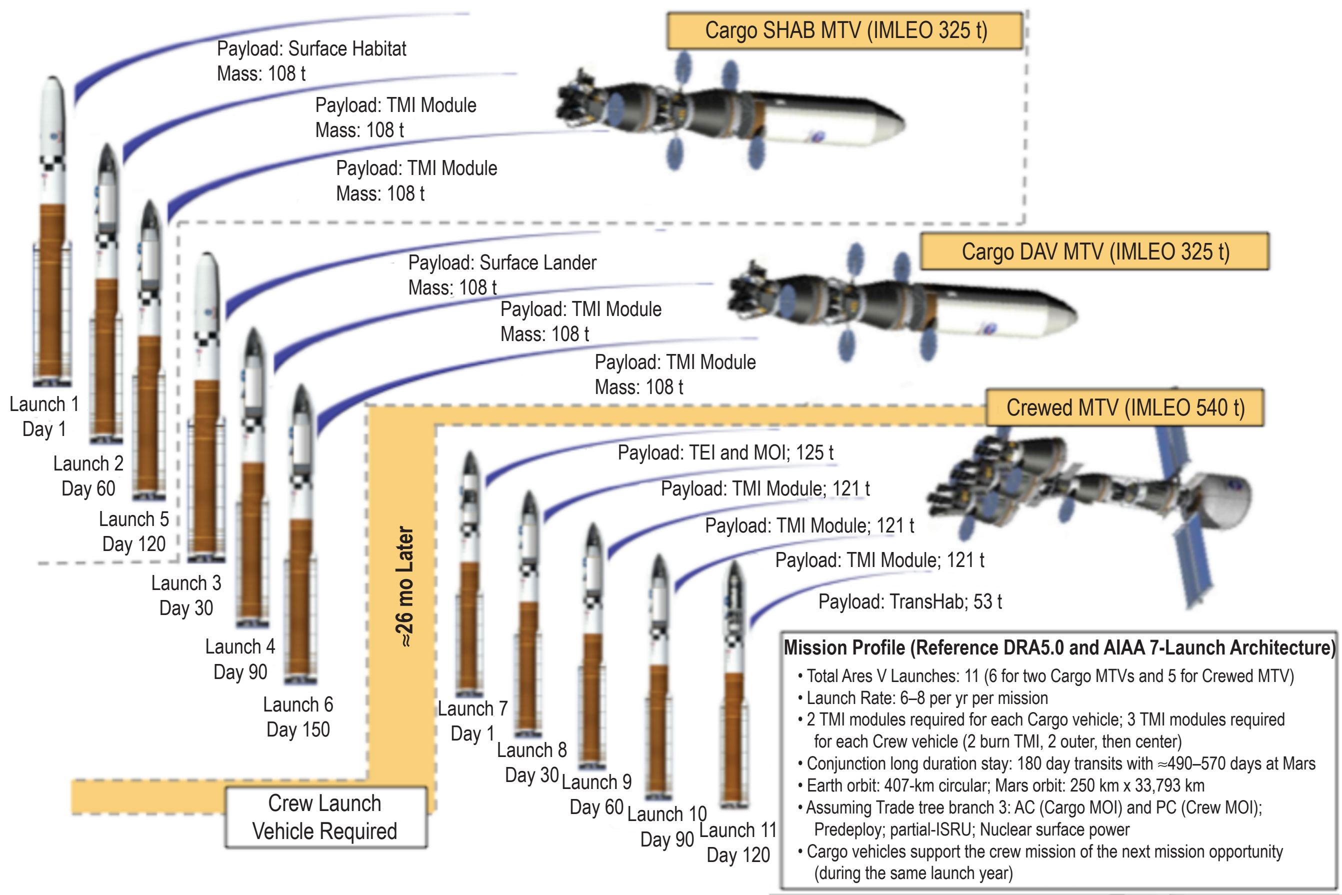

Figure 18. DRA 5.0 chemical propulsion module option.

4.2.2 Earth Departure Stage as Mars Transfer Vehicle Trans-Mars Insertion Stage: Propellant Tanker With In-Space Propellant Transfer

While the chemical propulsion module option found that a dedicated in-space RL-10-derived stage may better serve agency needs for a chemical propulsion option for sustained human exploration of Mars, a less costly, less complex, and potentially more reliable approach may involve also utilizing the same assets for the Mars exploration campaign that are to be built and operated for a decade or more for the lunar campaign. The main advantage for a chemical propulsion system with an EDS heritage would be a clear evolution path for the lunar EDS when applied to the Mars DRM and the shared mission commonality which that would entail (in the case of concurrent lunar and Mars missions for instance or for increased reliability confidence). This extensibility would result in a much greater return-on-investment in the lunar exploration elements that would otherwise be underutilized for the Mars campaign. In addition, the cost of the ETO and in-space transportation elements would be reduced to the needed technological investments above-and-beyond those planned for development during the lunar missions. This EDS-derived propellant tanker option was given the informal moniker scavenger option due to the fact that all available performance up to the CARD requirement of 125 t was used for either payload, propellant, docking and propellant transfer hardware, or a combination thereof. The propellant tanker option consists of 9 Ares V launches with an option to grow to 10 or 11 to add greater margin to the architecture, if required. This is reasonably competitive with the seven-launch NTP architecture as proposed in DRA 5.0 in terms of total Ares V launches required to carry out the proposed mission. It is currently assumed that both will require an additional CLV to deliver the crew once the crewed MTV is assembled in LEO.

As shown in figure 19, the first five launches deliver components for the two cargo MTVs. In this scenario, the first and second Ares V launches deliver the dual-use shrouds with encapsulated payloads (102.8 t each) for the cargo MTVs, along with the EDS that will perform the TMI maneuver for each, which is assumed to have a small amount of propellant remaining in the tanks (currently assumed to be ≈22 t of LOX/LH_2 propellants) once inserted into the LEO assembly orbit. The third and fourth launches deliver ≈120 t of propellant (in other EDS modules) that is transferred to the 22 t of residual propellant from the first EDS TMI stage (for each cargo MTV). The third and fourth launch EDSs are then jettisoned and disposed of. These launches result in about 142 t of total propellant loaded in the EDS TMI stage. A total of 192 t of propellant is needed to meet the *dV* requirement for the cargo mission (with current assumptions for stage mass, payload mass, *dV*, etc.), so a fifth and final launch acts as a split-tanker stage EDS that divides 100 t between the two cargo MTVs, (50 t allocated to each). Obviously, this shared launch has more performance capacity so it would also be able to deliver on the order of 10 t of additional propellant to serve as margin or decrease sensitivity to boiloff assumptions made for this assessment. Furthermore, instead of sharing a launch between the two cargo MTVs, a dedicated second propellant delivery launch could be planned to provide much more propellant to each MTV to either decrease the need for zero boiloff (ZBO) and/or provide margin for the propellant transfer functionality. Once this EDS meets with and tanks 50 t to each cargo MTV, it is jettisoned and disposed of. The net result of these five Ares V launches is two cargo MTVs assembled in LEO, each comprised of a 102.8-t, dual-use, shroud and encapsulated payload and a single EDS loaded with 192 t of LOX/LH_2 propellant that will perform the TMI maneuver.

Also shown in figure 19, the crewed MTV requires four Ares V launches (with current assumptions). The first launch delivers the TEI and MOI stages (assumed at 125 t from previous analysis) and has fully exhausted all propellant in the EDS en route to LEO. The second launch delivers a TransHab, small truss structure and contingency consumables to LEO, in total ≈50 t.

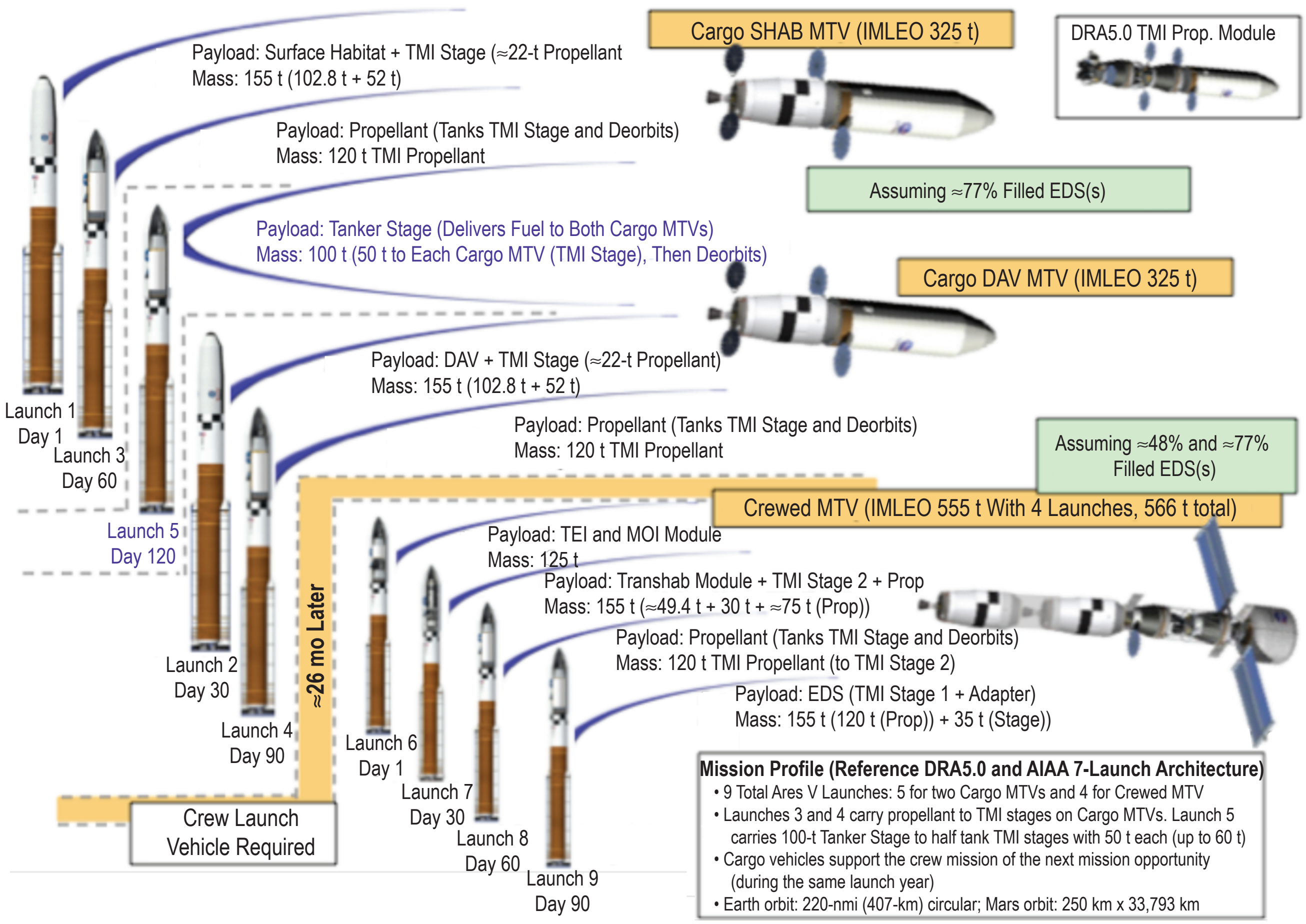

Figure 19. Propellant delivery and transfer option (EDS-as-MTV).

This leaves an ≈75 t total performance capability that would otherwise be unutilized. Therefore, this second launch is assumed to also deliver an EDS that has a residual propellant load of 75 t, which serves as the second stage of the two-burn TMI maneuver for the crewed MTV. The third launch is a dedicated propellant launch, which transfers 120 t of propellant to the previously 75 t filled EDS. This results in about 195 t of LOX/LH_2 propellants in that stage. Once the 120 t of propellant is transferred, the empty EDS is jettisoned and disposed of. The fourth launch brings up the first stage of the two-burn TMI maneuver and appropriate docking hardware to connect to the other EDS. It is currently assumed that the docking hardware will be on the order of 5 t, which leaves ≈120 t of capability for delivering propellant on this launch. That gives a total of 120 t in the first TMI stage EDS and 195 t in the second TMI stage EDS resulting in 315 t of total LOX/LH_2 propellants (still slightly short of the dV requirement but close for this fidelity analysis).

Another option for the crewed MTV is a dedicated propellant tanker for the first TMI stage. It is currently assumed that it will be delivered with 120 t of propellant, which is less than 50% of the EDS tank capacity. Another launch would provide up to 120 t of additional propellant to this EDS in order to increase the dV capability of the assembled stack, provide margin for the transfer of propellant in-orbit, and/or decrease the need for a ZBO thermal control system.

Figures 20–23 depict additional details on this launch and assembly sequence for both cargo MTV cases. Each cargo MTV would be assembled over a period of months (current assumption is 30 days between Ares V launches). The first component that would be delivered is the dual-use shroud with encapsulated payload, as shown in figure 20. This payload is injected into the final assembly/loiter orbit, and the associated EDS for this launch will serve as the platform for receiving the propellants that will be delivered on later Ares V launches.

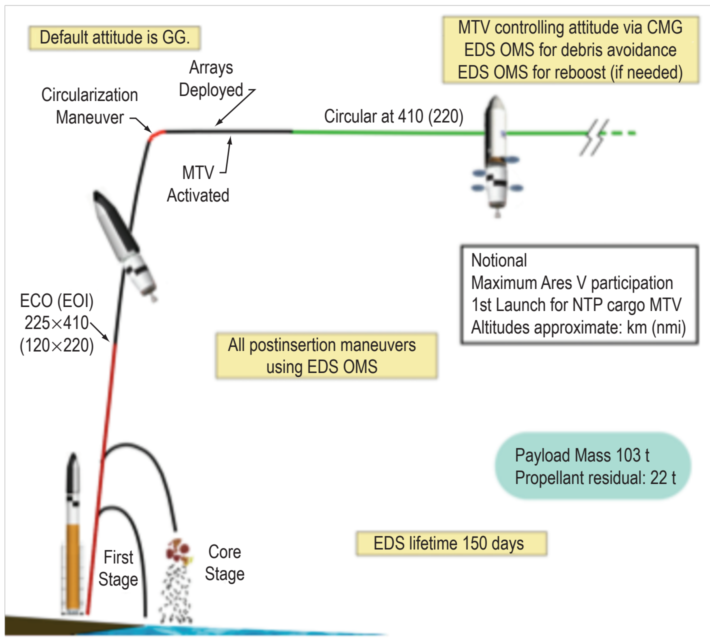

Figure 20. Dual-use shroud/encapsulated payload launch (chemical cargo).

Figure 21 shows the first dedicated propellant tanker launch. The assumption depicted for delivering the maximum propellant to orbit is a tank set located above the EDS that also functions as the outer-mold line (OML) of the vehicle (similar to the Space Transportation System (STS) external tank). This is done in order to decouple the performance capability of the vehicle from the actual geometric constraints of the EDS tanks. There are other methods for delivering propellant to LEO including loading the EDS tanks to 100% capacity and burning the optimum amount of propellant to deliver the maximum amount of EDS residual propellant to orbit (no payload forward of EDS, only a nose cone) or using an encapsulated tank set (traditional shroud as the OML) that may be the optimum solution for constructing and delivering propellant transfer hardware to orbit. Most of these details require much more in-depth analysis. Figure 22 shows the shared tanker launch.

The key feature of this shared propellant launch (shown in figures 22 and 23) is its ability to rendezvous and dock with both cargo MTVs on-orbit. Therefore, it is assumed that it will be launched into a phasing elliptical orbit, rendezvous/dock/transfer propellant to the first cargo MTV, then jettison and transfer to a lower orbit in order to phase appropriately to the second

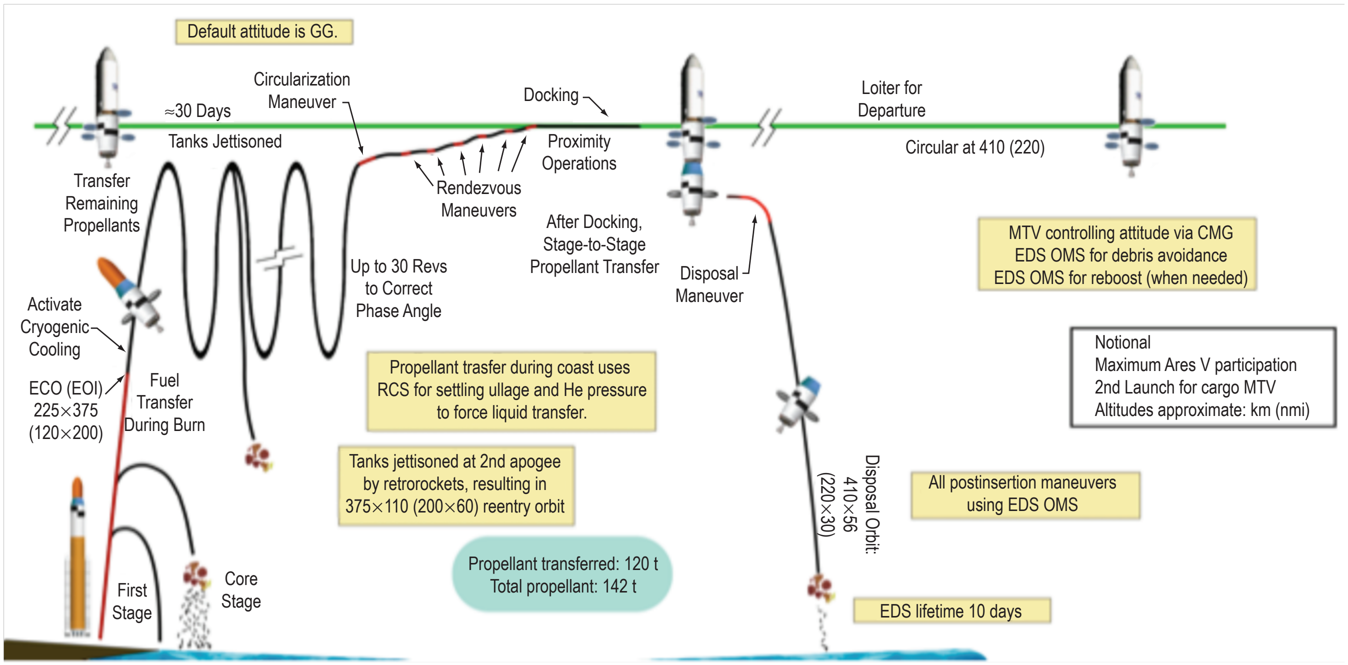

Figure 21. Propellant launch for cargo MTV (first chemical cargo).

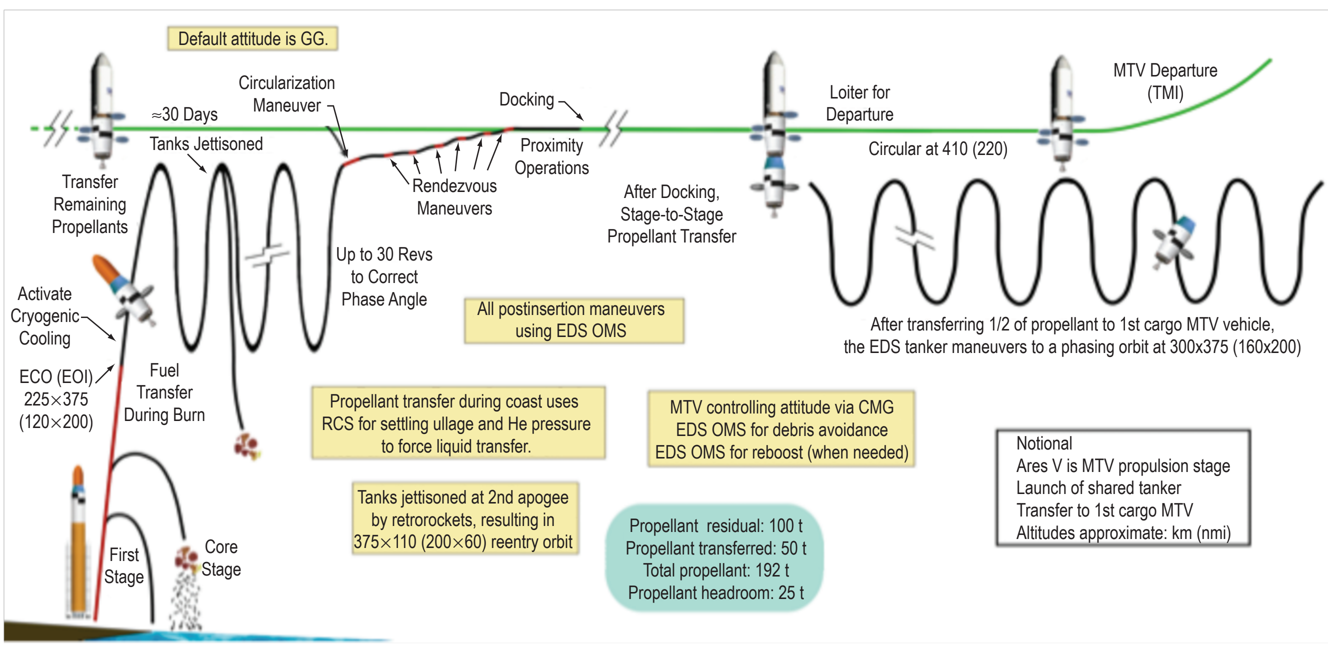

Figure 22. Shared propellant launch for first MTV (second chemical cargo).

