Utilizing Lunar Architecture Transportation Elements for Mars Exploration

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In 2004, President Bush announced a new U.S. Vision for Space Exploration that includes plans to return humans to the moon followed by human missions to Mars. While human Mars missions have been studied and analyzed for decades, the current technical and political environment presents mission designers with new objectives and constraints. Given the significant investment required to develop new launch vehicles, habitat systems, and supporting technologies for the preparatory lunar campaign, it is likely that those systems will serve as the centerpiece of any future Mars exploration architecture. This paper summarizes the efforts of SpaceWorks Engineering's advanced design team to develop a viable Mars architecture based on Project Constellation launch vehicles and related lunar transportation technologies. All-chemical LOX/LH2 transfer vehicles are used. No in-situ resource utilization (propellant manufacture) has been assumed. An overall concept of operations is outlined. Details are provided on element masses, Earth-Mars transfer times, development and operations costs, and estimated mission reliability. Throughout this internally-funded effort, emphasis has been placed on maturing design tools and multidisciplinary processes in order to develop a useful national capability should formal studies of Mars architectures be undertaken.

Nomenclature

ASRG	=	Advanced Stirling Radioisotope Generator	MEV	=	Mars Excursion Vehicle
CER	=	Cost Estimating Relationship	MLI	=	Multi-Layer Insulation
CEV	=	Crew Exploration Vehicle	MOI	=	Mars Orbit Insertion
DDT&E	=	Design, Development, Testing & Evaluation	MOR	=	Mars Orbit Rendezvous
ECLSS	=	Environmental Control & Life Support System	MSAT	=	Mission Scenario Analysis Tool
EDL	=	Entry, Descent, and Landing	NTR	=	Nuclear Thermal Rocket
EDS	=	Earth Departure Stage	OMS	=	Orbital Maneuvering System
ERV	=	Earth-Return Vehicle	PICA	=	Phenolic Impregnated Ceramic Ablator
ETO	=	Earth To Orbit	POST	=	Program to Optimize Simulated Trajectories
EVA	=	Extra-Vehicular Activity	PPM	=	Power and Propulsion Module
ISPS	=	In-Space Propulsion Stage	RSRB	=	Reusable Solid Rocket Booster
ISRU	=	In-Situ Resource Utilization	TEI	=	Trans-Earth Injection
LCC	=	Life Cycle Cost	TFU	=	Theoretical First Unit
LEO	=	Low Earth Orbit	TMI	=	Trans-Mars Injection
LMO	=	Low Mars Orbit	TPS	=	Thermal Protection System
LOC	=	Loss of Crew	VSE	=	Vision for Space Exploration
LOM	=	Loss of Mission			

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I. Introduction

ANNOUNCED in 2004 by President George W. Bush, the U.S. Vision for Space Exploration (VSE) outlined steps for refocusing NASA on the goal of space exploration. The VSE charges NASA with the responsibilities of completing the space station, retiring the space shuttle by 2010, building a new Crew Exploration Vehicle (CEV), returning humans to the moon no later than 2020, and establishing a foundation for eventually sending humans to Mars and beyond. In support of a future lunar exploration campaign, work is already well underway on the new Orion CEV and the new Ares family of launch vehicles. Advanced studies are being conducted throughout government and private industry on designs for potential lunar landers, lunar habitats, next-generation space suits, and lunar surface operations.

Given the series of technical and political accomplishments that must precede an eventual human Mars mission, it may seem premature to invest significant resources in the study of potential Mars architectures. By current schedules, human Mars exploration would not take place before 2030, if then. However, even at this early stage, it is very prudent for the advanced design and space systems analysis communities to consider the basic form and function of a future Mars architecture. Advanced studies can help determine overall concepts of operation, mission durations, basic transportation element sizes and shapes, and rough cost and reliability estimates. This information can help guide near-term policy decisions. In addition, advanced architecture studies help identify long-range technology requirements and therefore influence near-term technology investment decisions.

This paper introduces a notional long-stay Mars architecture developed internally by the Advanced Concepts and the Engineering Economics groups at SpaceWorks Engineering, Inc. (SEI). Details of the transportation system are presented including payload masses, stage masses and sizes, launch vehicle requirements, mission parameters, and overall cost and reliability estimates. A notional four-mission Mars exploration campaign is presented starting in 2030. Key technology requirements for the baseline architecture are identified and presented as long-range investment priorities.

While conducting this study, our team has placed emphasis on maturing our disciplinary analysis capabilities in preparation for any more formal study efforts that might be conducted by NASA and its prime contractors. We have developed a fast-acting multidisciplinary process that integrates interplanetary trajectory optimization, transfer stage sizing, launch vehicle constraints, parametric habitat scaling, Mars lander/entry vehicle design, cost analysis and quantitative mission reliability prediction. The Mission Scenario Analysis Tool (MSAT) allows architecture element data to be shared amongst supporting disciplines in a tightly-integrated and consistent manner that simplifies architecture ‘closure’. Outputs from one parametric or engineering model is passed to the next and internally iterated if needed. MSAT is a unique national capability that could be leveraged for additional study work and analysis of candidate architectures[1].