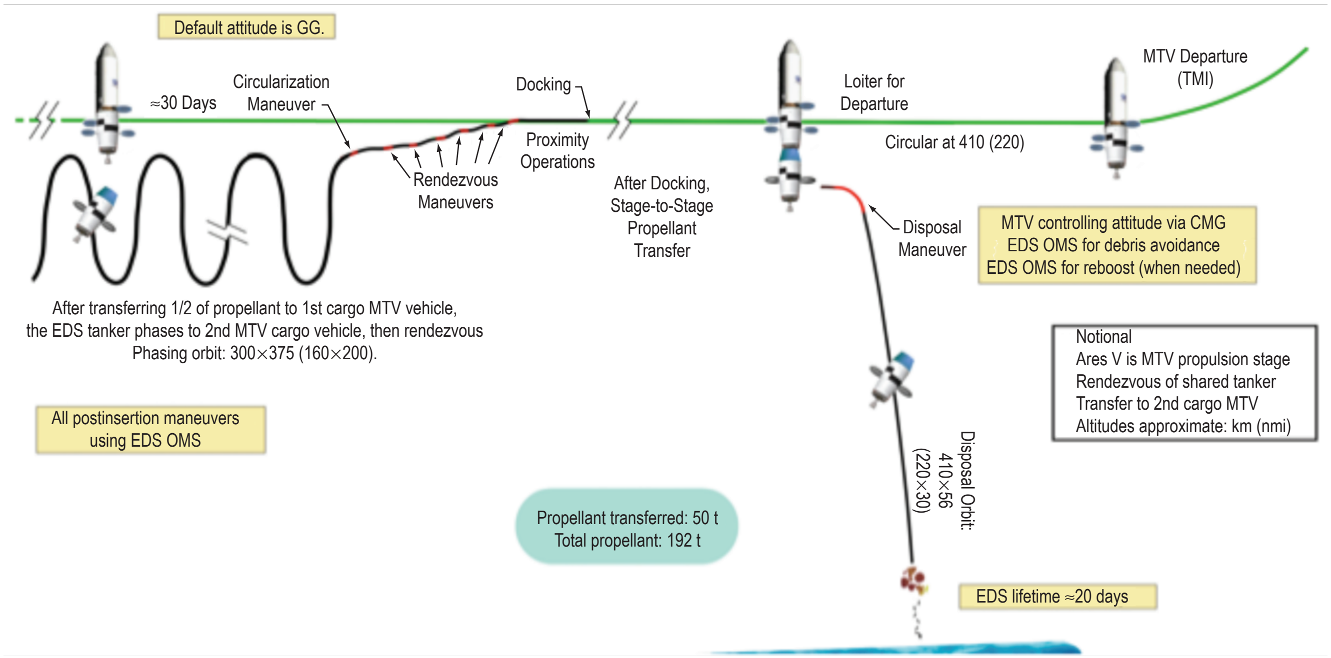

Figure 23. Shared propellant launch for second MTV (two and one-half chemical cargo).

cargo MTV. In this manner, the auxiliary propulsion system will have to be sized appropriately to perform these additional in-space maneuvers, including any changes in the other orbital elements of the vehicle.

Figures 24–27 show the assembly sequence currently assumed for the crewed MTV. The first launch (depicted in fig. 24) is the MOI/TEI chemical propulsion stages, which are delivered to the final assembly/loiter orbit. Subsequence launches are delivered to this orbit.

The second launch of the crewed MTV assembly sequence is depicted in figure 25. This launch delivers the crew TransHab (a small truss structure that connects to a contingency consumables canister and provides the aft interface for docking with the MOI/TEI modules delivered on the first launch) and the consumables located in the canister that provides provisions for the mission duration in the case of the crew not being able to descend to the surface of Mars. The summation of all of these components is less than 50 t; therefore, the EDS is loaded with an additional ≈75 t of LOX/LH_2 propellant. Once the payload is successfully docked with the MOI/TEI modules, the EDS is undocked and redocks at the aft end of the vehicle to serve as the second stage for the TMI maneuver.

The third launch (fig. 26) depicts the delivery of a dedicated tanker launch for the second TMI stage. Figure 27 shows the delivery of the first TMI stage and the CLV delivery of the crew to the completed crewed MTV stack.

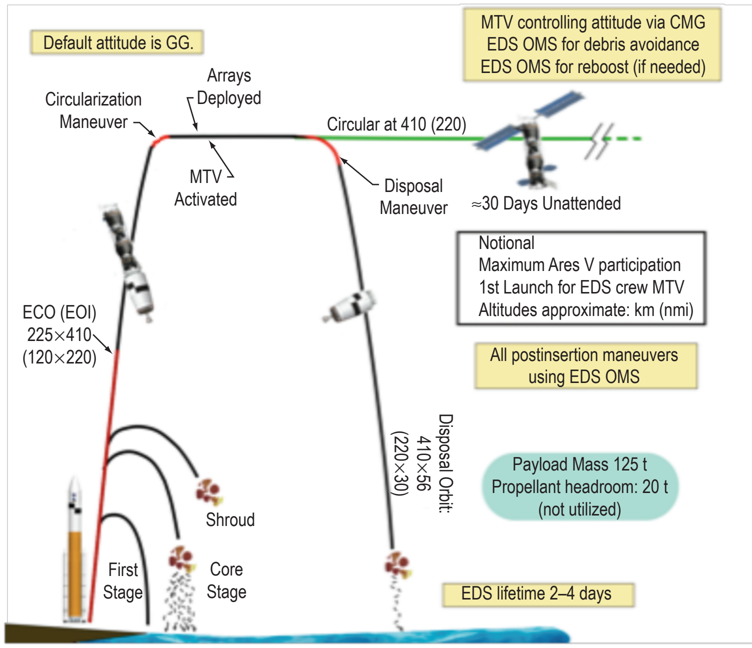

Figure 24. MOI/TEI module launch, first launch (first chemical crewed).

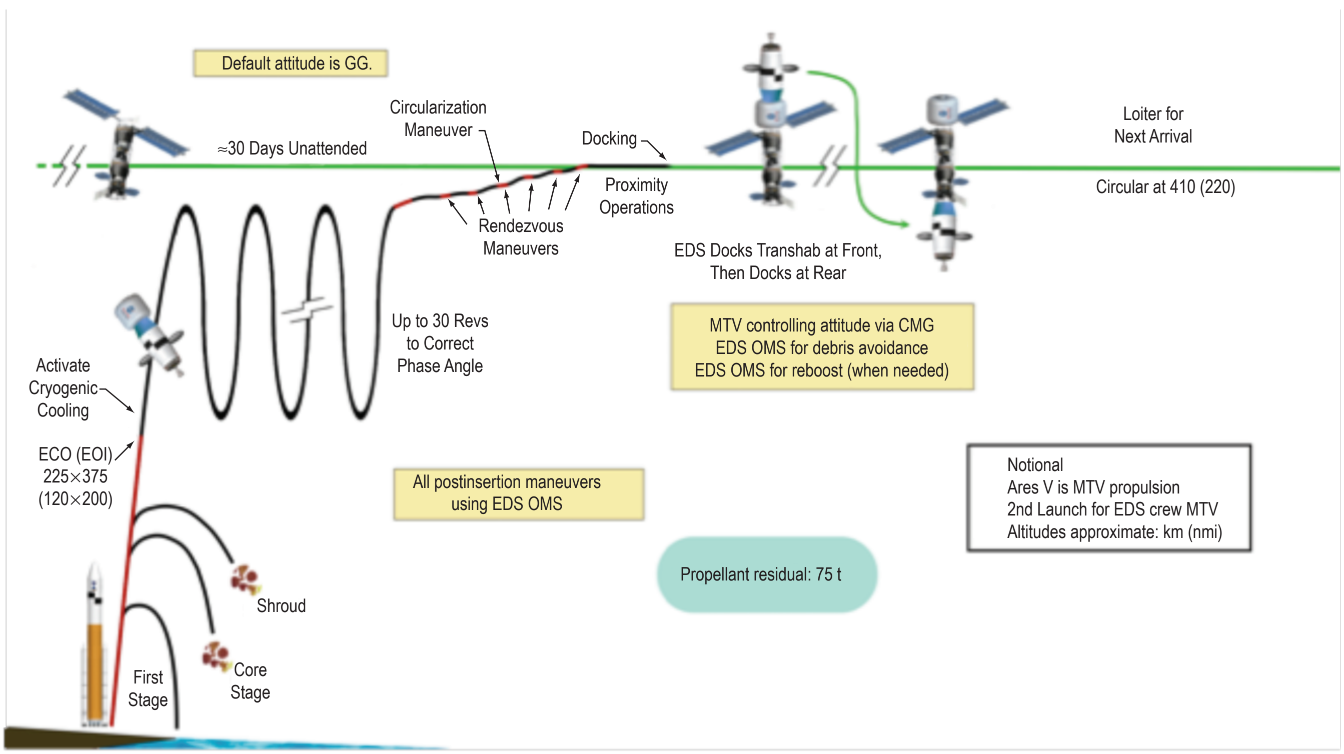

Figure 25. TransHab/truss/consumables/‘second TMI stage,’ second launch (second chemical crewed).

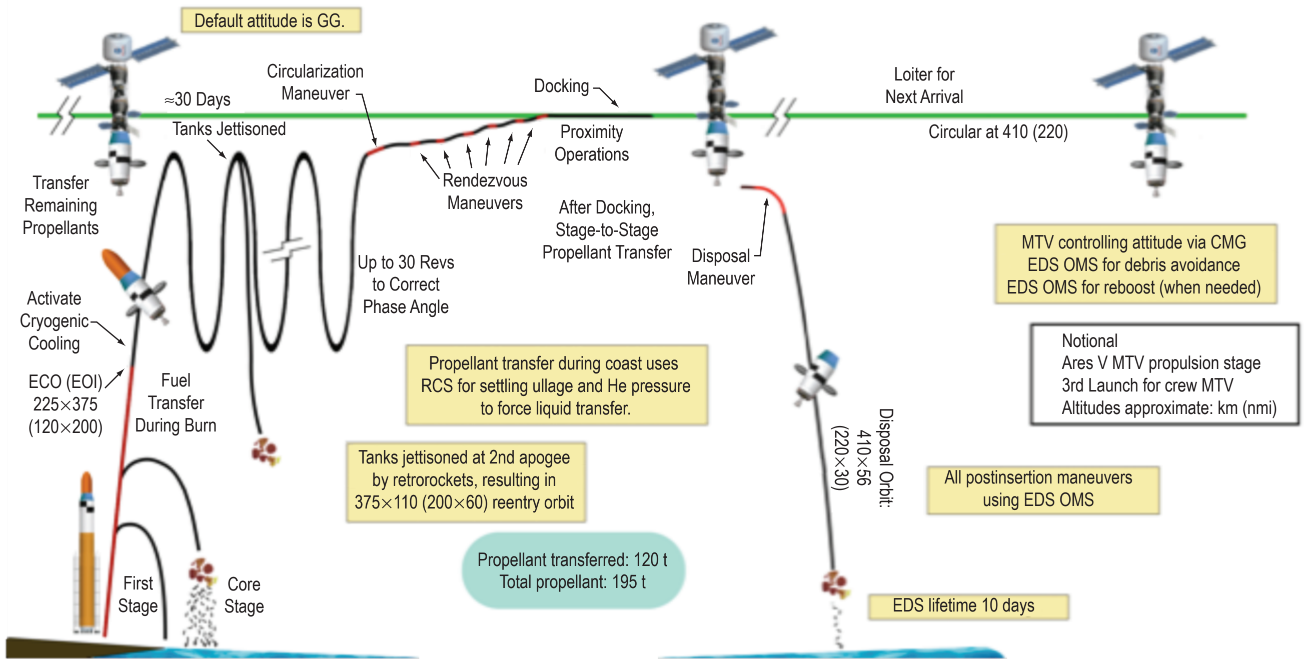

Figure 26. Propellant launch for ‘second TMI stage’ (third chemical crewed).

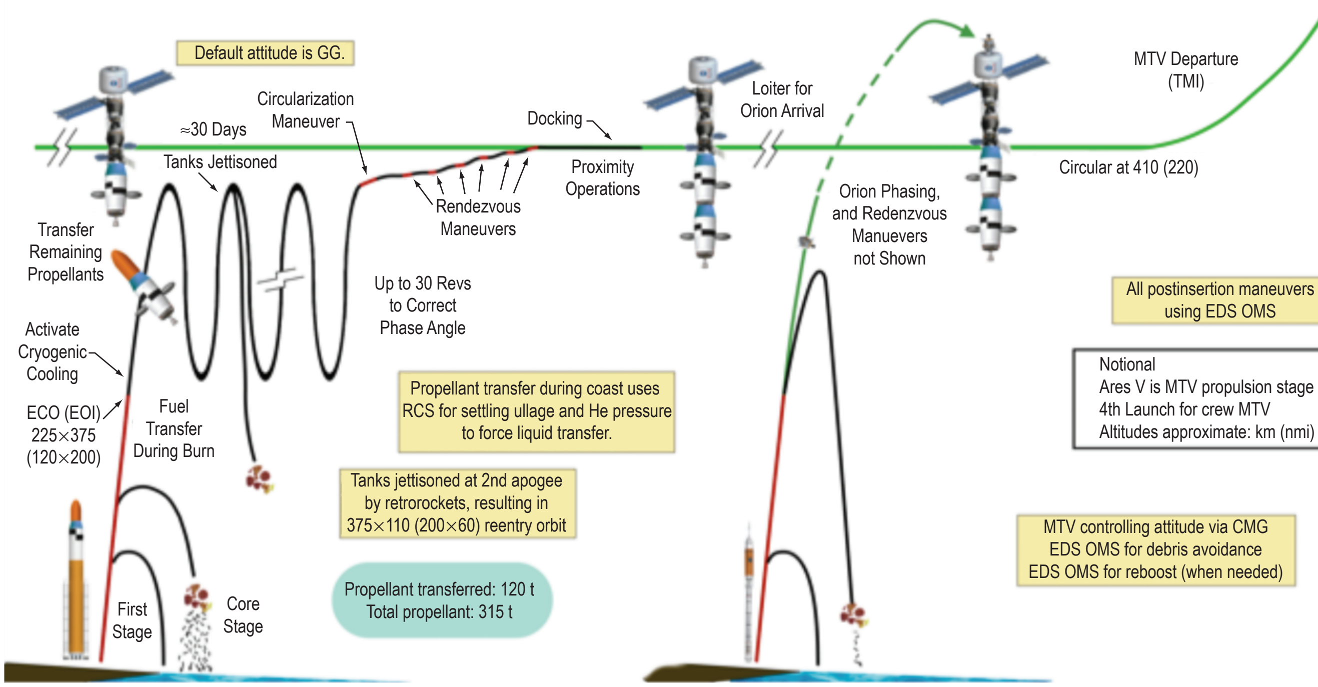

Figure 27. ‘First TMI Stage’ launch (fourth chemical crewed).

4.2.3 Earth Departure Stage as the Trans-Mars Insertion Stage: Commercial Tanker

This option is comparable to the previously mentioned propellant tanker option, but it reduces the number of required Ares V launches by utilizing commercial tankers. As the Ares V propellant tanker option evolved, it was noticed that most of the required launches are dedicated propellant delivery launches for the TMI stages (five of the nine launches). As a potentially enhancing capability, it became evident that if a commercial market for LOX/LH_2 propellants indeed exists 20–25 yr in the future, the dedicated Ares V launches to deliver this propellant could be replaced by this commercial market. Therefore, the absolute minimum number of required launches for Ares V was found, which was four. This consists of the very large masses and the platform for the commercial tanker services to deliver the propellants to (i.e., empty or nearly empty EDSs). In addition, there will be numerous, but not yet determined, commercial launches necessary to provide the additional propellant to the TMI stages for the MTVs.

Two of these four Ares V launches will deliver the dual-use shroud and encapsulated payload along with an empty (or nearly empty) EDS. These two launches will serve as the two cargo MTVs. The later two launches would deliver the TransHab module (associated elements, small truss, contingency consumables, etc.), the MOI/TEI modules, and two empty (or nearly empty) EDS elements that will serve as the crewed MTV TMI stages. Upon R&D with the MTV, these EDS elements would await more propellant from the commercial tankers.

While this unfetters NASA from the responsibility of delivering all required mass to LEO, there are also numerous other considerations such as the interactions between commercial and government vehicles, ability to accept multiple propellant loads with a single EDS, reliability of the significant increase in required launches per mission, and a host of others. If only NASA-to-NASA interfaces are desired, commercial industry must position and maintain a depot in space where fuel may be extracted, such that a NASA launch vehicle interfaces with a NASA-assembled MTV. Further, one might consider the cost associated with the number of commercial launches to deliver up to 200 t of propellant to each cargo MTV and up to 250 t to the crewed MTV, as opposed to the marginal cost of two or three additional Ares V flights within a year. Nevertheless, it would allow for the establishment of a relatively stable market for up to 650 t of LOX/LH_2 cargo to LEO on a 2-yr repeating cycle and engage the commercial (and potentially international) industry in the overall Mars mission strategy—a potentially attractive development that may help ensure the longevity of the Mars campaign. Furthermore, the capability would still exist to deliver the propellants with the Ares V tanker if the commercial market does not present itself or if there are unforeseen insurmountable challenges associated with this propellant delivery method. Figure 28 further describes this option.

The general assembly sequences are the same as the EDS-as-MTV option, but it can be seen that several commercial tankers will replace the dedicated propellant launches. These assembly sequences have not been created.