II. Study Assumptions and Guidelines

Studies of human Mars exploration architectures are certainly not new. Many, many studies have been conducted dating back to the 1950’s.[2-6] Studies vary in fidelity and completeness, but most significant differences in the findings and outcomes are due to the study’s starting assumptions and guidelines. For this internal study, we developed a set of starting assumptions that we believe are most consistent with the political and technical environment that currently exists in the United States (circa 2007).

A. Availability of Lunar Campaign Elements and Supporting Technologies

We assume the availability of major elements from NASA’s Project Constellation development effort.[7] Given the significant investment being made to develop those flight elements, it is likely that they will serve as the centerpiece of any future Mars transportation system. Specifically, we assume the existence of the Orion CEV as a crew module capable of transporting crew from Earth’s surface to Low Earth Orbit (LEO) and back. Further, we assume that a modified, downsized CEV design can be used as an Earth-Return Vehicle (ERV) for Mars missions.

The Ares I launch vehicle is assumed to provide a highly reliable method for launching a crew to LEO. The Ares V launch vehicle is assumed to be the primary workhorse for delivering Mars-bound cargo to LEO and eventually on a Mars transfer mission using its Earth Departure Stage (EDS) upper stage. We do assume that the fairing diameter of the Ares V can be increased to 10m to accommodate a large rigid aerobrake. This configuration is not reflected in the baseline Ares V concept, but has been studied internally. The J-2X LOX/LH2 engine is assumed to have been developed and be available for use on the EDS. In addition, we assume that RL10B-2 engines or similar hardware will still be available from commercial sources since they form the basis of the current lunar lander descent stage concept.

Technologies related to closed environmental control and life support systems (ECLSS) for surface habitats is assumed to have been developed to support the lunar campaign and is therefore available for a follow-on Mars exploration campaign. Closed ECLSS systems are assumed to encompass both the gas (oxygen/CO_2) cycle and water cycle. These technologies are assumed to be adaptable to both surface habitats (partial-g) and in-space habitats (zero-g).

Inflatable in-space habitat technology, while not specifically being developed to support the lunar exploration campaign, is nevertheless assumed to exist and be available for use as a Mars transfer habitat. NASA has significant history with inflatable habitat technology (TransHab) and has recently licensed this technology to Bigelow Aerospace for use as Earth-orbiting hotel structures.[8-9]

B. Architecture Assumptions

In addition to the guiding assumptions regarding the transportation elements and technologies given above, the following assumptions have been made to help define the notional architecture:

1. Conjunction Class Interplanetary Mission: A long-stay mission class was assumed with surface stay times on the order of 500 days. In contrast to a short-stay or opposition-class mission, the long-stay approach yields lower energy transfers from the Earth to Mars and back and lower atmospheric entry velocities. However, the penalties of this approach include a larger crew consumables requirement, increased radiation exposure due to longer in-space mission times, and negative impacts on mission reliability due to the long mission durations. For this mission class, transfer opportunities exist roughly every 26 months. Specific launch dates and mission parameters were determined analytically.

2. Crew Size of Three (3) Astronauts: We assume an Apollo-style crew complement is sufficient to handle the flight duties, vehicle/habitat maintenance, and science duties of the mission. Other investigators have supported the notion of larger crew sizes, but obviously there is a mass and volume penalty for each additional crew person (chiefly consumables and habitat volume).

3. Sortie-type Missions (no permanent base): Each complete mission is assumed to visit a different site on the surface of Mars and must therefore be self-contained. There is no buildup of a surface base with repeat visits to the same location. Our sortie-type assumption is consistent with a science-driven campaign (versus a colonization strategy).

4. No Nuclear Thermal Rocket (NTR) Engines for Transfer Vehicles: We assume that political forces will favor a non-nuclear transfer vehicle. We therefore assume that all transfer, descent, and ascent stages will use conventional LOX/LH2 chemical propellants and rocket engines. This is despite the recognition of the technical advantages attributable to NTR in terms of increased engine specific impulse. We do, however, assume that small nuclear radioisotope based power units will be available as an alternative to solar power arrays for use on rovers, Mars entry vehicles and habitats.

5. No In-situ Propellant Production: No ISRU (In-Situ Resource Utilization) or local production of liquid propellants (typically Methane) or buffer gasses (N_2, O_2) is assumed for this study. By eliminating ISRU, we eliminate a development and operational risk associated with this technique, as well as the costs to demonstrate the technology as a precursor mission. We recognize that the availability of in-situ propellants would have a significant and positive impact on the required transportation system, but the current funding climate at NASA does not place a high priority on developing a mature propellant production capability on Mars by 2030.