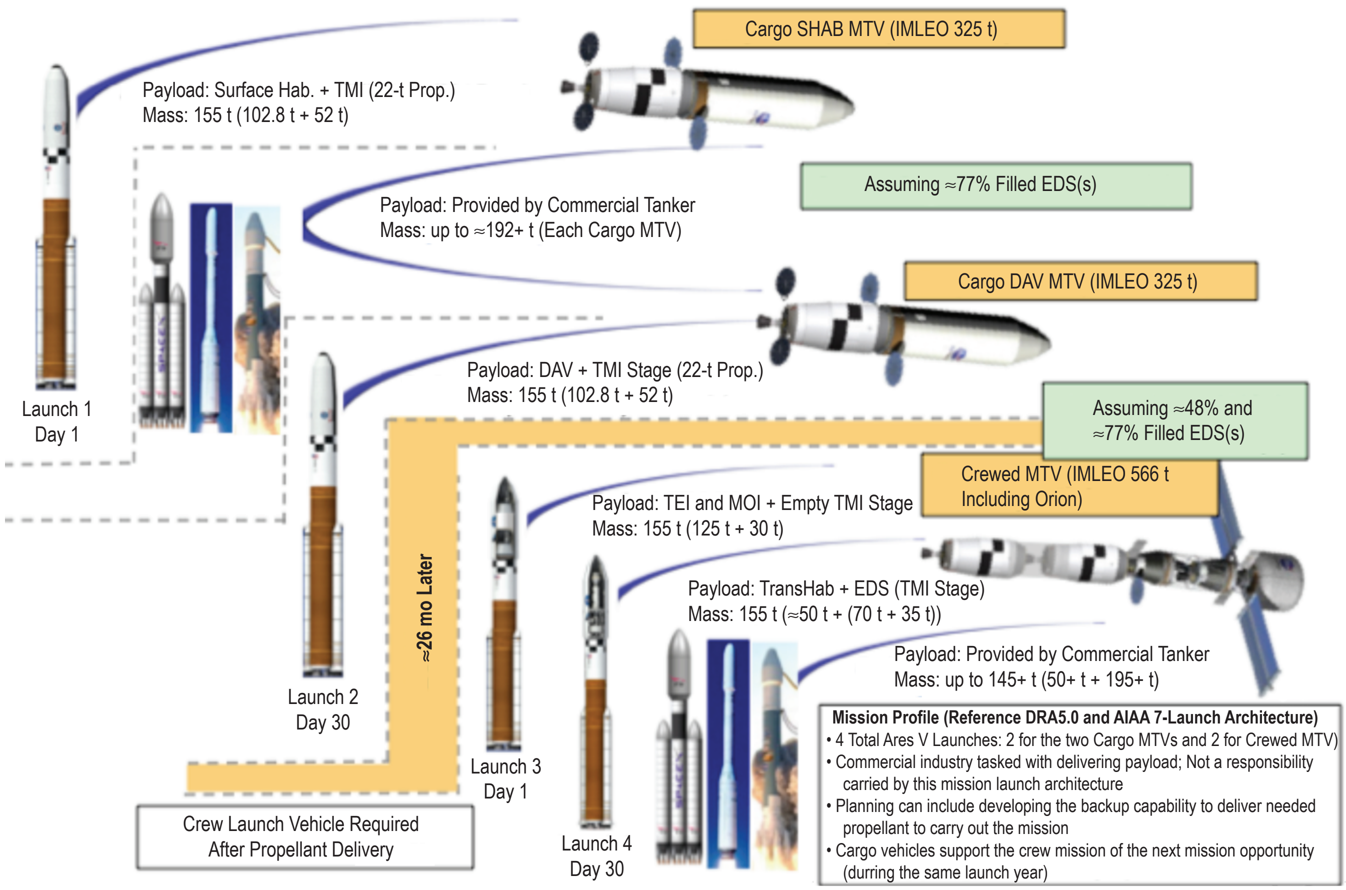

Figure 28. Commercially delivered propellant option (EDS tanker derivative).

5. MARS MISSION ANALYSIS: KEY CHALLENGES

5.1 Required Launch Vehicle Build, Storage, and Transportation Rate

The number of vehicles that will be manufactured in order to meet mission requirements must be addressed as part of ground operations in order to successfully drive towards Mars architecture. The options assessed for Mars, and emphasized herein, are the NTP and chemical propulsion systems, with between four and six consecutive launches, at a rate of one every 30 days.

In order to address extensibility of the current Ares V concept to the Mars DRM, the CS element team investigated modifications needed to support some of the particularly stressing aspects of that particular mission including various manufacturing, transportation, and storage approaches required to support the higher flight rates. As the largest of the Ares V elements, the CS is particularly interested in these operational changes and the necessary testing infrastructure required to develop the Ares V vehicle (at the Marshall Space Flight Center (MSFC), Stennis Space Center (SSC), and/or others) and ground infrastructure needed to successfully assemble and launch the vehicle at this increased rate (at the Kennedy Space Center (KSC)).

For this assessment, the PA-C3D candidate POD vehicle was used as a baseline, as seen in figure 29. This offered a CS from which others could be derived, such as a vehicle with more (or less) dV output from the CS. Driving factors were identified in order to determine this, including the amount of propellant or dry mass required and performance of the engines (e.g., six RS-68B-E/0s may be employed to provide a higher dV from a PA-C3D-derived CS). It was determined that, if allowed to grow, the CS would exceed the Vehicle Assembly Building (VAB) height limit, and the Michoud Assembly Facility (MAF) diameter limit. These considerations may require decisions on additional investments in infrastructure, vehicle concept/configuration changes, or a new vehicle design.

The CS manufacturing team generated a plan for MAF to support the Mars DRA 5.0 missions. This plan maintains the current manufacturing and facility approaches planned for the Ares projects. These plans, which are documented in a 2009 MAF manufacturing study, include the capability of producing a CS every 3 mo as a one-shift operation. Although this rate cannot keep pace with consecutive 30-day launch intervals, the MAF timeline, as seen in figure 30,[6] shows the need for operations to start soon enough to build ahead and store the necessary quantity of stages. This approach will require storage facilities to be constructed either at MAF or KSC. If stored at MAF, stages will ship to KSC every 30 days, probably requiring additional barges. If stored at KSC, each stage will ship upon completion from MAF every 3 mo.

5.2 Required Launch Vehicle Launch Rate

The preliminary Mars Ground Operations Project's (GOP) Operational Concepts (Ops Con) for the Mars DRM builds off launch rate assumptions made during the Mars DRA 5.0

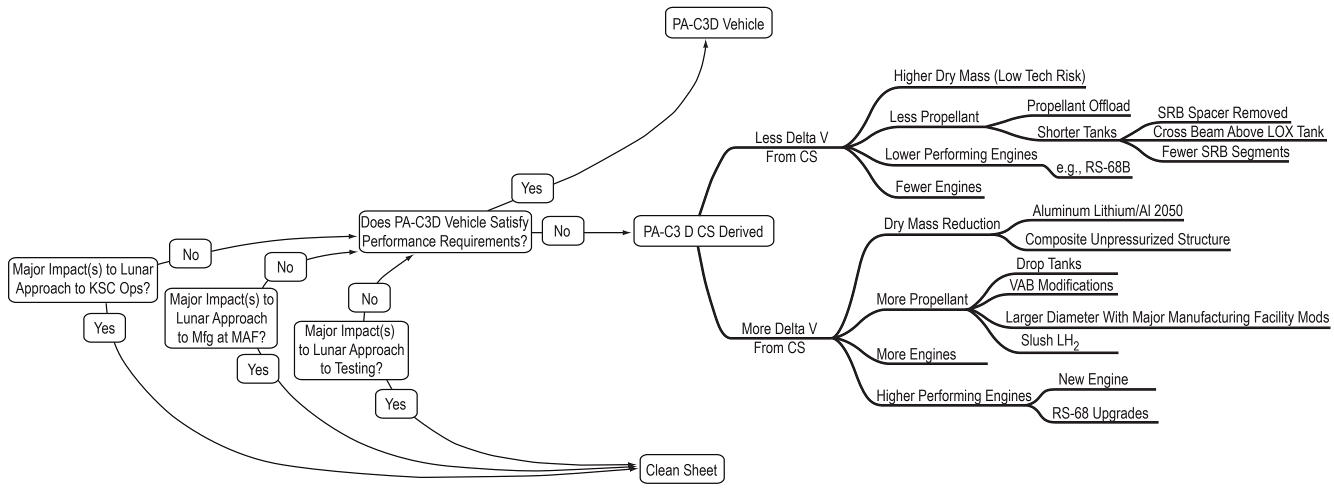

Figure 29. General decision tree for Ares V CS extensibility to Mars DRM.

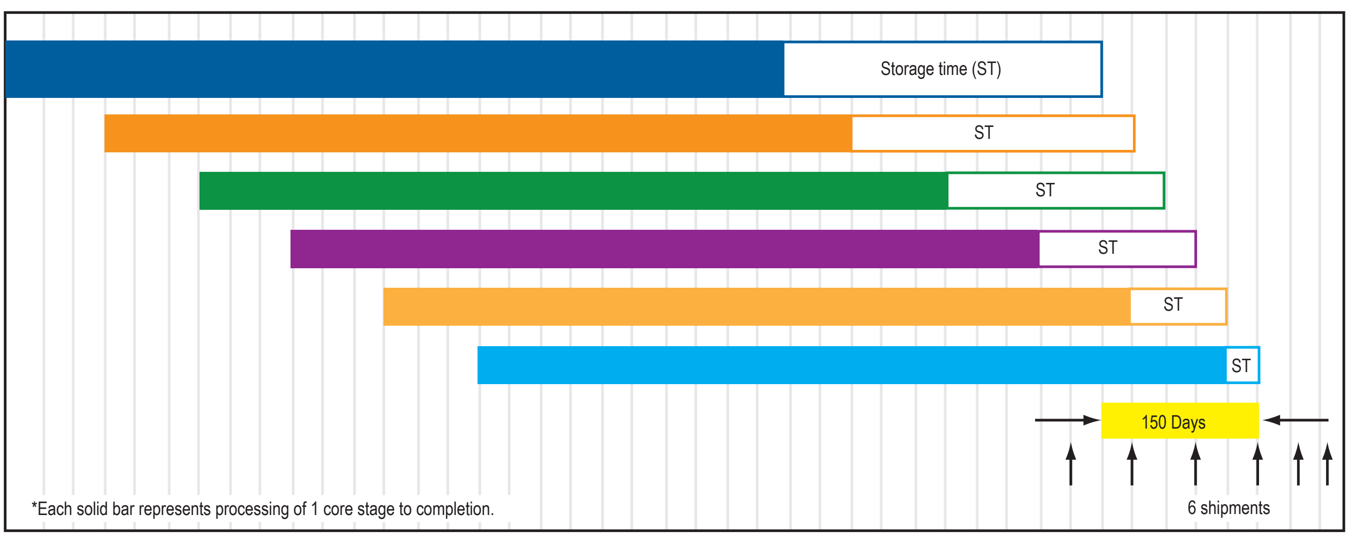

Figure 30. MAF CS processing timeline.

assessment. The launch manifest objectives in DRA 5.0 include either 9 consecutive launches for the NTP option or 12 consecutive launches for the chemical option supporting the back-to-back cargo and crew Mars campaigns, as shown in figure 31. For this study, launches were assumed to be every 30 days.

In order to determine a preliminary Mars GOP architecture that would meet the launch spacing and flight rate recommendations, several assumptions were needed. The vehicle would

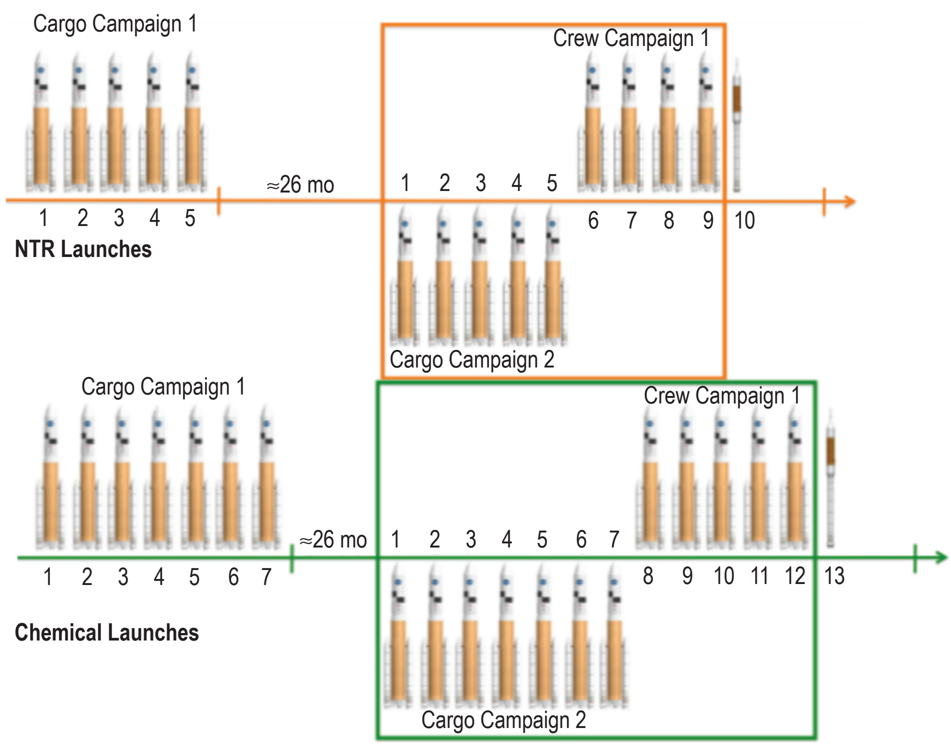

Figure 31. Proposed Ares V flight rate assumptions (NTR or chemical).

fit within the size constraints of the VAB (high bay (HB) heights, widths, etc.). The CS and EDS arrive in the VAB in time for launch Vehicle Integration (VI) (ship-to-integrate). The ship-to-integrate concept is chosen to minimize flight hardware processing at the launch site and to minimize infrastructure to support storage of flight elements. No concurrent lunar missions or Ares V launches for other programs would be available for the GOP while supporting a Mars launch campaign.

This study assumes the lunar program has been completed and the follow-on program is the Mars campaign. The Mars program begins with the following lunar program assets already in place:

- Two Ares V mobile launchers (MLs) and two Ares V VAB HBs.
- Two transporters.
- Two launch pads (LC39 Pads A and B).
- The Space Station Processing Facility (SSPF).

The preliminary GOP Mars architecture was developed utilizing existing lunar campaign assets, and then it identified any additional launch and processing infrastructure needed to meet the Mars DRA 5.0 requirements. This exercise also met the GOP objectives to lower design, development, test, and evaluation (DDT&E) and operations and maintenance (O&M) costs going into a new program.

The proposed Mars GOP architecture is based on a preliminary deterministic analysis using the first cut of the Ares V ground operations processing timeline. With the limited number of processing requirements defined, the Ares V timeline used Ares I timeline analogs where applicable, along with historical data (where available), and subject matter experts' engineering expertise. This deterministic assessment analyzed four options:

(1) The baseline lunar Ops Con.
(2) SRB Offline Stacking Facility (OSF) Ops Con.
(3) Vertical Integration Facility (VIF) Ops Con.
(4) The baseline lunar Ops Con without ML launch mounts.

The GOP favored the SRB OSF to be the proposed Ares V Mars architecture after taking into account development cost, O&M cost, life cycle cost, and mission manifest satisfaction. This option would allow the GOP to meet the proposed manifest with the fewest assets and facilities with the lowest upfront development cost. The subsequent preliminary Ares V Mars Ops Con, based on the proposed architecture, is depicted in figure 32.

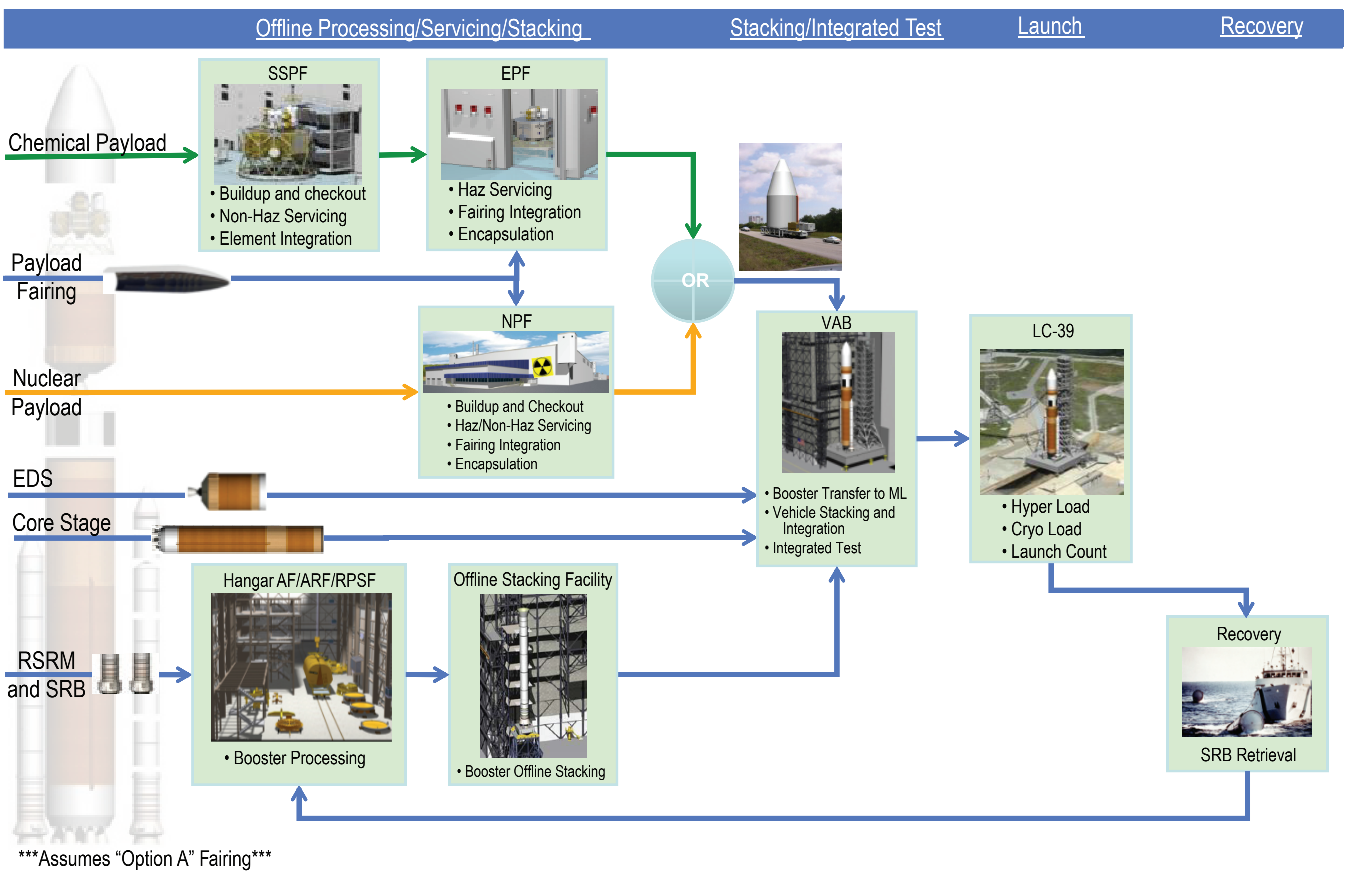

Figure 32. Proposed Ares V Mars Ops Con (chemical or nuclear).

To meet the launch spacing assumption of 30-day centers for Mars missions, several upgrades to the GOP lunar architecture would be required. If a Nuclear Processing Facility (NPF) is not part of the lunar architecture, one would need to be brought on line to support nuclear payloads processing for either chemical or NTP missions and to support nuclear stage handling for NTP missions. If an NPF is needed for the lunar program, it should be sized or modifiable to support a future Mars program. In support of the Mars manifest, the GOP will need one additional (for a total of three) Ares V MLs and one two-bay OSF. The proposed additions to the lunar architecture are depicted in figure 33.

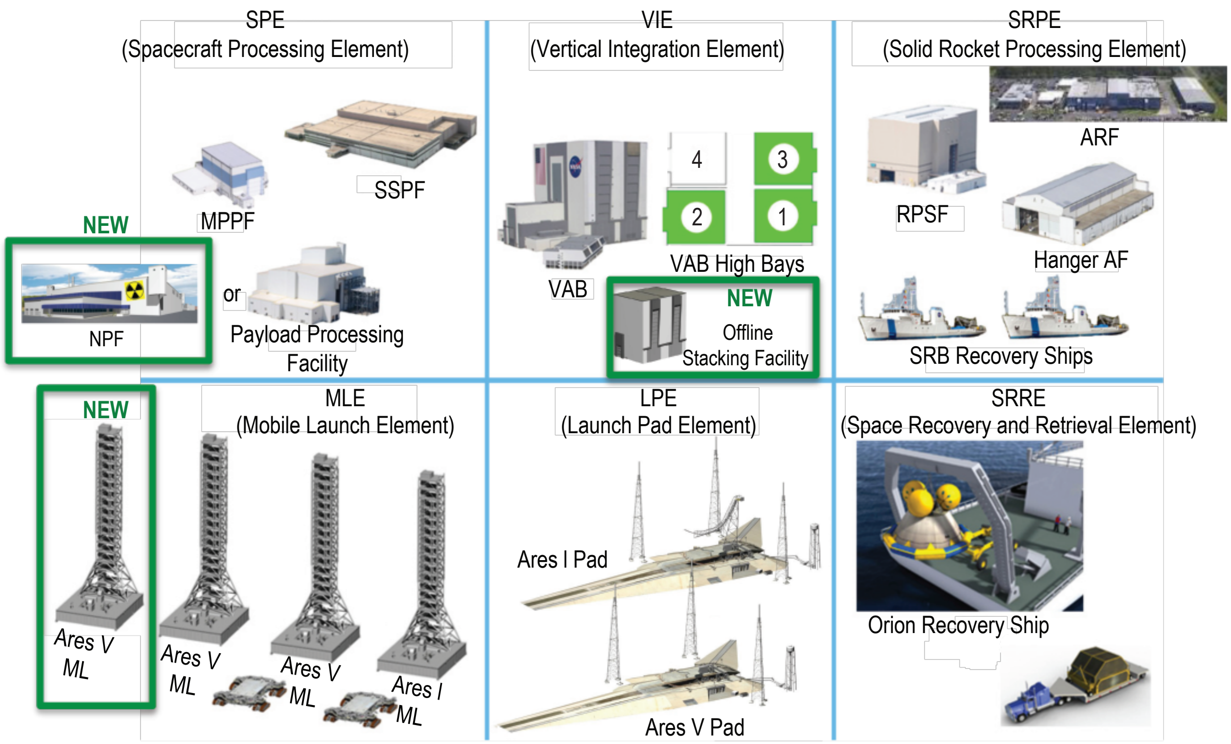

Figure 33. Preliminary proposed GOP Mars architecture (upgrades to the lunar architecture).

Bringing on line an OSF allows ground operations to stack the SRBs in a facility designed to minimize quantity distance (QD) restrictions and to perform SRB stacking operations outside the critical path. DDT&E and O&M costs for this new facility would be lower than that of a much larger VIF. This deterministic assessment also resulted in the fewest Ares V MLs and VAB HBs needed to support the launch manifest and flight rate. Since every option required the use of an NPF, this facility was not a discriminator in the Ops Con and architecture decisions.

Preliminary deterministic analysis that supports a flight rate of 11 launches per year with a minimum of 26-day launch spacing between launches is a 5-day, 3-shift workweek. This work schedule scenario allows for surge capabilities, if needed, by working a 6-day, three-shift workweek or the 7-day, three-shift workweek. A 6-day, 3-shift workweek is required to meet the chemical manifest, which allows for a maximum flight rate of 13 per year with a minimum 22-day spacing between the launches. The preliminary deterministic launch rate and launch spacing analysis results for the OSF option are shown in table 8.

Table 8. Effects of number of ML, HBs, and days in workweek for OSF option as it relates to flight rate and launch spacing, respectively.

Maximum Flight Rate								
				×1 Pad				
Scenario				×2 ML (+2 HB)	×3 ML (+2 HB)	×3 ML (+3 HB)	×4 ML (+2 HB)	×4 ML (+3 HB)
Offline facility	×1 OSF bay	5-day week	No MOD	7.2	7.2	7.2	7.2	7.2
			W/Mod					
		6-day week	No MOD	8.6	8.6	8.6	8.6	8.6
			W/Mod					
		7-day week	No MOD	10.0	10.0	10.0	10.0	10.0
			W/Mod					
	×2 OSF bays	5-day week	No MOD	8.9	11.3	11.3	11.3	11.3
			W/Mod					
		6-day week	No MOD	10.6	13.5	13.5	13.5	13.5
			W/Mod					
		7-day week	No MOD	12.3	15.6	15.6	15.6	15.6
			W/Mod					

Minimum Launch Spacing							
			×1 Pad				
			×2 ML (+2 HB)	×3 ML (+2 HB)	×3 ML (+3 HB)	×4 ML (+2 HB)	×4 ML (+3 HB)
			2	3	3	4	4
			2	2	3	2	3
Scenario			1	1	1	1	1
Offline facility	×1 OSF bay	Standard (5-day work week)	40	40	40	40	40
		Standard (6-day work week)	34	34	34	34	34
		Standard (7-day work week)	29	29	29	29	29
	×2 OSF bays	Standard (5-day work week)	33	26	26	26	26
		Standard (6-day work week)	27	22	22	22	22
		Standard (7-day work week)	24	19	19	19	19

5.3 Payload Mass Required

The Mars DRM is very challenging in many aspects—durations on orbit (LEO, trans-Mars coast, and in the Mars system), a bioastronautics perspective, the technology developments required to support the mission, and many others. Perhaps one that is most often discussed is the large launch mass requirement to complete a human mission to Mars. Some mission architectures have chosen to use many smaller vehicles to achieve this mass requirement, often at the expense of mission reliability, mission complexity, payload packaging considerations, infrastructure investments for the large launch rate, and overall mission cost. However, the Ares V system is capable of delivering very large masses to LEO, allowing for more contingency planning, increasing the mission reliability, and allowing for cost sharing with other on-going efforts by the Agency (space science mission, lunar mission, near-Earth object (NEO) missions, and others).

Figure 34 allows for a very top-level comparison of these driving space exploration mission payloads in comparison with other potential missions.[7] Where most LEO applications are volume driven (ISS habitation modules or a next-generation space telescope), the exploration missions have very high mass and volume requirements. Having a heavy-lift launch system that is capable of delivering these very large, very massive payloads to LEO in support of the mission is a necessity for sustained exploration of the solar system.

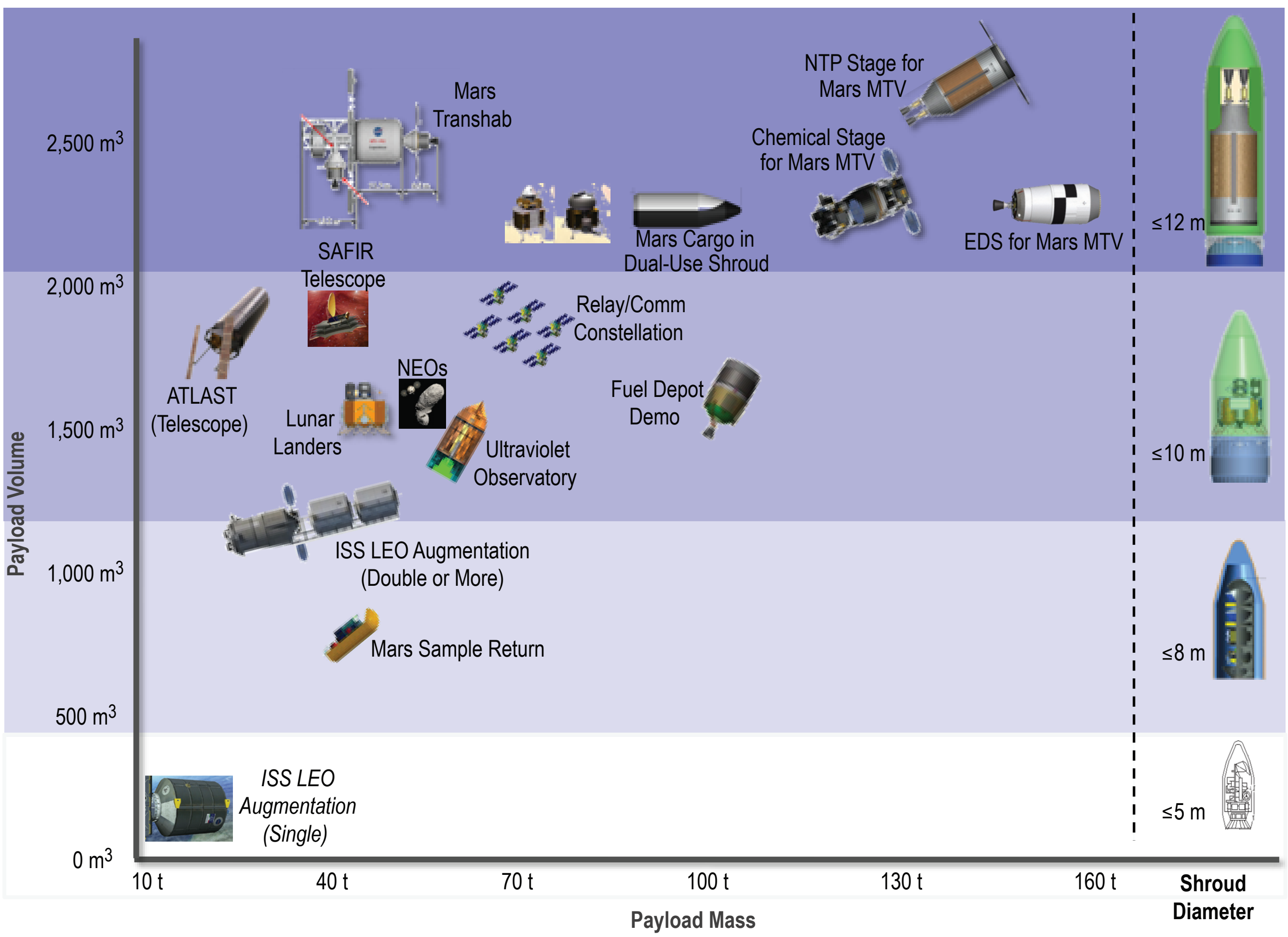

Figure 34. Mass and volume requirements for multiple missions.

5.4 Mission Reliability

Achieving an acceptable mission reliability estimation will be a key challenge when undertaking initial human missions to Mars. While the DRA5.0 architecture is structured to reduce the likelihood of crew loss (cargo deployed before crew commits to mission, contingency consumables to allow the crew the ability to not commit to surface operations, etc.), the mission still consists of many systems across multiple MTVs working in concert to successfully perform a complex and challenging mission. The Ares V Mars campaign risk assessment is a top-level, preliminary estimate of the risk of campaign loss during the ground to LEO and TMI segments of the Mars campaign. The focus is on launch vehicle risk, LEO operations risk, and TMI engine burn risk from a loss of vehicle or delivered payload component perspective. Campaign loss is assumed to occur with the first loss of vehicle or payload component. All technology components are assumed to be fully mature such that their reliability does not improve any further with each use.

The campaign risk is assessed for the following three campaign architectures:

(1) NTP (maximum Ares V participation).
(2) Chemical scavenger.
(3) Commercial tanker.

The sensitivity of the commercial tanker campaign risk to tanker propellant payload capacity and interval between flights is also examined. Loss-of-crew risk is not evaluated in this assessment due to uncertainty in the NTP failure modes and due to its likely being a minor consideration compared to the loss-of-campaign risk.

The common characteristic among these campaigns is that a total of three MTVs are assembled in orbit and sent to Mars. The differences lie in the number of launches needed to supply the components and propellant for the MTVs, the propulsion systems (nuclear or chemical) driving the MTVs, and the duration the MTV components are forced to loiter in LEO as they wait for other components or propellant. The first two MTVs will carry cargo payloads to the Mars surface. After successful deployment of these payloads on the surface of Mars and after the return of the launch window (in about 26 mo), the third MTV, which carries the crew TranHab and crew, will be assembled in LEO from components and propellant and sent to Mars.

Under the assumptions made about the missions, the loss-of-campaign risk is found to be the least for the NTP campaign (17%, 1 in 5.9), followed by the chemical scavenger (22%, 1 in 4.6), and the commercial tanker (78%, 3 in 4). The risk increases with number of launches, which range from 8 launches for the NTP campaign to 41 for the commercial tanker campaign (assuming an 18.5-t tanker payload). The LEO loiter risk sustained by MTV components, which is found to form a significant fraction of overall loss-of-campaign risk, may offer the best opportunity for reducing overall risk if one considers campaign architecture modifications that minimize MTV component loiter time or reduce the amount of loiter risk per day of loiter for each component.

5.4.1 Ground Rules and Assumptions

The following seven assumptions were adopted to identify leading campaign risks:

(1) The first loss of vehicle, payload component, or propellant in the campaign results in a complete loss-of-campaign. No spares, relaunches, or repairs are assumed. Likewise, the consequence of a launch or delivery failure, other than the complete campaign loss, is not assessed.

(2) No launch, in-orbit, or operational delays are assumed.

(3) Common risks are assumed across launches and components. All Ares V launches are treated as carrying identical loss-of-mission (LOM) risk. All components are assumed to carry EDS-like loiter risk in LEO. All commercial tanker launchers are assumed to carry equal risk (though different than Ares V), with orbital operations assumed to carry the same risk as their Ares V counter parts.

(4) LOM risk probabilities for rendezvous, docking, and undocking failures incorporate retries.

(5) Loiter is assumed to carry duration-based risk per component. In-space components are assumed to carry the same loiter risk, and the duration is based on the accumulated time due to the launch-gap durations. Postdocking/pre-TMI burn loiter is assumed to be negligible.

(6) Launch gaps are assumed to be 30 days for Ares V, 15 days* for commercial tanker, and 1 day for the CLV.

(7) 14-day loiter in LEO after MTV assembly for checkout.

(8) Commercial tanker propellant delivery assumptions: 18.5 t* per flight (based on Delta IV Heavy), 201 t each required for MTV1 and MTV2, and 250 t required for MTV3.

(9) A 26-mo interval is assumed between the Earth departure window utilized by the cargo and Mars habitat transfer vehicles (MTV1 and MTV2) and the window used by the crew transfer vehicle (MTV3). All launches from the ground are scheduled ahead of the respective TMI burns to maintain the 26-mo interval.

*Commercial tanker payload and launch intervals are varied in the commercial tanker sensitivity study.

5.4.2 Per-Flight Mission Operations and Risk Inputs

The mission operations for which risks are assumed are given in table 9.[8] They are grouped as follows: (1) Launch and delivery, (2) LEO loiter, and (3) MTV TMI risks. Only LEO loiter risks are accumulated by loiter duration and number of loitering components (as component days). All other risks are assumed to be demand based. Launch vehicle and MTV TMI risks account for burn durations. Ares V ascent risks and LEO loiter risks are taken from the PA-C3′ loss-of-mission estimates. CLV ascent risks are based on Ares I. Commercial tanker ascent failure risk is assumed to be 1 in 100. The LEO loiter risk is based on thermal control system (TCS), RCS, avionics and communications, fuel cell, and structural MMOD failures. The values are scaled directly from the 4-day loiter risks assumed in the Ares V PA-C3′ lunar sortie mission. Automated rendezvous and docking information is based on Russian heritage figures, and no adjustments are made for crew docking, which would allow for manual override dockings. Undockings are considered very low risk and are not explicitly included. Ascent and orbital fuel transfer and solar array deployment use placeholder failure probabilities. MTV NTP operational risk during TMI burn is primarily based on RL-10 risk, given the likely similarities of NTP and RL-10 engines. MTV chemical burn (used for scavenger and commercial tanker campaigns), which utilizes the J-2X engine, is assigned risk based on the restart burn failure rates assumed in the Ares V lunar sortie. The duration from first to last launch varies from approximately 850 days (NTP) to over 1,700 days (commercial tanker), assuming a 26-mo (780-day) Earth-Mars alignment window between the TMI burns of the first and third MTVs.

Table 9. Risk elements.

Mission Operation	Risk	Risk Unit	1 in
Launch and Delivery	Ares V ascent	8.6×10^{-3}	1 in 120
	CLV	2.8×10^{-3}	1 in 360
	Comm. tanker	1.0×10^{-2}	1 in 100
	Ascent fuel transfer	1.0×10^{-3}	1 in 1,000
	Solar deploy	2.0×10^{-3}	1 in 500
	Rendezvous and docking	1.5×10^{-3}	1 in 690
	Orbital fuel transfer	2.0×10^{-3}	1 in 500
LEO Loiter	LEO Loiter	1.3×10^{-4}	1 in 7,400
MTV TMI	TMI MTV1/2 NTP	2.9×10^{-3}	1 in 350
	TMI MTV3 NTP	4.6×10^{-3}	1 in 220
	TMI MVT1/2 chemical	2.0×10^{-3}	1 in 500
	TMI MVT3 chemical	3.5×10^{-3}	1 in 290

5.4.3 Loss-of-Campaign Risk Estimation

The overall campaign risk (based on the first loss-of-mission experienced in the delivery of any of the components or propellant) during LEO loiter or in the failure of an MTV during TMI burn is computed by accumulating the failure probabilities of all the operations in all of the launches. The relationship between launch and operations by demand or duration is given in table 10 for the three campaigns.

5.4.4 Campaign Comparisons

The comparison of loss-of-campaign risk is shown in table 11. The risk is further decomposed into risk carried by the launch vehicles and their payload (including loiter and MTV risks).

Table 10. Launch manifests with assigned activities (each of which incurs a risk indicated in table 9).

Launch	Qty	Ares V or CLV or CHLLV	Planned Day of Launch	Payload	Ascent—Core—Engine	Ascent—Core—Nonengine	Ascent ED—Engine	Ascent EDS—Nonengine	CLV—Engine	CLV—Nonengine	CHLLV—Engine	CHLLV—Nonengine	Ascent Fuel Transfer	Solar Deploy	Rendezvous	Docking	Orbital Fuel Transfer	Undocking	Loiter	TMI MTV1/2 NTP Engine	TMI MTV1/2 NTP Nonengine	TMI MTV3 NTP Engine	TMI MTV3 NTP Nonengine	TMI MTV1/2 Chem. J-2x	TMI MTV1/2 Chem. Nonengine	TMI MTV3 Chem. J-2x	TMI MTV3 Chem. Nonengine
NTP																											
1	1	Ares V	0	NTP propulsion	1	1	1	1						1	1	1		1	134	3	1						
2	1	Ares V	30	NTP propulsion	1	1	1	1						1	1	1		1	134	3	1						
3	1	Ares V	60	Mars cargo	1	1	1	1						1				1	14								
4	1	Ares V	90	Mars habitat	1	1	1	1						1				1	14								
5	1	Ares V	779	NTP propulsion	1	1	1	1						1				1	136			3	1				
6	1	Ares V	809	NTP fuel	1	1	1	1						1	1		1		45								
7	1	Ares V	839	TransHab module	1	1	1	1						1	1		1		15								
8	1	CLV	840	CEV					1	1				1	1		1		14								
Chemical Scavenger																											
1	1	Ares V	0	Mars habitat + TMI stage	1	1	1	1						1					268					1	1		
2	1	Ares V	30	Mars cargo + TMI stage	1	1	1	1						1					208					1	1		
3	1	Ares V	60	Tanker EDS	1	1	1	1					1	1	1	1	1	2	0								
4	1	Ares V	90	Tanker EDS	1	1	1	1					1	1	1	1	1	2	0								
5	1	Ares V	120	Tanker EDS	1	1	1	1					1	1	1	2	2	3	0								
6	1	Ares V	809	TEI and MOI stages	1	1	1	1						2				1	210							1	1
7	1	Ares V	839	EDS TMI stage	1	1	1	1					1	1	1	1		1	75								
8	1	Ares V	869	TransHab module	1	1	1	1						1	1	1		1	45								
9	1	Ares V	899	TMI stage	1	1	1	1						1	1	1		1	15								
10	1	CLV	900	CEV					1	1				1	1	1		0	14								
Chemical Commercial Tanker (18.5 t)																											
1	1	Ares V	0	Mars habitat + TMI stage	1	1	1	1						1					1,978					1	1		
2	1	Ares V	30	Mars cargo + TMI stage	1	1	1	1						1					2,008					1	1		
3	11	CHLLV	60	Propellant							11	11			11	11	11	11	0								
4	11	CHLLV	105	Propellant							11	11			11	11	11	11	0								
25	1	Ares V	1,094	TEI, MOI, TMI (empty) stages	1	1	1	1						2				1	2,025							1	1
26	1	Ares V	1,124	TransHab module	1	1	1	1						1	1	1		1	1,290								
40	14	CHLLV	1,154	Propellant							14	14			14	14	14	14	0								
41	1	CLV	1,740	CEV					1	1				1	1	2		2	0								

Table 11. Loss-of-campaign risk for three architectures, with commercial tanker assumed to carry 20 t of propellant per launch, with 15 days between commercial launches.