6. Aerobraking at Mars: It is assumed that aerocapture/aerobraking maneuvers at Mars will be a viable technique for human-class entry masses by 2030 using rigid (non-inflated) aerobraking structures. Today, this remains a developing technology, but recent successful uses of the technique by robotic probes gives confidence that it will be available for use by human-class Mars missions.

7. Zero-boiloff Cryogenic Storage: Any all-chemical propellant architecture will depend on the ability to retain and use any propellants brought from Earth. For this study, boil-off of cryogenic hydrogen and oxygen is assumed to be minimized through the application of advanced cryocoolers and refrigeration technologies. Since boil-off is also a challenge to the lunar campaign, we assume that a viable technical solution can be developed for Mars missions by 2030.

8. No cryogenic propellant transfer: We did not assume the existence of orbiting propellant depots and do not allow the transfer of propellants between stages once they are in orbit. Orbiting refueling stations can have a large positive impact on all-chemical architectures, but would require a significant infrastructure investment and a new technology development (zero-g propellant transfer).

9. Mars Communications/Navigation Assets Exist: Individual spacecraft and/or spacecraft constellations are assumed to have been placed in Mars orbit to serve as navigation aids and communications relays in support of a human exploration campaign. Our cost estimates assume an operations cost for using these assets, but we do not account for their initial development and deployment costs.

III. Proposed Mars Architecture

As mentioned previously the Mars architecture outlined in this paper leverages existing hardware and subsystems whenever possible to reduce required new technology infusion. Existing hardware includes those elements that are assumed to be developed under NASA's Project Constellation lunar exploration campaign as well as technology/hardware that is currently available (circa 2007).

The proposed Mars architecture consists of a conjunction-class sortie mission with separate cargo and a crew transfer phases (Figure 1). The cargo transfer phase begins with two Ares V launches to LEO. The first places the Trans-Mars Injection (TMI) stage in Earth orbit, while the second lifts an In-Space Propulsion Stage (ISPS), the Mars-Earth leg return TransHab, and the Earth-Return Vehicle. The ISPS provides the require Mars Orbit Insertion (MOI) and Trans-Earth Injection (TEI) delta-V's. The TMI stage and ISPS/TransHab/ERV stack rendezvous in LEO and wait for the Earth-Mars departure window to open. After TMI, the ISPS/TransHab/ERV stack undocks from the spent TMI stage. Solar arrays on the TransHab provide keep-alive power to itself, the ISPS, and ERV during the transit to Mars. During the Martian close approach the MOI burn is performed by the ISPS stage and the ISPS/TransHab/ERV stack remains in Low Mars Orbit (LMO) until it is needed for the crew's return trip to Earth. For all missions, the cargo leg occurs in the Earth-Mars transfer opportunity preceding the crew transfer.