Campaign	NTP	Chemical Scavenger	Commercial Tanker
Launches (Ares V + comm. + CLV)	7+0+1	9+0+1	4+35+1
Overall loss of campaign	17%, 1 in 5.9	22%, 1 in 4.6	78%, 1 in 1.3
Risk breakdown by launcher type			
Ares V + payload	16%, 1 in 6.1	21%, 1 in 4.7	65%, 1 in 1.5
Commercial + payload	N/A	N/A	38%, 1 in 2.6
CLV + payload	1 in 120	1 in 120	1 in 140
Breakdown by launcher mission phase			
Launch ascent only	1 in 16	1 in 13	1 in 3.0
Postascent delivery	1 in 43	1 in 23	1 in 7.7
Loiter (component days)	510	840	7,300
Loiter	6.6%, 1 in 15	10%, 1 in 10	63%, 1 in 1.6
TMI burn	1 in 32	1 in 120	1 in 120

It is also broken down by mission segment across all launches of each campaign. The differences between the NTP and chemical scavenger are small, with NTP gaining an edge across all breakdown categories except "CLV + payload" and TMI burn, both of which have minor overall impact. The numbers of launches, including CLV, are 8 for NTP and 10 for chemical scavenger. The chemical commercial tanker results indicate considerably worse risk for campaign loss. For this comparison, a tanker payload of 18.5 t, which corresponds to current Delta IV heavy lift capability, and a tanker launch spacing of 45 days were assumed. The 18.5-t assumption requires that there be 36 tanker launches or 41 launches overall.

The mission phase risk breakdown in table 11 indicates that, in all cases, both launch (ascent) and LEO loiter carry a similar order of risk and represent the leading risks across all three campaign architectures. LEO loiter risk is carried exclusively by the payload of the Ares V launches in all cases, and commercial tanker launches are assumed to carry no loiter risk directly (except through the orbital phasing operation prior to docking and propellant transfer). Because there are only four Ares V launches in the commercial tanker case, its loiter risk is imposed entirely on those flights. Even without tanker risk factored in, the Ares V + payload campaign loss risk, at 67%, 1 in 1.5, still far exceeds the risk incurred by the NTP and chemical scavenger architectures. This is due to the total number of loiter component days imposed on the installed components in LEO as the 36 tankers fill up the MTV tanks with propellant. In all campaigns, loiter risk is significant; in the commercial tanker case, it is the dominant nontanker source of risk.

The commercial tanker campaign loiter risk is directly dependent on the number of tanker launches required (a function of the tanker payload) and the launch spacing between the tanker launches. In table 12, the results of a sensitivity study of commercial tanker loss-of-campaign risk are given for tanker launch gaps of 15, 30, and 45 days, and for 18.5-, 30-, and 120-t payloads. The 15-day launch gap is a best-case optimistic launch rate, and the 45-day gap would correspond to a more currently viable launch rate. The 18.5-t payload is representative of the current commercial

Table 12. Loss-of-campaign risk sensitivity against commercial tanker payload and gap between commercial tanker launches.

Campaign	Tanker Gap	18.5 t	30 t	120 t (Ares V)
Tanker launches (MTV1, 2, 3)		36 (11, 11, 14)	23 (7, 7, 9)	6 (2, 2, 2)
Overall loss of campaign	15 days	59%, 1 in 1.7	47%, 1 in 2.2	20%, 1 in 4.9
Ares V + payload		33%, 1 in 3.0	25%, 1 in 3.9	13%, 1 in 7.6
Commercial + payload		38%, 1 in 2.6	27%, 1 in 3.7	7.8%, 1 in 13
Total loiter (comp. days)		870	1,800	640
Overall loss of campaign	30 days	70%, 1 in 1.4	55%, 1 in 1.8	24%, 1 in 4.1
Ares V + payload		51%, 1 in 1.9	39%, 1 in 2.6	17%, 1 in 5.8
Commercial + payload		38%, 1in 2.6	27%, 1 in 3.7	7.8%, 1 in 13
Total loiter (comp. days)		5,000	3,300	1,000
Overall loss of campaign	45 days	78%, 1 in 1.3	64%, 1 in 1.6	28%, 1 in 3.6
Ares V + payload		65%, 1 in 1.5	50%, 1 in 2.0	21%, 1 in 4.7
Commercial + payload		38%, 1 in 2.6	27%, 1 in 3.7	7.8%, 1 in 13
Total loiter (comp. days)		7,300	4,700	1,400

heavy lift capability, 30 t is a likely commercial payload capability in the near future, and a 120-t payload corresponds to the Ares V payload capacity. The range of loss-of-campaign risk varies from 78% (1 in 1.3) for a 18.5-t payload at 45-d intervals to 20% (1 in 4.9) for the 120-t payload at 15-d intervals. The use of Ares V as a tanker at 15-d launch intervals brings the tanker campaign architecture in line with the NTP and scavenger options. However, the use of the Ares V diminishes the campaign architecture's definition as a commercial tanker option.

One can argue that, of the three campaign architectures, the commercial tanker approach offers the greatest opportunity for backup flights should there be a propellant delivery failure. In one scenario, if one assumes that the commercial tanker launchers can always be relaunched and that the Ares V cannot, then the entire loss-of-campaign risk is indicated by the "Ares V + payload" risk alone. If one considers the 18.5- and 30-t cases of the commercial tanker but omits the tanker launch risk, then the commercial tanker campaign architecture can begin to approach the NTP and scavenger range of campaign risk if the tanker launch gap is in the 15-day range (this neglects the marginal increase in loiter risk by any relaunches).

5.4.5 Conclusions

Loss-of-campaign risk was estimated for three campaign architectures: NTP, chemical scavenger, and chemical commercial tanker. The risk estimate assumes that a loss-of-campaign is observed with the first loss of a launch vehicle, payload, or MTV, and that component reliability does not mature with each launch and use. Of these campaigns, the NTP architecture was found to bear the least risk (of loss of campaign (17% or about 1 in 6), followed by the chemical scavenger (22% or about 1 in 5), and the commercial tanker (78% for a 45-day launch gap and a 18.5-t tanker payload). The loss-of-campaign risk is found to be strongly dependent on the total number of launches and accumulated loiter time, which is partly a function of launch gap. Given the strong dependence found on loiter duration, it is recommended that the basis of the loiter risk per component day and the assignment of loiter risk be reexamined to verify their importance in overall campaign risk. If loiter risk persists in its impact, it becomes a viable target for redesign of the architecture in the effort to reduce overall loss-of-campaign risk. One such architecture that may bear examination in this context is the propellant depot variant of the commercial tanker, where the tankers fill a permanent propellant storage facility in LEO to which assembled MTVs would dock, acquire needed propellant, and embark toward Mars with a minimum of LEO loiter. While reducing loiter duration and the risk attributed to that key driver, the risk associated with the number of launches will be unaffected. Furthermore, more analysis will have to be performed to assign a reasonable risk value to the propellant depot itself and its associated accumulated loiter duration in LEO.

5.5 On-Orbit Assembly Operations/Functionality Required

The EDS Mars Extensibility Team conducted an assessment of the potential roles of the Ares V EDS element in support of a Mars mission campaign as laid out in Mars DRA 5.0 and the on-going Ares V assessment of the CxP Mars DRM during PA-C3′ and PA-C3″, including on-orbit assembly operations for NTP, chemical, and EDS-as-MTV-stage campaigns. The objective was to identify the functionality required, to assign functions to EDS or MTV, and to develop EDS-extensibility options in the Mars DRM trade space.

Ground rules and assumptions of the assessment included the following:

- Basic Ares V EDS conceptual design for the PA-C3D configuration is used as a point-of-reference for assessing additional functionality requirements.
- EDS includes the J-2X engine as designed for lunar DRMs (LOX/LH_2 engine).
- Within each campaign, a similar EDS configuration is utilized when possible.
- MTV payloads provide for their own needs.
- Appropriate guidance, navigation, and control (GN&C)/ACS/RCS components are included on EDS stage configurations as applicable.
- Assessment includes only nominal missions (no contingencies).
- Disposal burns are not addressed.

EDS Extensibility Team members included EDS element management, project lead systems engineers, VI systems integration and vehicle operations engineers, support systems engineers, and discipline engineers from propulsion, cryogenic fluid management (CFM), avionics, and operations.

Product deliverables resulting from the TIM include an EDS functionality/capability matrix, hardware/configuration changes matrix, and trade/technology trees. Functions identified and assessed for on-orbit assembly operations include the following:

- Attitude control.
- Debris avoidance (≥150-day loiter).
- Active rendezvous.
- Passive rendezvous.

- Docking for assembly.
- Deorbit and disposal.
- Power.
- Communications.
- Propellant settling.
- Reboost.
- Long-term CFM.
- MMOD protection.
- Insert payload to orbit.
- Undocking.
- Separation.
- Propellant exchange or transfer (chemical and EDS-as-MTV-stage only).

Assessment of the above functions included identifying the current EDS capability coverage to the functions, identifying any major issues with EDS providing a given function, and identifying major trades, analyses, or other drivers that need to be assessed to determine functional capability. The information was recorded in the EDS functionality/capability matrix (see table 13). Trade/technology trees were also constructed for each of the campaigns (see figure 35).

Extensibility Team members recorded discipline observations and system impacts. Key trades and required forward work/future analysis was identified in several areas. Key system level impacts noted include the following:

- Need for auxiliary propulsion system, including substantial mass and fuel requirement.
- Power requirements for ZBO capability.
- Increased radiation hazard for a ≥150-day loiter.
- Nuclear disposal requirement for all campaigns.
- In-space propellant transfer.
- Avionics required for automated docking.
- Significant increase in number and complexity of communication interfaces.

5.6 Earth Departure Stage as Mars Transfer Vehicle Stage

The EDS Mars Extensibility Team assessed the NTP, chemical and EDS-as-MTV stage campaigns for Mars exploration. Of the three campaigns, the EDS-as-MTV stage has numerous challenges that are specific to this campaign. These are summarized as follows:

- J-2X not available for circularization burn because second burn has to be TMI.
- Cooperative debris avoidance of multiple EDSs.
- EDS-to-EDS interfaces, including structural and communication.
- Disposal of stages providing second part of TMI.
- Propellant settling for multiple EDSs.
- Propellant transfer between stages.
- Propellant transfer between drop-tanks and main tanks.

Table 13. Sample section of EDS functionality/capability matrix.

	Chemical MTV Assembly (c&h)			EDS as MTV Stage ((c&h)			NTP MTC Assembly		
Nomenclature: **• Mission: NTP, CMTV, EDSMTV** **• cargo (c), human)(h)** **• Element: MTV, EDS, DISS**	**Current Capability**	**Issues**	**Trades & Tech**	**Current Capability**	**Issues**	**Trades & Tech**	**Current Capability**	**Issues**	**Trades & Tech**
Docking for Assembly	• No capability in current EDS. • EDS does not currently own payload interface	• MTV will need to have structural docking system • Part of the adapter to EDS may stay with MTV elements for later assembly of MTV.	• Proximity maneuvers. • Off-CG control. • Constraints imposed by MTV docking systems and procedures. • Additional H/W.	No capability current in EDS.	• Extra systems (MPS, structure, etc.) have to be jettisoned prior to RPOD. • Configuration of EDS-to-EDS berthing/docking hardware a potential design, structural avionics, and ops issue when also combined with drop tanks concept. • Slightly different implications of this issue for 2-EDS inline cargo and multi-EDS human mission.	• Configurational trades to account for different MTV configurations and docking options. • Proximity maneuvers. • Constraints imposed by MTV docking systems and procedures. • Additional H/W.	No capability current in EDS. EDSS does not currently own payload interface.	Assumption that NTP (launch 1) power system provides power for NTP element docking (power LH2, line connection, etc.). If not, what are implications to EDS? • MTV will need to have structural docking system. • Part of the adapter to EDS may stay with MTV elements for later assembly of MTV (especially human launch 3).	• Proximity maneuvers. • Constraints imposed by MTV docking systems and procedures. • Additional H/W.
Deorbit and disposal	Limited capability in current EDS (no aux engines).	• Different back-away, deorbit, disposal scenarios.	• What additions to RCS H/W system needed? • Sizing of propellant to Aux and RCS system.	Limited capability in current EDS (no Aux engines).	• Different back-away, deorbit (for stages doing first part of TMI), disposal scenarios, deep space disposal for stages providing second part of TMI.	• What additions to RCS H/W system needed? • Sizing of propellant to aux and RCS system.	Limited capability in current EDS (no Aux engines).	• Different back-away, deorbit, disposal scenarios. • Need to be able to perform disposal to higher NSO for NTP and surface nuclear power system on cargo mission. Raising full stack to NSO could be prohibitive to this option without an additional EDS, which has lots of other implications.	• What additions to RCS H/W system needed? • Sizing of propellant to Aux and RCS system.
Power	Limited duration power, which is currently fuel cell/ batteries (mission 1).	Current system cannot support ≥150 day loiter.	• New solar power array system. Can it be kitted? • Can we use MTV power in certain configurations?	Limited duration power, which is currently fuel cell/ batteries (mission 1).	Significantly more power required to accommodate ZBO of main propellant tank. How much power is needed?	Power system options.	Limited duration power, which is currently fuel cell/ batteries (mission 1).	Current system cannot support 180+ day loiter.	• New solar power array system. Can it be kitted? • Can we use MTV power in certain configurations?
Communications	Limited capability in current EDS.	• How does one EDS communicate with other EDSs? • What are comm. Implications for AR&D? • Knowledge of MTV health and status during ascent and around RPOD.	• Who has control during various phases? • Comm interfaces.	Limited capability in current LEO.	What communication needs are there during highly elliptical and Earth escape phases?		Limited capability in current EDS .	• Do the NTP elements (before transhab) have independent capability to comm. Health and status to MS, or does EDS ascend that data? • What are comm. Implications for AR&D? • Knowledge of MTV health and status during ascent and around RPOD.	• Who has control during various phases? • Comm interfaces.

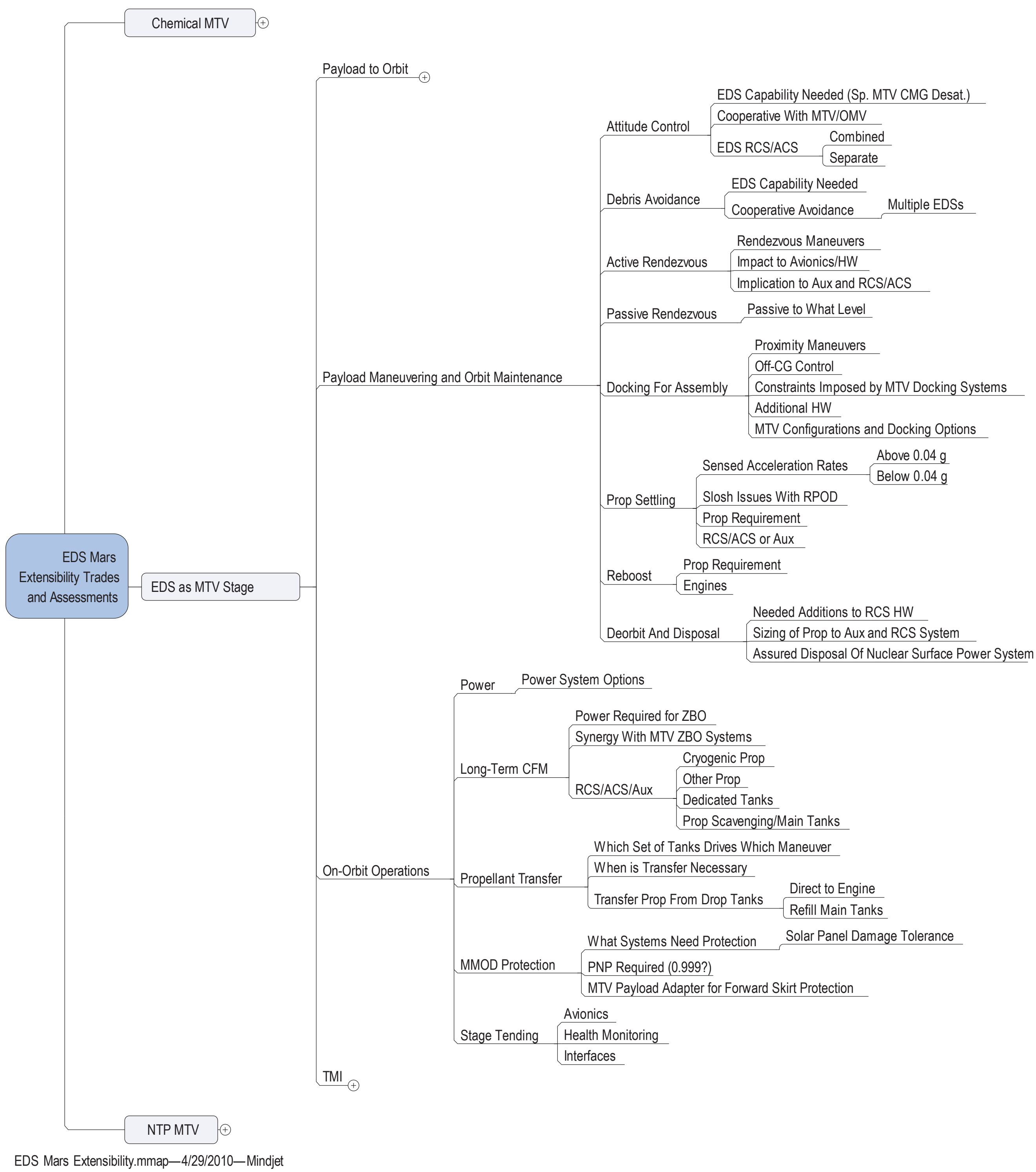

Figure 35. Trade tree for EDS as MTV stage.

- Disposal of drop-tanks.
- Multiple passes (at least two, perhaps four) through the Van Allen radiation belts.
- Restart conditioning of multiple J-2Xs.
- Staging for multiple J-2Xs during TMI.
- Longer single-burn duration and total burn-time duration for J-2X.
- Assured nuclear disposal (surface fission reactor power source).

5.7 Technology Developments Required

Many technologies are required to be developed to support the Mars mission concept; there is little debate in that regard. The common item that is mentioned early in any conversation on this matter concerns the in-space transportation system developments required. This can range from relatively low-technology investments using traditional chemical propulsion systems (either the DRA5.0-derived chemical propulsion modules or partially filled Ares V EDSs), to a more moderate technology development program that develops and incorporates in-space cryogenic propellant transfer, to even more challenging development programs from a variety of aspects (technical, political, development cost, etc.) that utilize NTP, nuclear-electric propulsion (NEP), solar electric propulsion (SEP), or plasma-based electric propulsion (e.g., the Variable Specific Impulse Magnetoplasma Rocket (VASIMR)).

A relatively low technology investment portfolio is needed for the propulsion system itself if a path is chosen along the lines of the DRA5.0 RL-10 derived Chemical Propulsion Modules or if a J-2X derived stage is used. The use of these engine systems has been demonstrated for decades, but modest investment will be needed to ensure overall system integrity over a long–duration assembly operation, control of propellant boiloff during assembly, autonomous rendezvous and docking (AR&D), etc. Furthermore, a J-2X derived solution will need to determine the optimum method for propellant delivery to LEO whether it be propellant residuals in the main tanks, usage of drop tanks, or other methods.

A more moderate technology investment approach can be made in the chemical rocket arena. A technology development that could potentially decrease long-term costs, possibly establish a commercial market, and provide a performance benefit is cryogenic propellant delivery and transfer. Development programs have shown the feasibility of transferring propellant in orbit (e.g., Orbital Express, and noncryogenic propellant)[9] and plans are in place to further demonstrate the feasibility of transferring cryogens (CRYogenic Orbital TEstbed (CRYOTE)[10] and Fiscal Year 2011 proposal for technology development programs). When coupled with the development of an in-space transportation platform such as the Ares V EDS, the possibilities are enormous—up to 250 t of propellant powering a system of almost 300,000-lbf thrust and ≈450-s I_{sp}.

Calculations have shown the potential of delivering almost 200 t to the Moon or over 150 t to Mars (per EDS utilized). Furthermore, it takes an ≈80% full EDS to deliver the cargo payloads assessed in DRA5.0, and two EDSs (one ≈75% full and one at ≈50% full) to deliver the crewed elements of that Mars architecture, as mentioned previously. Overall, this technological development would open up the inner solar system to human exploration by using the propulsion systems that have been designed, developed, tested, and used over the past few decades. Furthermore, it has the potential to use Agency investments made for initial human activities on the Moon. This can result in a decrease in the marginal cost of the Mars campaign by using the predeveloped assets to the maximum extent possible and increase mission reliability by demonstrating the in-space propulsion system by a decade or more before committing to the Mars program.

From the von Braun era of rocket propulsion onward, NTP was thought to be the new era of in-space transportation systems. The Nuclear Engine for Rocket Vehicle Application (NERVA) development program demonstrated the feasibility of the technology, but other aspects of the system of a nontechnical nature have since stunted the development of this technology. Efforts in the early 1990s to underscore the need for this system type (Synthesis Group Report) and the since-cancelled Project Prometheus of the early 2000s still have not led to the full development of this system (let alone a domestic in-space demonstration of its capability). Present day excitement exists in the areas of NEP and/or plasma-based electric propulsion. However, even these technologies require large nuclear systems for electric power when scaled up to the class of propulsion required for a human mission and the mass requirements that entails.

Beyond coupling in-space cryogenic propellant transfer with chemical propulsion or developing nuclear-based systems, several other technology developments were assumed for the assessment (and in DRA5.0). These are further depicted in table 14. This table can be viewed two distinct ways. First, if these technologies are developed for a 'Mars first campaign' a lot of them are backwards compatible with any other potential DRM that the agency may undertake in the foreseeable future (Moon, L2, NEO, etc.). Second, it can be viewed as an incremental technology development roadmap, walking through the inner solar system until a suite of technologies can be utilized to undertake the demanding Mars mission.

Table 14. Technology investments assumed for chemical options in PA-C3′.

Technology	Lunar Cargo	Lunar Crew	L2	NEO	Mars	Beyond Mars
Propulsion enhancement	●	●	●	●	●	●
Multiuse EDS	Planned	Planned	●	●	●	●
Light(er) weight structures	●	●	●	●	●	●
Automated R&D	◘	●	◘	●	●	●
Near zero boiloff rates	◘	●	◘	●	●	●
Build/launch rates	◘	◘	◘	◘	●	●
Larger shrouds	○	○	●	●	●	●
In-space cryogenic propellant transfer	○	○	○	◘	●	●
Multiuse shroud	○	○	○	○	●	◘

Notes: ●–Directly applicable
◘–Somewhat applicable
○–Not applicable

The rationale behind the items includes (at a very top level) the following:

- Propulsion enhancements include any improvements made in J-2X performance that can be applied across the board if that engine is utilized for the in-space transportation propulsive element. Furthermore, any upgrades in the CS or booster set would also have direct implications across the board if a common system is utilized, leveraging these assets in an appropriate manner. This allows for maximum return on investment of taxpayers' dollars in the realm of transportation systems.

- Multiuse EDS implies an EDS (or second stage) that can provide a multitude of services to both the payload and as part of the mission architecture. It is currently planned to provide ETO ascent, maintenance of the Altair lander payload during a 4-d loiter period, passive control during an R&D maneuver with the Orion crew exploration vehicle (CEV), and finally a TLI maneuver to inject the Altair lander/Orion CEV to the Moon. This same rationale could potentially be applied across the board for inner-solar system exploration.

- Lighter weight structures would obviously have implications across the DRM set if a common system is used for exploration.

- Automated R&D (or autonomous R&D) would be useful across the DRM set if some type of assembly operation is required (including 1.5 launch or dual-launch scenarios).

- Will address ZBO in further detail.

- Build/launch-rate improvements could potentially allow the Agency to undertake multiple mission options concurrently or allow a more demanding mission (such as the Mars DRM) to be undertaken.

- Larger shrouds would obviously be required for the more demanding missions, such as Mars and beyond Mars, but it would also be useful for placing large space telescopes or other payloads in either the Sun-Earth L2 or Earth-Moon L2 (Lagrangian point). It may also be required to deliver assets to LEO in support of a human mission to a NEO.

- Finally, the dual-use shroud concept has shown the potential to provide ETO ascent protection, some thermal/space environment protection during the long loiter/trans-Mars coast period, thermal protection during initial MOI, and thermal protection for the entry, descent, and landing phase required to deliver the lander assets to the surface of Mars. This type of shroud may also provide a better systems approach to delivering other assets to destinations with an atmosphere.

ZBO is a fundamental assumption that was assumed in DRA 5.0 and in follow-on studies, and it may be a driving technology need for the type of architecture presented. If the LOX/LH_2 (or LH_2 in the case of the NTP option) boils off (temperature increases cause a phase change from liquid to gas, gas increases pressure in the tank, increase in pressure activates pressure relief valves, and pressure release results in mass loss), the amount of propellant required to provide the appropriate dV may not be there when needed. This either requires more propellant than is actually required to be delivered to LEO so some boiloff may occur without jeopardizing mission success, or a thermal control system capable of keeping the cryogenic propellants cool may be required to prevent the phase change from occurring in the first place.

The amount of cryogenic propellant required for the cargo MTV is almost 90 t of LH_2 for the NTP option or over 190 t of LOX/LH_2 propellants for the chemical option. This is further demonstrated in figure 36, where over one-third of the mass at TMI is propellant for the NTP option, and about 60% of the mass is propellants in the chemical case. For the crewed MTV (fig. 37), the NTP option requires about 120 t for the TMI maneuver (about one-third of the total mass), while the chemical option requires about 315–340 t (about 60%). Even a small percentage of these propellants boiled off over a period of months would be detrimental to the mission architecture.

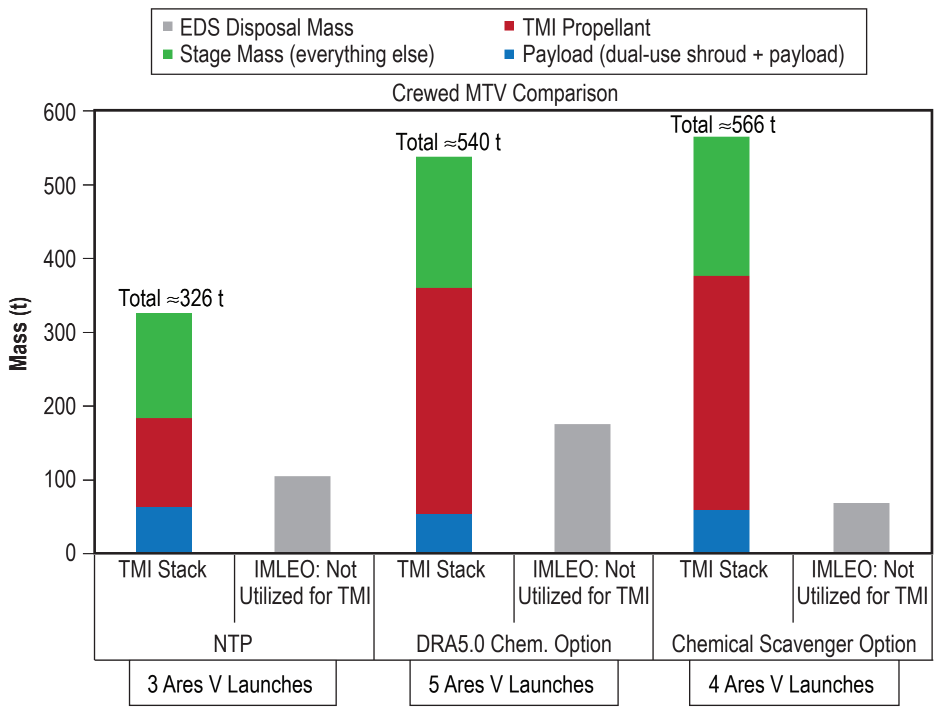

Figure 36. Cargo MTV propellant inventory at TMI.

A very simple method of finding the potential impact of various boiloff rates includes assuming an allowed ‘average boiloff rate per day.’ As shown in figure 38, various boiloff rates per day over a long duration on orbit have various impacts to the system. Assuming 200 t of propellant is delivered to LEO on day 0 and it is allowed to boiloff at the various rates over 180 days, the impact can range from 70 t of propellant lost (rate of loss equal to 0.25% per day) to virtually no loss in propellants at all or ZBO. Therefore, a quantification of ZBO could be described as an average boiloff rate of 0.001% of propellant loss per day over the entire assembly operation.

However, the propellants for the mission options assessed are not always delivered on the first launch. This requires a slightly different approach to calculating the boiloff rate allowed for

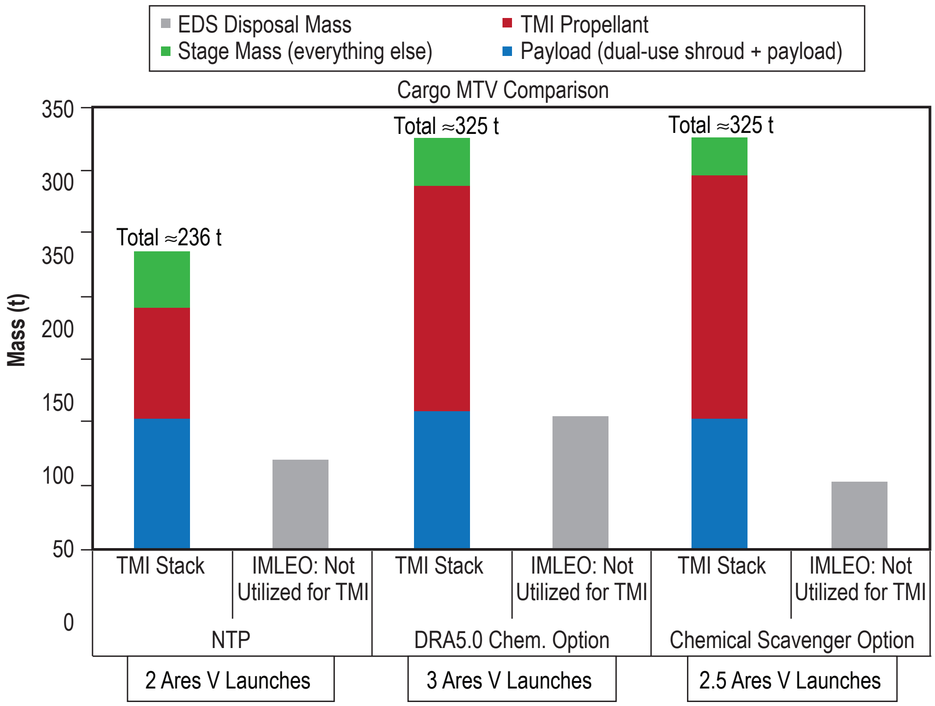

Figure 37. Crewed MTV propellant inventory at TMI.

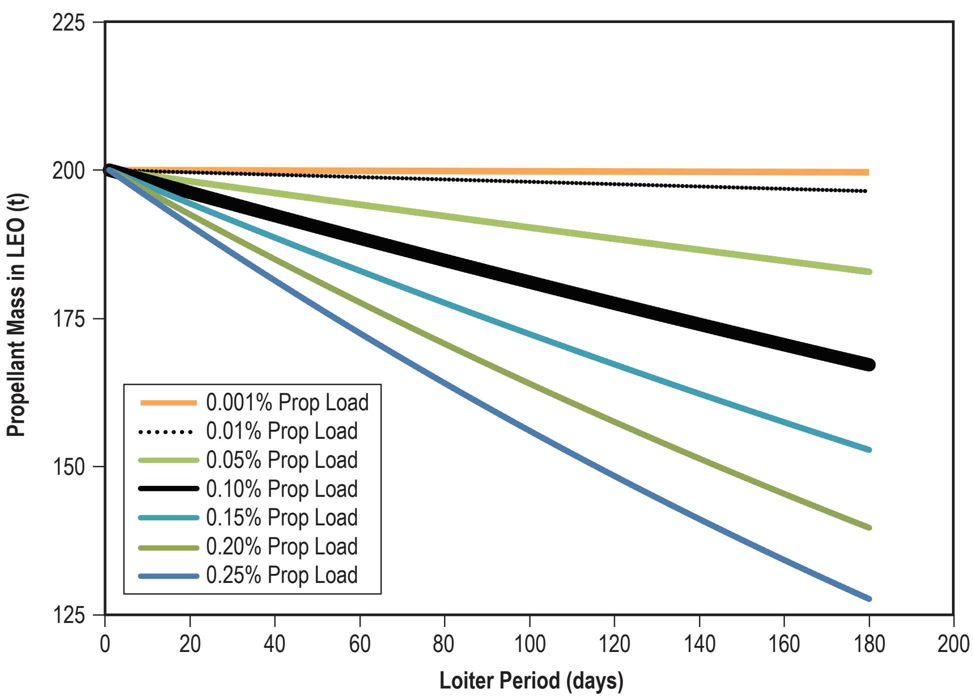

Figure 38. Mass loss of various boiloff rates (200-t initial, 180-day assembly).

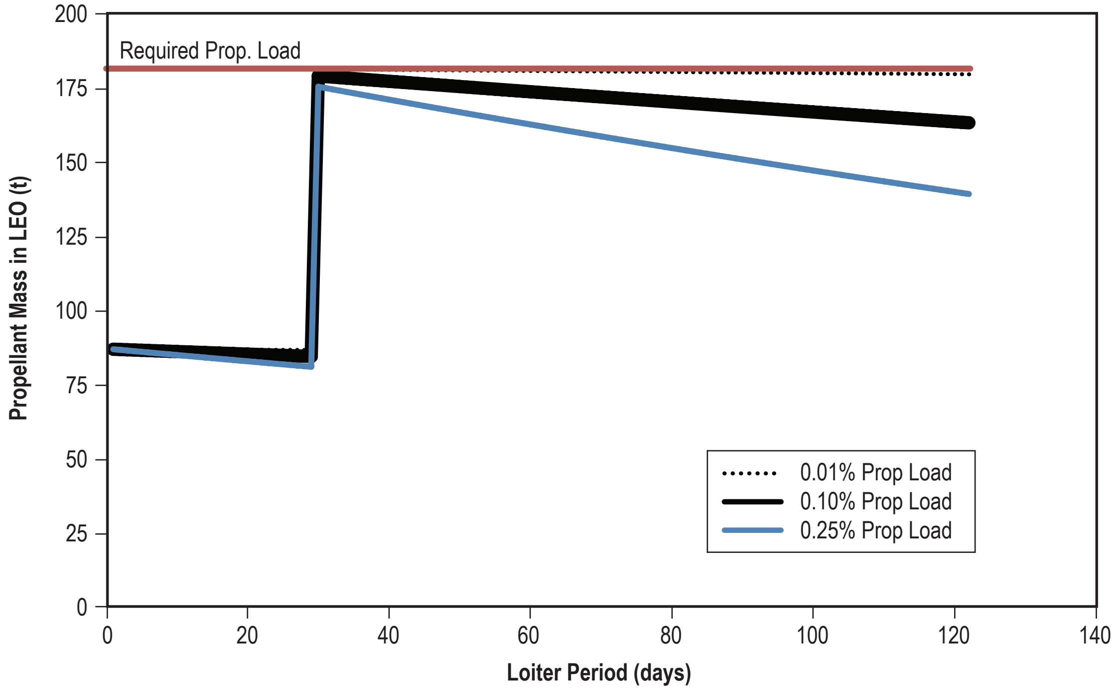

Figure 39. Mass loss of various boiloff rates for NTP crewed MTV.

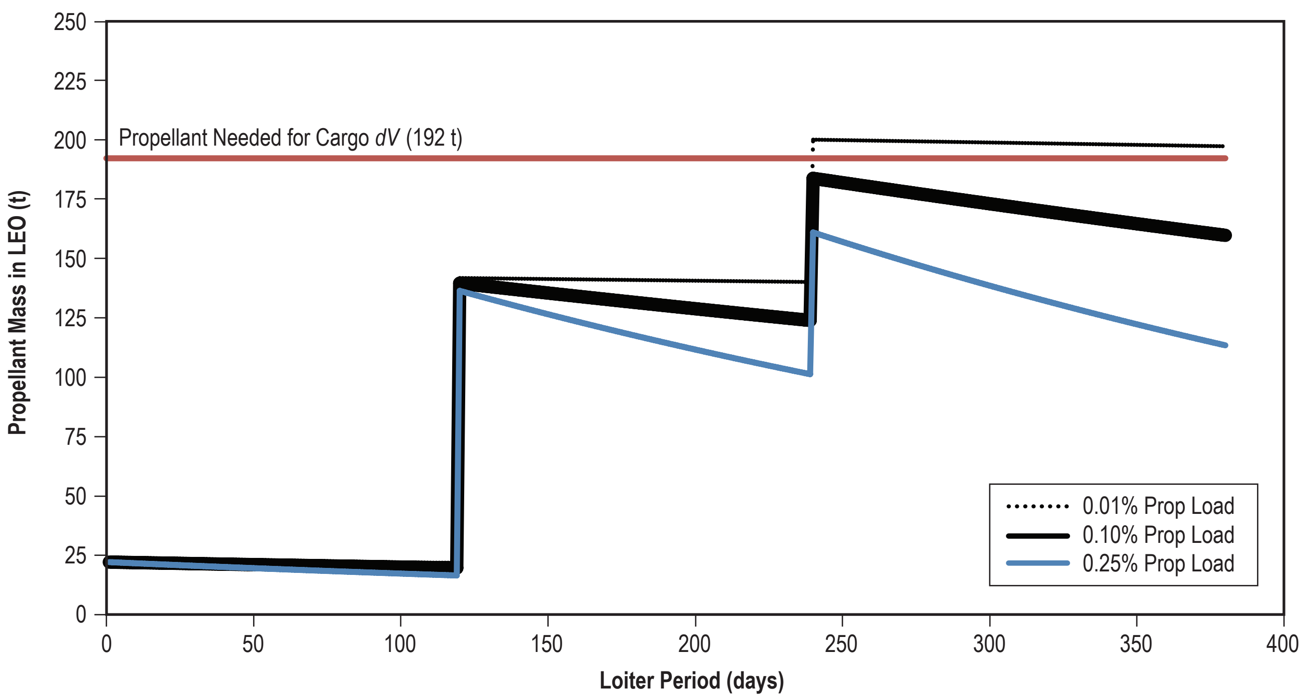

Figure 40. Mass loss of various boiloff rates for chemical cargo MTVs.