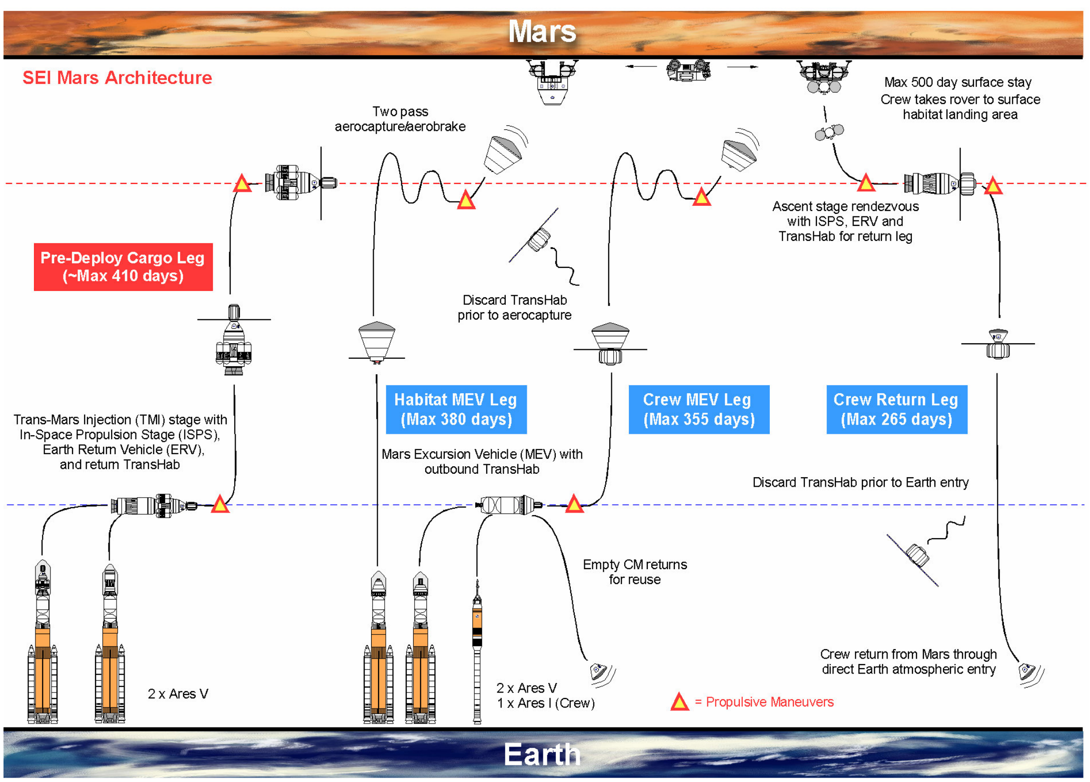

Figure 1: Proposed Mars Exploration Architecture

The crew transfer requires two Ares V and a single Ares I launch. The first Ares V launch places a Mars Excursion Vehicle (MEV) containing the Martian surface habitat and its descent stage directly on a TMI trajectory. Power during the transfer is provided by solar arrays on a power and propulsion module attached to the MEV. When the MEV arrives at Mars an aerobraking / aerocapture maneuver is conducted which places the MEV in a circular LMO. From there a terminal descent maneuver is initiated which culminates with the descent stage and surface habitat arriving on the Martian surface.

The second Ares V launch places another MEV and the crew's Earth-Mars outbound TransHab in LEO. This MEV contains an ascent stage with a minimal crew habitat, descent stage, and a pressurized rover. Placing this payload in LEO only consumes a portion of the propellant in the Earth Departure Stage. The remaining EDS propellant provides the required TMI delta-V after the crew is transferred to the outbound TransHab from the CEV. The CEV and crew are delivered to LEO by an Ares I. During the transfer to Mars the crew lives in the inflated TransHab. The TransHab also provides keep-alive power to the MEV during the transfer. When nearing Mars arrival the crew transfers to the MEV ascent stage habitat and the TransHab is jettisoned. After successfully landing on the Martian surface the crew uses the pressurized rover to drive to the surface habitat which was pre-deployed by the other MEV.

Once the surface mission is complete, the crew will again use the pressurized rover to travel from the surface habitat back to the ascent stage of the crewed MEV. The ascent stage then lifts off from the surface and performs a Mars Orbit Rendezvous (MOR) with the waiting return TransHab, ISPS, and ERV which were placed in orbit during the previous transfer opportunity. After TEI, the crew lives in the return TransHab for the trip back to Earth. Near Earth, the crew transfers into the ERV and the TransHab and ISPS are jettisoned. The ERV then performs a direct Earth entry.

A. Earth-Mars Transfer Opportunities

In order to size the individual elements of the Mars exploration architecture specific details about the required Earth-Mars and Mars-Earth transfer trajectories must be determined. The cargo transfers will use the slowest most energy efficient transfers during a given opportunity, while the crew transfers will have limits on allowable maximum transfer and surface stay times.

1. Interplanetary Trajectory Modeling

Earth-Mars transfer trajectories were modeled for this activity using Bullseye. Bullseye, which is written in Java and was internally developed at SEI, combines ephemeris data and a Lambert's problem solver to determine the interplanetary trajectory between two planets for a given time of flight. Ephemeris data for nine planets and one asteroid are included in Bullseye. Two different simulation modes can be used when running the code. The first simulates a single interplanetary trajectory (used for cargo missions), while the second simulates an outbound trajectory followed by a surface stay and then a return inbound trajectory (used for crewed missions – Figure 2). Ranges on desired departure and arrival dates, trajectory time of flights, and surface stay times are available inputs to the simulation. Bullseye will calculate the required C3 at each end of the transfer trajectory, along with a required delta-V if a given parking departure orbit or desired arrival orbit is specified. Various optimization criteria can be chosen to find the most desirable transfer trajectory. These include any of the calculated C3 values or combinations thereof, transfer times, surface stay time, or total mission time. Additionally, the maximum allowable C3 values for any transfer segment can be specified. These are used as constraints during the optimization problem and are also used to determine all feasible transfer trajectories given the desired C3 limits. These feasible transfer trajectories help determine various departure launch windows and arrival time ranges.