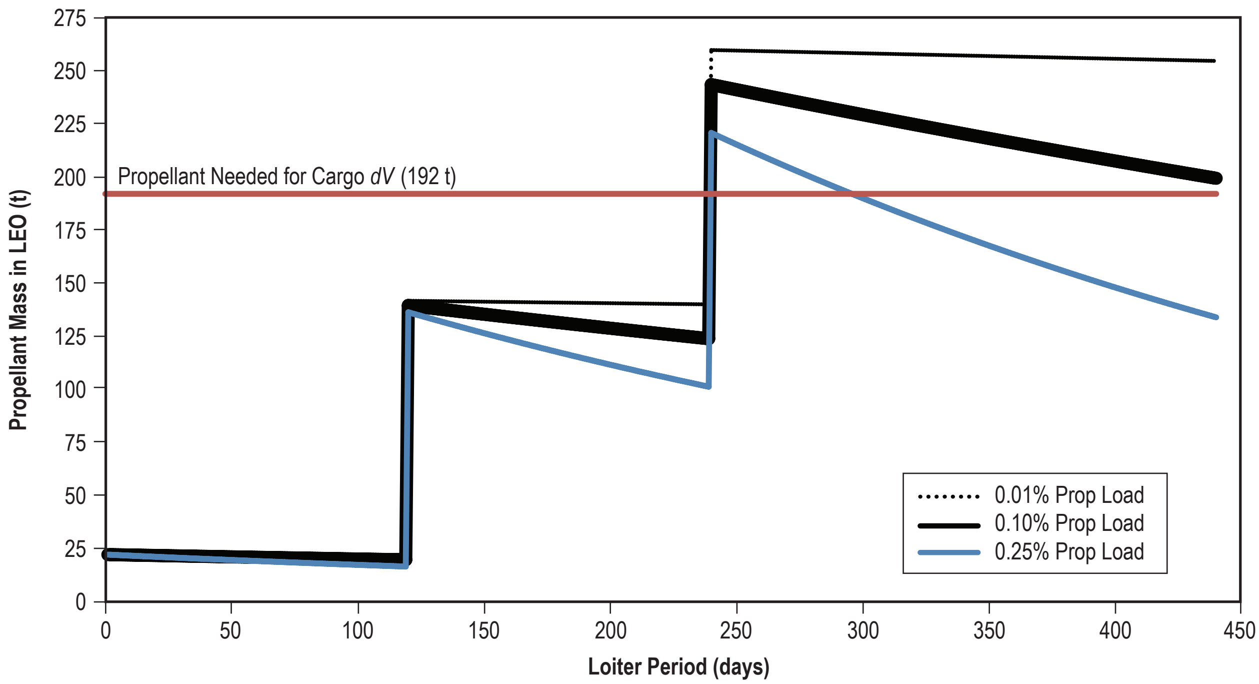

Figure 41. Benefit of adding sixth launch to cargo MTV assembly sequence.

the mission. The NTP crewed MTV assembly sequence is shown figure 39, where about 87 t of LH_2 propellant is delivered on the first launch and then about 94 t of LH_2 propellant is delivered on the second launch. It can be seen that a boiloff rate as low as 0.01% per day would be close to meeting mission requirements, but a rate greater than that would cause the MTV to be significantly short of the propellant required to meet the *dV* target.

For the chemical MTV option, the same relationship basically holds true. As shown in figure 40, a boiloff rate of 0.01% would allow for the cargo MTV to also meet its *dV* requirement.

For the chemical option though, a sixth dedicated launch could be added in order to have two dedicated propellant tankers delivering LOX/LH_2 to the two cargo MTVs. This could be planned as an option to decrease the mission sensitivity to ZBO, but still plan to have ZBO developed to decrease the number of launches required for the mission. As shown in figure 41, the boiloff allowance could be increased to 0.10% boiloff per day while still meeting the required propellant to perform the mission. Furthermore, mission planning could be arranged in such a way that the last propellant launch could arrive but transfer would not take place until the boiloff in each tank reduced the propellant load to a sufficient level that would not exceed the tank capacity of the TMI stage (or 80% fill constraint assumed during PA-C3′).

The same relationship holds true for the chemical option crewed MTV assembly sequence. Increasing from four to five launches for that MTV assembly allows for up to 0.10% boiloff per day while still meeting mission requirements (fig. 42).

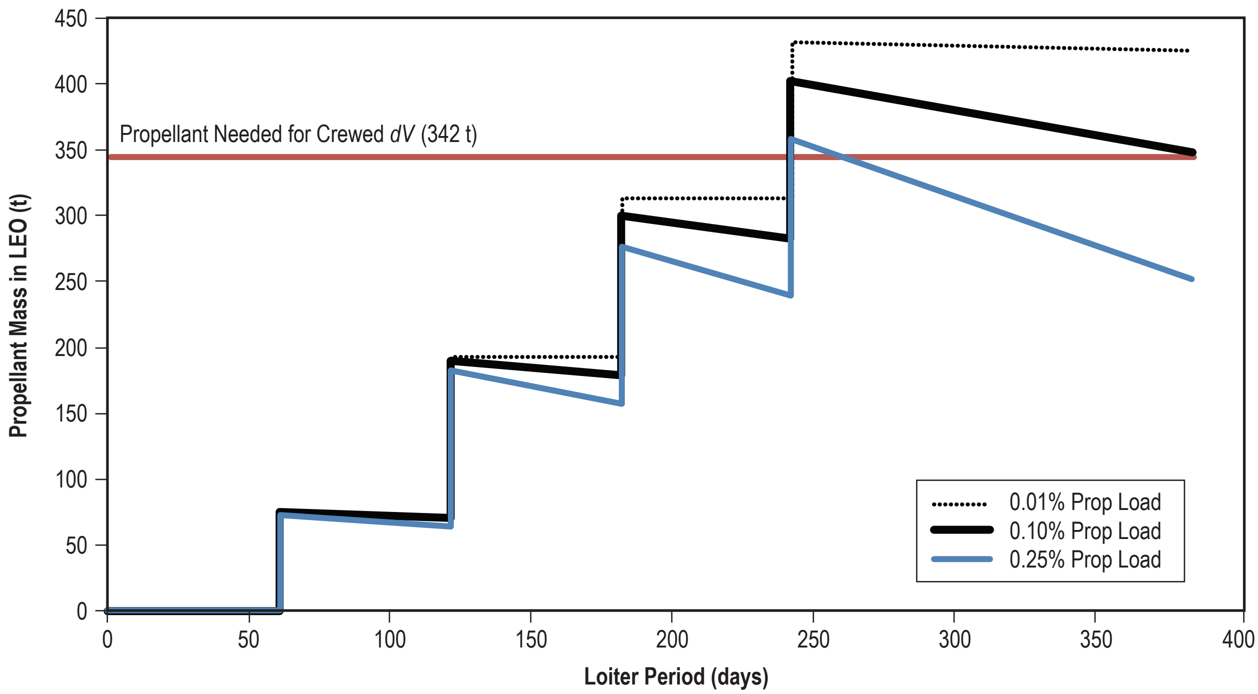

Figure 42. Benefit of adding fifth launch to crewed MTV assembly sequence.

Another integral benefit of this cryogenic propellant transfer approach is the ability to decrease sensitivity to the launch spacing requirement. DRA 5.0 assumed that launches had to occur within 30 days of one another to carry out the mission. However, propellant transfer and the ability to deliver 'top-off propellant' later in the sequence also allows for the 30-day launch center approach to be increased to either 45 days or 60 days, depending on how much boiloff the program is willing to accept and the technological development progress of ZBO systems. Figure 43 depicts sensitivity to the launch spacing requirement.

As shown in figure 43, the ability to transfer propellants onorbit can significantly reduce the sensitivity of the mission architecture to key design variables (ZBO of cryogenic propellants and spacing between launches). Furthermore, it is not required to increase the launches if further developments in either area result in ZBO or infrastructure to support increased launch rates.

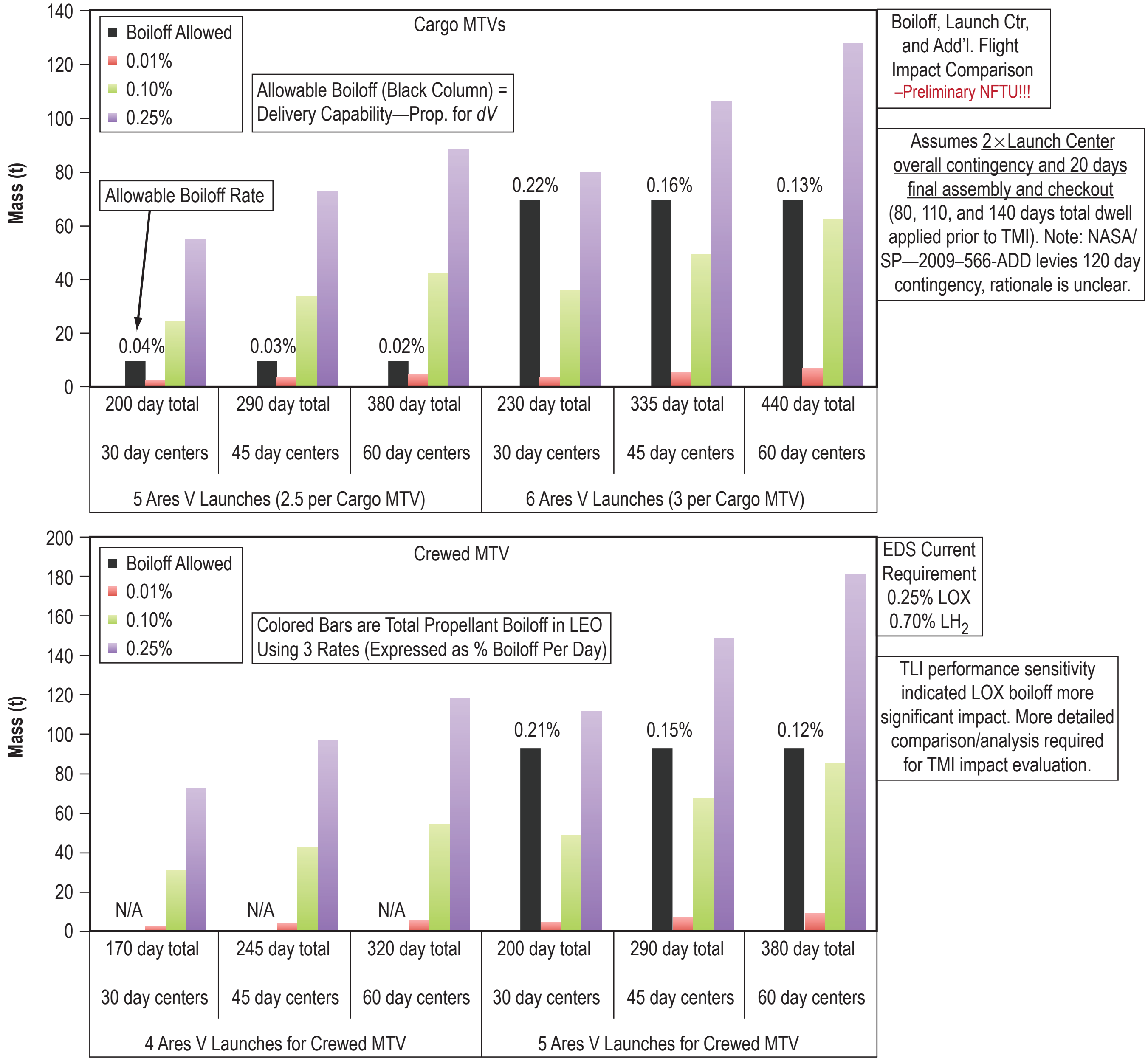

Figure 43. Benefit of adding fifth launch to crewed MTV assembly sequence.

6. LUNAR AND MARS MISSION SYNERGISTIC POTENTIAL

6.1 Infrastructure Sharing

Identifying the Mars architecture depended, in large part, on the commonalities between that and the lunar architecture already being assessed. It has been highly desirable to maintain as many commonalities as possible between the two transportation architectures in order to maximize the synergy, thereby reducing complexity and cost. As the Mars DRA 5.0 specifies, this synergy is defined by the subsystem technologies, space transportation elements, and the use of common ETO launch vehicles. Figures 44 and 45 illustrate the DRMs for the Moon and Mars, respectively, from the Ares V Ops Con (the chemical option is depicted in figure 45).

In figure 44, one may see that one Ares V (in addition to one Ares I with a crew of four) is required for an extended lunar surface stay. Once in LEO, the EDS and Altair rendezvous and dock with Orion. The EDS is employed to perform the TLI and then it is jettisoned. After the lunar stay, the TEI is performed by Orion to return to Earth.

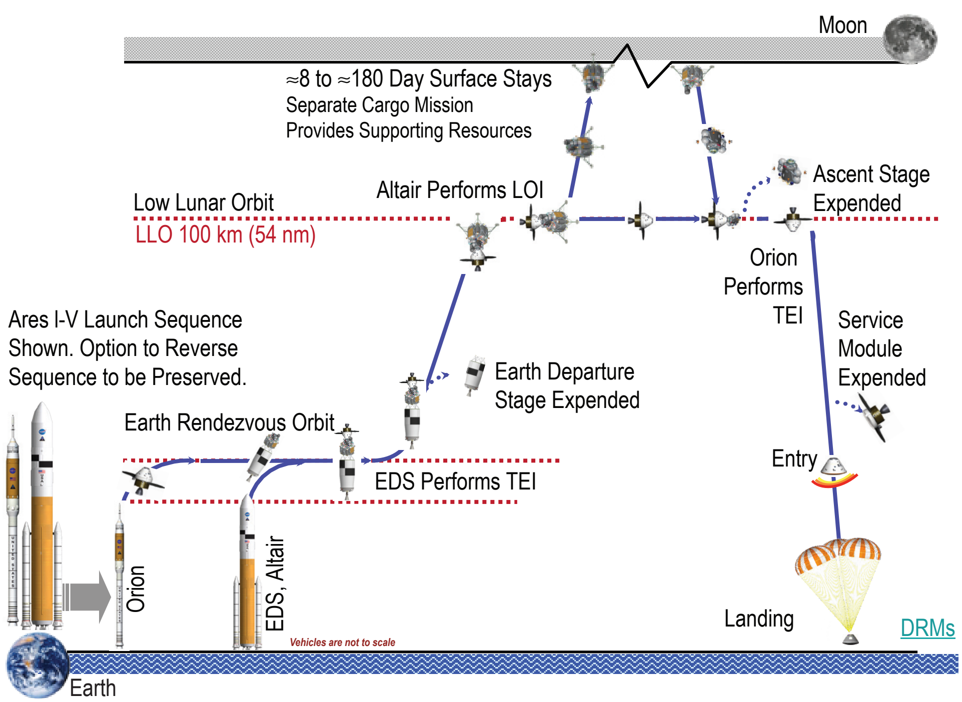

Figure 44. One-and-a-half launch lunar architecture from Ares V Ops Con.

In figure 45, a similar assembly takes place in LEO, though on a larger scale. Two cargo MTVs and one crewed MTV are assembled and loiter in LEO until enough propellant is delivered (via other Ares V launches) to perform a TMI maneuver during the next synodic opportunity. This would require 9–11 Ares V launches for the chemical option. Likewise, seven Ares V launches, plus an additional Ares I crewed launch, would be required for this proposed Mars NTP campaign.

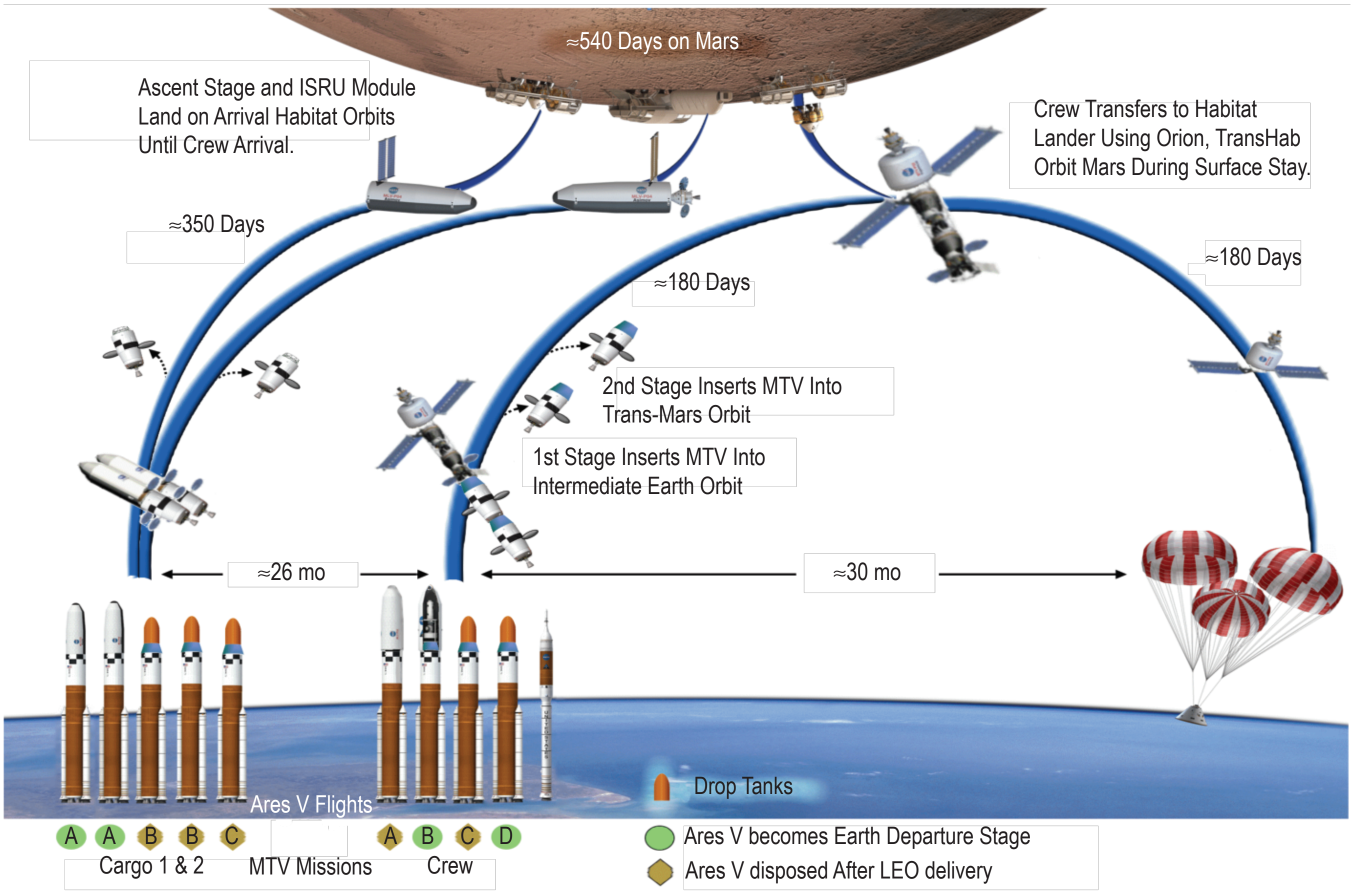

Figure 45. Chemical option (i.e., EDS-as-MTV, or unofficially scavenger option).

If systems are independently developed for Moon and Mars architectures, multiple obstacles may need to be addressed, such as delayed Mars operations due to a technology gap between the two missions. Similar to the impending Space Shuttle gap, the ability (cost and labor) to curtail lunar operations to enable Mars missions would likely occur after lunar missions. Even with a renewed investment in a Mars mission, there would be no heritage technology or experience with systems designed for such.

In order to develop a sustainable architecture and space program, technologies developed for lunar missions should carry considerable commonalities to those required for Mars missions. Several aspects of the architecture offer potential for such synergy, including a common EDS module acting as either a TLI or TMI stage. Based on driving requirements, common lunar-Mars systems may be decomposed into such elements with similar capabilities.

6.2 Earth Departure Stage as Trans-Mars Insertion Stage

There are many benefits associated with employing an EDS as the TMI module. Sections 4.2 and 5.6 of this TM address the application of EDS for Mars extensibility options in more detail. Eliminating the need to develop a new propulsive stage for TMI would have considerable benefits. The Mars DRM requires near ZBO technology to allow the cryogenic fluid management system to function during the TMI maneuvers. This technology development would prevent otherwise necessary modifications to EDS propellant tanks or MPS. Figures 46 and 47 represent upgrades made to the EDS if it would serve as a multiarchitecture stage.

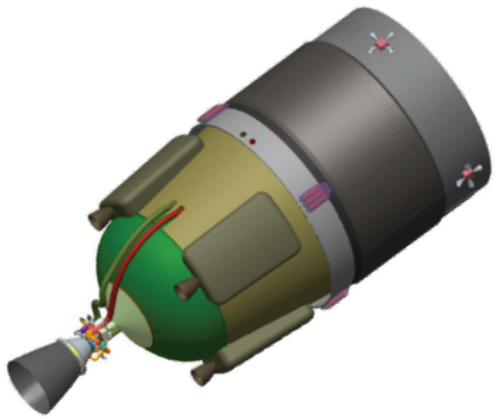

Figure 46. EDS concept for Mars architecture.