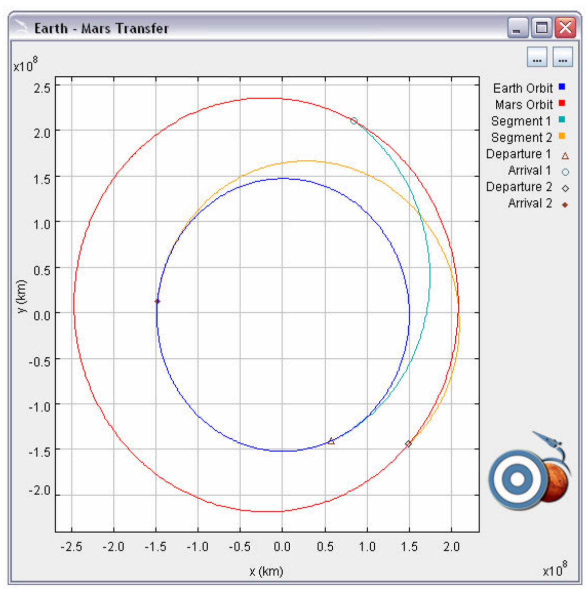

Figure 2: Sample Crew Mission Trajectory Plot

2. Mars Architecture Transfer Trajectories

The Mars exploration architecture is designed to allow four sortie conjunction-class missions between the 2030 and 2040. There are five transfer opportunities during the decade which are characterized by relatively low energy Earth-Mars transfer missions. The first opportunity near the end of 2030 will only consist of a cargo transfer. The next three opportunities will have both cargo and crew transfers. The final opportunity at the end of the decade will consist of only the last crew transfer.

In order to allow transfers in all desired opportunities, the appropriate transfer trajectory data needed to size the architecture must be determined. This transfer data includes the C3's, transfer times, and surface stay times encountered during each opportunity. Limits placed on these values bound the feasible departure and arrival dates for both the crew and cargo transfers for each opportunity. The maximum C3's and times encountered across all the desired launch dates were used to size the architecture. In this way it was guaranteed that both crew and cargo missions could be conducted during each required transfer opportunity throughout the decade.

For the cargo missions, the TMI stage was sized to provide an Earth departure C3 of 15.32 km^2/s^2. This represents the maximum Earth departure C3 encountered during any of the four cargo mission transfer opportunity launch windows (14.88 km^2/s^2), plus a 3% margin. Likewise the ISPS was sized to provide a Mars arrival C3 of 12.24 km^2/s^2, which represents the maximum Mars arrival C3 encountered during any of the four cargo mission transfer opportunity launch windows (11.88 km^2/s^2), plus a 3% margin. The maximum required C3's for the cargo transfers in each mission opportunity are shown in Figure 3. As can be seen, for every opportunity the required C3's are less than the maximum C3's used in the design of the cargo mission in-space stages.

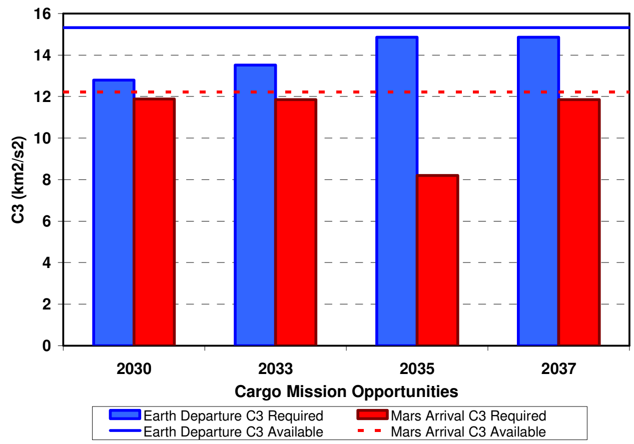

Figure 3: Cargo Mission Maximum Required C3 Values

For each opportunity a range of dates is available where the required C3's are less than the maximum design C3's. This range of dates represents the LEO departure window for the cargo mission in each opportunity. As can be seen in Figure 4, for the chosen design C3's, the cargo LEO departure windows vary in range from 6 to 60 days. Also in some opportunities (2030 and 2033) the arrival C3 constrains the launch window, while in others (2035 & 2037) the departure C3 determines the launch window length.

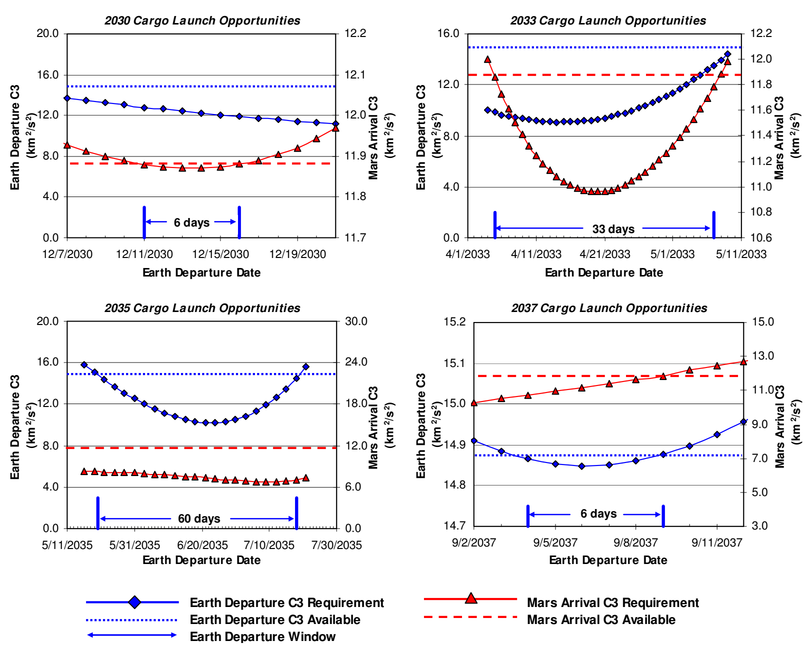

Figure 4: Cargo Mission Departure Windows

For the crew transfer mission, constraints were placed on the maximum allowable departure and arrival C3's (both Earth-Mars and Mars-Earth) and the maximum transit times and surface stay time. The maximum allowable Earth departure C3 is a function of the Ares V performance capability for both the surface habitat MEV and crew MEV launches. The arrival C3's determine the maximum entry velocity seen by the MEVs and the ERV upon atmospheric entry. For the crewed transfer opportunities the departure C3's were the dominate constraint compared to the arrival C3 values (Figure 5). The maximum outbound transit time, inbound transit time, and surface stay time were limited 355, 265, and 500 days, respectively. These values determine the maximum amount of required crew consumables in the two TransHabs and the surface habitat. The Ares V is able to provide a maximum Earth departure C3 of 19.97 km^2/s^2 for the crewed MEV launch and a maximum of 33.02 km^2/s^2 for the surface habitat MEV launch. The Mars arrival C3 limit used was 39.57 km^2/s^2 which corresponds to an approximate Mars entry velocity of 8.0 km/s. The Earth arrival C3 limit was 38.16 km^2/s^2 which gives an approximate Earth entry velocity of 12.7 km/s. This represents a 15% increase in entry velocity when compared to those experienced during a typical Apollo mission reentry.[10]

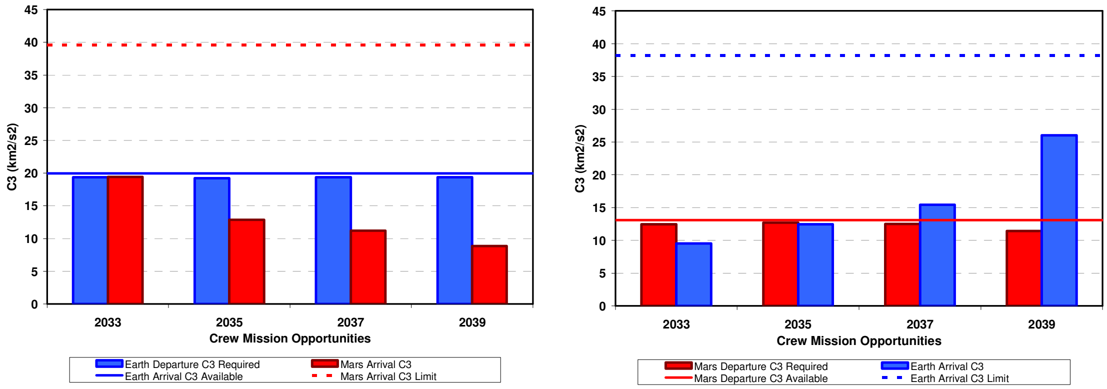

Figure 5: Crew Mission Maximum Required C3 Values

The C3 limits (after subtracting a 3% performance margin), time of flight, and surface stay time constraints determined for the MEV flights were used to develop Earth departure windows for each transfer opportunity. The launch windows for the crewed MEV flight range from 25 – 100 days depending on the transfer opportunity (Figure 6). The launch windows for the surface habitat flights are longer because the maximum C3 available from the Ares V is larger for this launch. Recall from the architecture description that the surface habitat MEV is launched by itself, while the crewed MEV is launched with the outbound TransHab. The reduced payload for the surface habitat MEV launch allows for the increased Ares V C3 capability.