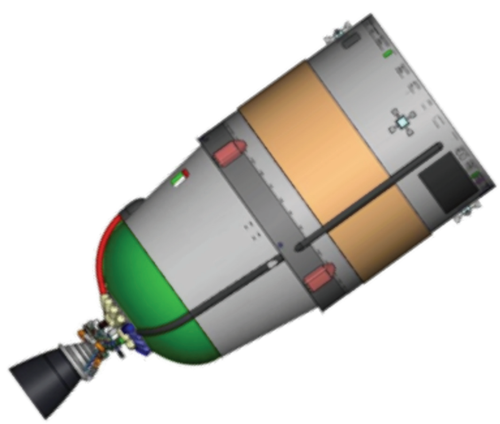

Figure 47. EDS concept for lunar architecture.

The lunar EDS and TMI stage subsystems were compared as part of the Mars DRA 5.0 study. The subsystems identified were: Trajectory and Mission Analysis, Propulsion, Structures, Thermal, Power, and Avionics. A summary of the comparison results follows.

6.2.1 Trajectory and Mission Analysis

The Mars missions would need to allow for significantly larger midcourse correction maneuvers. In addition to this, there would also be a difference seen in the main propulsion system (MPS) dV requirements. The LEO loiter time would also be extended for the Mars mission due to the time and support needed for assembly of the MTVs.

6.2.2 Propulsion

Currently, there are many similarities between the in-space stage being considered for the Mars mission and the Ares V upper stage (i.e., EDS). All modules would use the same LOX/LH_2 propellant combination, and the propulsion systems would include helium (He) and/or autogenous tank pressurization systems for the pump-fed engines. If one system is employed for both the TLI and TMI, decisions will have to be made regarding the engine choice. The current lunar EDS design uses a single J-2X engine; whereas, the Mars stages could use several RL10-B2 engines or a single J-2X. The RL-10 engine is characterized by its expander cycle that produces a nominal thrust of 24,750 lbf at an I_{sp} of 462.2 s. The current J-2X engine specifications state a much higher thrust of 294,000 lbf with a 448-s I_{sp} rating.

6.2.3 Structures

Material choice is an important factor when assessing structures, though most components would be sized by the launch and staging loads. Components of the system need to be designed for the Mars orbit environment, including possible material degradation and the effects of thermal cycling. The material choice must account for occurrences such as microcracking during launch and outgassing. MMOD protection would also be required for the Mars mission.

6.2.4 Thermal

Many of the following thermal features are common to both the lunar EDS and the Mars TMI stages:

- The heat collection and transport systems (e.g., circulating fluid systems, cold plates, and heat exchangers).
- Heat rejection systems (e.g., space radiators).
- The passive control systems (heaters, insulations, passive devices, phase-change materials, special materials and coatings, etc.).

However, the lunar EDS will be designed with passive cryogenic propellant storage while the Mars TMI stage will require ZBO active cryogenic propellant storage. The development of a two-stage cryocooler operating at 20 K for LH_2 storage will be required for the Mars mission active cryogenic propellant storage. The active approach affords essentially indefinite storage duration at the price of increased complexity, system hardware mass, power consumption, and heat rejection.

6.2.5 Power

Both the lunar EDS and the Mars TMI stages are powered by solar arrays. The lunar EDS contains a power management and distribution (PMAD) system to power itself and the lander during LEO loiter and TLI. There is no cryogenic cooling requirement on the EDS; it has an all-passive thermal system. On the other hand, the TMI stages will contain solar arrays and a PMAD system to power themselves only during LEO loiter and TMI and have no payload power capability. However, since the TMI stages are actively cryogenic cooled, they will require significantly more power than the EDS (about four times more), which means a much larger PMAD system than in the EDS. Also, because of the longer assembly and loiter time in LEO, the TMI solar arrays will need to handle more radiation degradation and MMOD damage.

6.2.6 Avionics

The following control features are common between the lunar EDS and Mars TMI stages:

- Both designs contain solar power generation and have navigation and communication components for LEO (sun and star trackers, inertial measurement units (IMUs), S-band transceivers, Global Positioning System (GPS), etc.); however, there are several differences. In particular, the instrumentation and data handling components will need to be larger in capacity in the TMI for the Mars mission to accommodate the active thermal cooling system. Also, the TMI requires an R&D system and low-gain interstage communication system for docking and assembly operations.

- The lunar EDS avionics system provides the GN&C capabilities of the stack during LEO loiter. It has a one-fault-tolerant system since the CEV takes over navigation control after docking, providing a two-fault-tolerant system for the crewed TLI operation. The Mars TMI stage avionics systems also consists of a one-fault-tolerant GN&C system to navigate the stage to assembly orbit and dock with the stack, but the reboost module will provide GN&C for the stack during assembly and testing.

- The TEI stage will be the controlling element during TMI burns and thereafter and will provide the two-fault-tolerant GN&C that is required with a crew aboard.

These operational differences mean that the two avionic systems will have differences in system architecture, interconnections, and software.[2]

6.3 Opportunity to Engage Commercial and International Partners

The role of global partnerships has evolved as the space industry has matured. Fueled by a competitive surge during the Cold War, America and the Soviet Union were incentivized to become the first nation in space. Since then, many actions have been taken to foster healthier relationships with space industries of fellow nations. The current program of record includes a gap in which the United States will rely on Russia to provide a crew-launch capability to the ISS, as needed. Table 15 illustrates current international relationships including multiple Space Station endeavors, signed treaties, and promised cooperation on projects such as satellite assembly and maintenance.

Table 15. Current sample of international relationships.

	China	Russia	India	United States	Canada	Brazil	Japan	Thailand	France
China		Jointly tabled a draft treaty on the prevention of weapons and use of force in outer space. Possible joint venture on the study of the Moon.		Have agreed to discuss expanded cooperation and to start a dialogue on human space flight and exploration.		Signed agreement to provide Africa with data from jointly held "Earth resources satellite" through ground station in South Africa, Egypt, and Spain.			Developed extensive space exchanges and cooperation including progress in space science, Earth science. life science, satellite application, and satellite research.
Russia	Jointly tabled a draft treaty on the prevention of weapons and use of force in outer space. Possible joint venture on the study of the Moon.		Space agreements signed for Global Navigation Satellite System (GLONASS).	Apollo-Soyuz, ISS (main players).			Interagency agreement drafted for a joint space program.		
India		Space agreements signed for GLONASS.		Next Steps in Strategic Partnership (NSSP)**; joint statement pledging to build closer ties in space exploration, satellite navigation, and commercial space; Chandrayaan-1.				THEOS*	Joint mission to understand tropical weather phenomena.
United States	Have agreed to discuss expanded cooperation and to start a dialogue on human space flight and exploration.	Apollo-Soyuz, ISS (main players).	Next Steps in Strategic Partnership (NSSP)**; joint statement pledging to build closer ties in space exploration, satellite navigation, and commercial space; Chandrayaan-1.				ISS		
Canada							Space panel regularly meets.		
Brazil	Signed agreement to provide Africa with data from jointly held "Earth resources satellite" through ground station in South Africa, Egypt, and Spain.								
Japan		Interagency agreement drafted for a joint space program.		ISS	Space panel regularly meets.			THEOS*	
Thailand			THEOS*				THEOS*		
France	Developed extensive space exchanges and cooperation including progress in space science, Earth science, life science, satellite application, and satellite research.		Joint mission to understand tropical weather phenomena.						

*THEOS: First remote sensing satellite that shares timely and indepth information with Sentinal Asia, a regional project on satellite information for disaster management.
**NSSP: Proposed expanded engagement on civilian nuclear activities, civilian space programs, and high-technology trade based on a series of reciprocal steps.

Such exploration missions as the Moon and Mars would undoubtedly benefit from international collaboration and commercial partnerships. There is considerable opportunity to not only foster the growth of commercial industry but to sustain it with technology that crosses the border between one mission architecture and the next. The design of such missions would require highly standardized components and processes, where the designing, development, manufacturing, and testing are prioritized such that all partners agree, in turn minimizing system complexity and cost for each partner. Furthermore, it is an unprecedented opportunity to involve and inherently incorporate the capabilities of the commercial industry and international partners in an overall Mars mission strategy.

6.4 Enabling Alternate Missions

This synergy would allow for opportunities to directly validate Mars elements during lunar missions (while encouraging advanced technological development for use on the Moon) instead of postponing development for Mars missions. The technologies employed herein may also have real-world applications. There is experience to be gained in routine manufacturing and system operation that would decrease risk and improve reliability.

6.5 Public Perception

A seamless transition from lunar to Mars architecture would also prevent workforce disruption. Such a direct correlation between Moon and Mars exploration would have highly visible milestones and applicability to more than one goal. Capabilities for flexible implementation allow the public to foresee the technology potential in future missions and for multiple mission opportunities. Without singularly focused technology development, these capabilities may easily transfer to Earth-focused applications.

Commonality between lunar and Mars exploration systems would significantly accelerate the onset of Mars exploration with early development of needed technologies. This would also reduce, if not evade, any development gap between lunar and Mars missions, even allowing for simultaneous exploration on the Moon and Mars. Life-cycle costs would also be considerably decreased, and it may be possible to show the benefits of developing assets for lunar exploration that can then be applied at smaller marginal cost to NEO exploration, which can then be applied at a much reduced marginal cost for Mars exploration. In this manner, the Mars exploration strategy would not have to carry the full development cost otherwise required in a ‘Mars Forward’ strategy.

7. FORWARD WORK

The Ares V team identified several areas of forward work on the Mars DRM. In addition to updating the requirements data set for a Systems Requirements Review (SRR), further developing the Ares V Operational Concepts Document, creating functional flow block diagrams of required functionality/capabilities, and increasing the team's exposure to the Mars DRM, several areas of specific focus were identified. These include the trajectory work, vehicle subsystems, the ground operations concept, the Safety and Mission Assurance (S&MA) team, and the synergistic potential between Mars and NEO DRMs.

7.1 Ares V Performance—Trajectory

Current performance estimates looked at both direct insertion circular cases and 120-nmi perigee insertion orbits. The elliptical orbit perigee was chosen to be high enough to allow EDS and payload to be inserted into a stable orbit. However, additional work is necessary to establish the potential performance gain from lowering the insertion orbit perigee below 120 nmi. In the extreme, this would result in negative perigee altitudes, with the optimal perigee being traded between ascent performance improvement and the propellant necessary to circularize the stack. Depending on the payload being flown, an additional constraint may be necessary to ensure the vehicle is inserted into an altitude sufficient to keep heating rates below payload requirements. Additionally, lower altitude perigees allow for the EDS to be disposed of without a deorbit maneuver, provided the EDS will not be needed on orbit. Further work will be necessary to consider how reliability and payload concerns affect the trade of insertion orbit used, as these concerns may not allow for a nonstable insertion orbit to be used. Initially, the IMLEO study included a –10-nmi-altitude perigee, but those cases were not run.

Although the EDS may not be needed for on-orbit operations, it may still be necessary to insert it into a higher perigee stable orbit due to payload and operational considerations. In these cases, it would be necessary to perform a disposal maneuver from the elliptical insertion orbit. For the current analysis, only disposal from the final circular orbit was considered. Further work will be necessary to develop the elliptical orbit disposal dV budget and incorporate the performance impacts into the LEO performance capability.

7.2 Earth Departure Stage—Subsystems

There are opportunities for continued work in the EDS subsystem realm. Possible subsystem mass impacts to EDS may be evaluated, since 30 t was assumed for an EDS burnout mass (post-TMI) for the purposes of dV, payload, and total mass assessments. There is also a 5-t placeholder (in addition to the 30 t) for required hardware to dock EDS and mechanisms required to transfer propellant. This implies an 18% increase over the lunar EDS burnout mass (post-TLI), which was approximately 25.5 t. This additional mass required for Mars missions is due to RCS/ACS and/or auxiliary propulsion, ZBO power and cryogenic coolers, structural mass impacts for larger payloads and shrouds, and ensured disposal of the empty EDS (for a two-burn TMI). Currently, there is no heritage technology that supports such ZBO capabilities, though it is necessary to enable the mission. In addition, in-space cryogenic propellant transfer is also of interest, since the proposed chemical propulsion options would utilize the EDS and reduce the number of Ares V launches. Future work may entail adding additional margin into the mission architecture by appropriately characterizing these fundamental technologies.

The engine of the EDS may also improve with continued analysis. The current J-2X burn duration requirements could be modified to burn for ≈9–10 min for ETO ascent followed by a required burn for ≈10–11 min for TMI. The total burn duration would then be on the order of 20 min; however, further analysis is needed.

7.3 Shroud

Further work might include identifying the impact of required shrouds. Perhaps a larger shroud would be needed to encapsulate both the TransHab and MOI/TEI modules. There is also the opportunity to assess impacts of maintaining shroud to orbit in a "multiuse shroud" case. Previous studies show an approximate 80% retained shroud mass impact to payload (1 lbm of shroud mass retained to orbit results in 0.8 lbm of payload reduction).

7.4 Mars Campaign Ground Operations

Forward work on Mars campaign ground operations are as follows:

- Continue following the vehicle design and modify the Ops Con and architecture as appropriate. Should the vehicle outgrow the VAB physical constraints, the vehicle architecture changes, or the GOP determines the Ops Con is not feasible for any other reason, alternate ground-system architectures will need to be determined and evaluated.

- The GOP will continue making updates to the Ares V timeline and rerunning the flight rate and launch spacing models as applicable.

- The option of a dual-use pad should be studied to determine its feasibility and impacts on the Mars manifest.

- Additional studies to understand facility modification periods and their effects on availability are needed. Commodity (LH_2, helium, etc.) quantities and availability for a higher manifest need to be studied further to determine impacts to launch availability. The amount of commodities required could drive additional ground systems and infrastructure to support a Mars campaign.

- Should launch site storage of any flight elements be required due to launch schedule delays or supply-chain production that is faster than integration/use, additional studies would be needed to look at the impacts to the GOP architecture to support such changes.

- Nuclear processing at KSC and a better understanding of ground processing requirements as related to nuclear stages also needs further study.

- Continued studies of the Ares V ML launch mounts are needed to coincide with future development of OSF operations to address feasibility of the preliminary proposed architecture. This study should include a more detailed analysis of ML launch mount feasibility and its impacts on the proposed Ops Con.

- Concurrent support of other missions, such as simultaneous lunar missions or other payloads, may affect the analysis and results.

- A discrete event simulation (DES) will provide better results by including common cause variability, modeling interactivities between VAB HBs, and accounting for delay probabilities. A DES will likely indicate requirements for additional ground systems resources to accommodate more realistic manifest performance.

7.5 Safety and Mission Assurance

Within the context of transportation, LEO assembly, and MTV TMI, there are a number of areas that should be addressed to allow the Mars campaign risk assessment to better reflect likely risks. The first such area is component reliability maturation. The risk figures used in this assessment assume fully matured technology and do not account for the reduced reliability associated with technology introduced early in its production cycle. A comprehensive study would include the effects of liquid and NTP engine reliability maturation, with sensitivity to the number of precursor and test flights with those engines.

The second area is the impact of launch and operational delays on campaign risks (e.g., the accumulation of launch delays that force an MTV to miss its TMI window or cause excess loiter and total boiloff to exceed the propellant margin). While the notion of relaunch in the event of a tanker launch failure was discussed above; relaunch of Ares V is not addressed.

Finally, given the large loiter risk and the high value of MTV assets, repair in LEO may become a necessary contingency. When considering the full extent of loss-of-campaign risk (and other FOMs) and its impact on campaign architecture, the post-TMI campaign elements of trans-Mars cruise, Mars vicinity and surface operations, Earth transit, and Earth entry, descent, and landing should be represented at the very least in the form of a proxy in the risk model. All of these areas constitute elements that are under consideration for inclusion in a campaign risk simulation model currently under development. Loss-of-crew risk will also be assessed.

7.6 Mars and Near-Earth Object Synergistic Potential

Similar to lunar and Mars synergy, there is potential for forward work when evaluating NEO extensibility options. Similar to the Mars architecture, a NEO mission would depend in large part on the commonalities between that and the Mars (and potentially lunar) architecture already in place or being assessed. Delta-velocity requirements would be assessed and compared to other potential DRMs, including details of the Earth-to-NEO transfer *dV* and the rendezvous and eventual Earth-return (TEI) dV. Propellant transfer evaluation for a NEO DRM may emphasize similar technologies employed for a Mars DRM such as the decision to use in-space cryogenic propellant transfer. Propellant fill efficiency and cryogenic propellant boiloff impact has been assessed for the Mars DRM up to 5% propellant mass loss and/or 0.25% average boiloff rate per day, respectively.

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REPORT DOCUMENTATION PAGE

Form Approved OMB No. 0704-0188

1. REPORT DATE *(DD-MM-YYYY)*: 01–11–2010

2. REPORT TYPE: Technical Memorandum

3. DATES COVERED *(From - To)*:

4. TITLE AND SUBTITLE: Ares V Utilization in Support of a Human Mission to Mars

5a. CONTRACT NUMBER:

5b. GRANT NUMBER:

5c. PROGRAM ELEMENT NUMBER:

5d. PROJECT NUMBER:

5e. TASK NUMBER:

5f. WORK UNIT NUMBER:

6. AUTHOR(S)

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7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

George C. Marshall Space Flight Center
Marshall Space Flight Center, AL 35812

8. PERFORMING ORGANIZATION REPORT NUMBER: M–1299

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

National Aeronautics and Space Administration
Washington, DC 20546–0001

10. SPONSORING/MONITOR'S ACRONYM(S): NASA

11. SPONSORING/MONITORING REPORT NUMBER: NASA/TM—2010–216450

12. DISTRIBUTION/AVAILABILITY STATEMENT

Unclassified-Unlimited
Subject Category 15
Availability: NASA CASI (443–757–5802)

13. SUPPLEMENTARY NOTES

Prepared by the Science & Exploration Vehicle Office, Science & Mission Systems Office
*Zero Point Frontiers Corp., **Jacobs ESTS Group/Triumph Aerospace, ***Glenn Research Center,
†Kennedy Space Center, ‡Ames Research Center

14. ABSTRACT

During the analysis cycles of Phase A-Cycle 3 (PA-C3) and the follow-on 8-wk minicycle of PA-C3′, the Ares V team assessed the Ares V PA-C3D configuration to the Mars Design Reference Mission as defined in the Constellation Architecture Requirements Document and further described in Mars Design Reference Architecture 5.0 (DRA 5.0) that was publicly released in July 2009. The ability to support the reference approach for the crewed Mars mission was confirmed through this analysis (7-launch nuclear thermal propulsion (NTP) architecture) and the reference chemical approach as defined in DRA 5.0 (11- or 12-launch chemical propulsion module approach). Additional chemical propulsion options were defined that utilized additional technology investments (primarily in-space cryogenic propellant transfer) that allowed for the same mission to be accomplished with 9 launches rather than the 11 or 12, as documented in DRA 5.0 and associated follow-on activities. This nine-launch chemical propulsion approach showed a unique ability to decouple the architecture from major technological developments (such as zero-boiloff technology or the development of NTP stages) and allowed for a relaxing of the infrastructure investments required to support a very rapid launch rate (30-day launch spacing as documented in DRA 5.0). As an enhancing capability, it also shows promise in allowing for and incorporating the development of a commercial market for cryogenic propellant delivery on orbit, without placing such development on the critical path of beyond low-Earth orbit exploration. The ability of Ares V to support all of the aforementioned options and discussion of key forward work that is required to fully understand the complexities and challenges presented by the Mars mission is further documented herein.

15. SUBJECT TERMS

Mars, launch vehicle, Ares V, EDS, Mars mission, scavenger, chemical campaign

16a. REPORT	16b. ABSTRACT	16c. THIS PAGE	17. LIMITATION OF ABSTRACT	18. NUMBER OF PAGES
U	U	U	UU	90

16. SECURITY CLASSIFICATION OF: (see table above)

19a. NAME OF RESPONSIBLE PERSON: STI Help Desk at email: help@sti.nasa.gov

19b. TELEPHONE NUMBER *(Include area code)*: STI Help Desk at: 443–757–5802

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