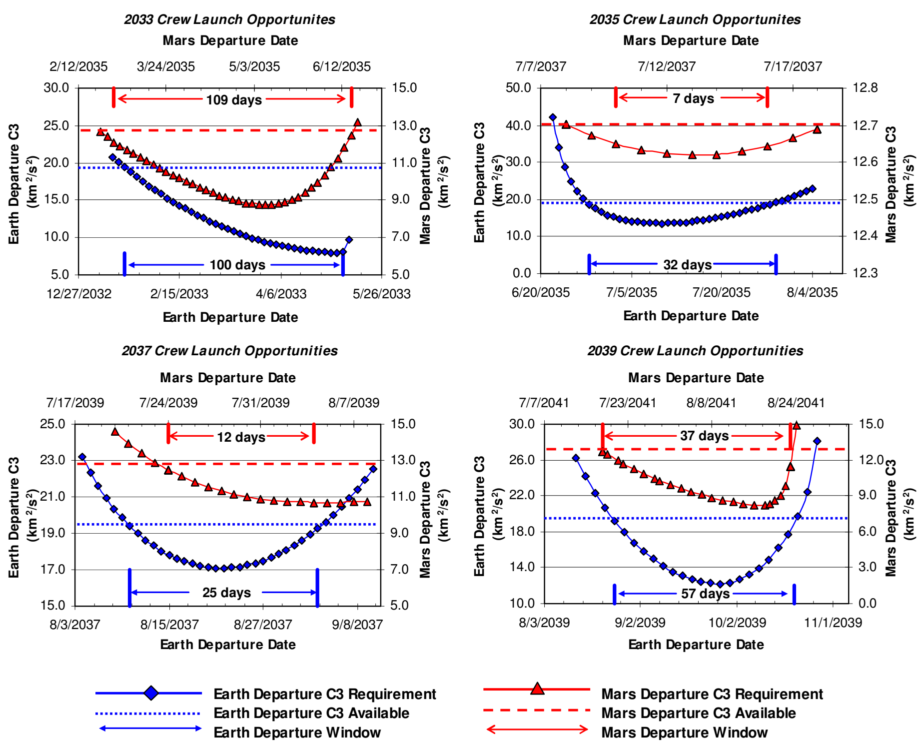

Figure 6: Crewed MEV Mission Departure Windows

IV. Architecture Element Descriptions

A. Earth-to-Orbit Launch Vehicles

A fundamental assumption of this architecture design is the use of fully-developed and operational Earth to orbit (ETO) launch assets circa 2030. To this end, it is proposed that Ares I and Ares V launch vehicles currently under development for the Project Constellation lunar exploration program be utilized for the candidate Mars architecture (Figure 7).

1. Ares-I Crew Launch Vehicle

Based on Space Shuttle heritage technology, the Ares I crew launch vehicle consists of a 5-segment reusable solid rocket booster (RSRB) first stage, a LOX/LH2 upper stage powered by a single J-2X engine, and a crew exploration vehicle. The CEV, in turn, is made up of three components: the crew module capsule, the service module, and the launch abort system. The CEV is intended to carry up to six astronauts to LEO, depending on the mission type, and will therefore have more than sufficient capability to meet the crew launch needs of this Mars architecture.

2. Ares-V Cargo Launch Vehicle

The Ares V heavy lift vehicle is the workhorse of each Mars exploration mission, with four launches required to send various architecture components to LEO and beyond. As with the Ares I, the Ares V incorporates heritage-derived systems from both the Space Shuttle and the Apollo Saturn rockets. The Ares V is comprised of a central LOX/LH2 core stage derived from the Shuttle external tank and powered by five RS-68 engines. Flanking the core stage are two 5-segment RSRBs, and atop the core is a LOX/LH2 EDS powered by a single J-2X engine.

Since the Ares V system is still undergoing design and development, it was necessary to select a particular design point from the literature to determine the available Ares V cargo ETO and C3 capabilities. Sufficient public data sources are available which give the Ares V LEO payload capability and allow for the simulation of the injection of the Ares V payload onto a Mars transfer trajectory.[11] The assumed Ares V LEO payload capability is ~130 t. Figure 8 shows the calculated Ares V C3 capability versus payload. These values where determined using a Program to Optimize Simulated Trajectories (POST) Ares V trajectory simulation.

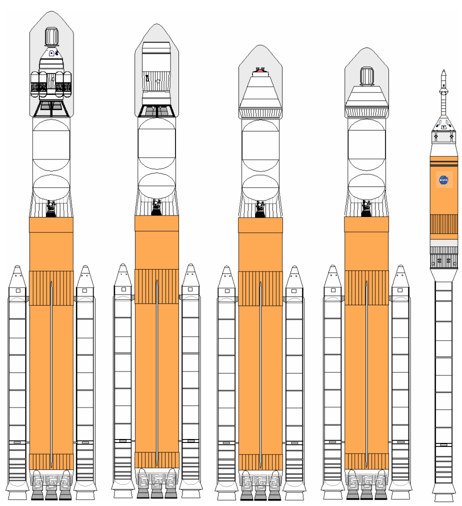

Figure 7: Ares I & V Launch Vehicles

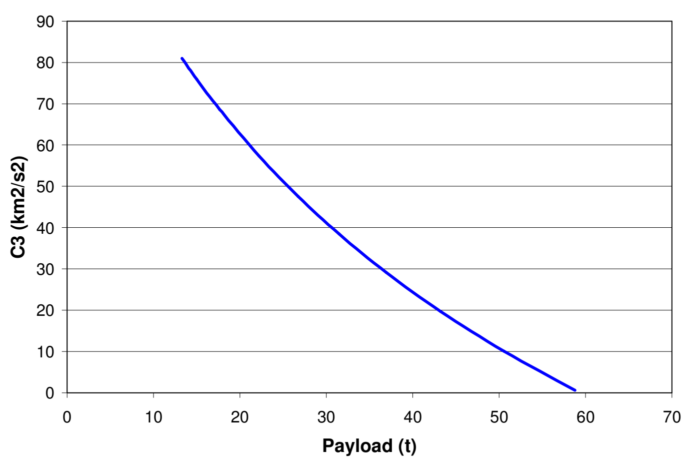

Figure 8: Ares V C3 Capability Versus Payload

B. Trans-Mars Injection Stage

The TMI stage places the ISPS, Return TransHab and ERV on a direct Mars transfer trajectory. Transit times for this transfer range from 170-402 days, depending upon the mission opportunity. The TMI stage sizing includes a 15% weight growth margin applied to all hardware items. Shown in Figure 9, The TMI stage is placed into LEO by a single Ares V launch.

The stage consists of two propellant tanks, an intertank adaptor, a payload adaptor, a thrust structure, four standard RL10B-2 engines, and power subsystems. The total stage, with engines but not including the payload stack, is approximately 17m in length and has a maximum diameter of 6.5m. During launch, the RL10B-2's are in their stowed positions with the ablative section of the nozzle retracted. The propellant tanks are made of 2219 Aluminum, with composites (Gr-Ep) used for the intertank, thrust structure, and payload adaptor to minimize weight. The total propellant loads for the fuel and oxidizer are 16.8 t and 98.3 t, respectively, providing the required engine operating mixture ratio (O/F) of 5.85. As previously noted for the architecture, no propellant boil-off is assumed to occur, but a weight estimate for the zero-boiloff hardware required to obtain this goal is included in the stage sizing.

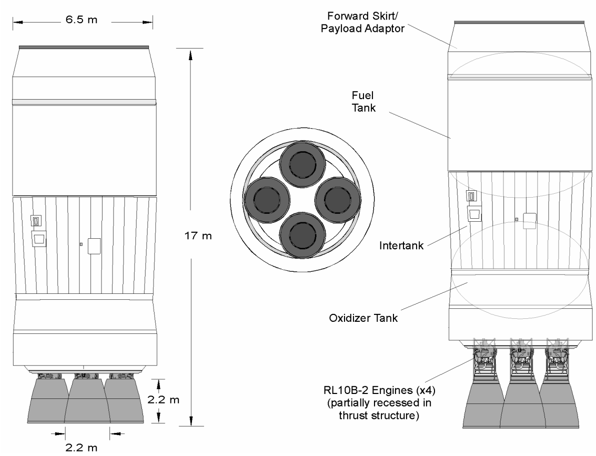

Figure 9: Trans-Mars Injection Stage

C. In-Space Propulsion Stage

The ISPS provides the MOI burn for the cargo leg of the mission and the TEI for the crewed return leg (Figure 10). The ISPS payload for MOI consists of the return TransHab and ERV. For TEI the payload includes those components as well and the crew and surface return payload. Again, a 15% weight growth margin is applied to all hardware items.

For the MOI propulsive burn, an independent set of propellant tanks, or drop-tanks, are used. These drop-tanks are attached circumferentially to the core stage and are jettisoned after completion of the MOI maneuver. Propellant feed lines transfer the fuel and oxidizer contained in these tanks to the engines on the core stage.

The ISPS core consists of two propellant tanks, an intertank adaptor, a payload adaptor, a thrust structure, three standard RL10B-2 engines, and power subsystems. The six (6) sets of cylindrical drop-tanks, each consist of a fuel and oxidizer tank along with an intertank adaptor. The total stage is approximately 12.8m in length (including deployed engines) and has a maximum diameter of 9.2m (with a core diameter of 5.5m). Prior to the TMI stage jettison, the RL10B-2 engines are in their stowed positions with the ablative section of the nozzle retracted. As with the TMI stage, the propellant tanks are made of 2219 Aluminum, with composites (Gr-Ep) used for the intertank, thrust structure, and payload adaptor to minimize weight. The total propellant load in the core stage tanks is 2.5 t and 14.8 t, for the fuel and oxidizers respectively. For the six drop-tanks, the total fuel and oxidizer propellant loads are 4.5 t and 26.2 t, respectively. This total propellant load provides the required engine operating mixture ratio (O/F) of 5.85. Again, as previously noted for the architecture, no propellant boil-off is assumed to occur, but a weight estimate for the zero-boiloff hardware required to obtain this goal is included in the stage sizing.

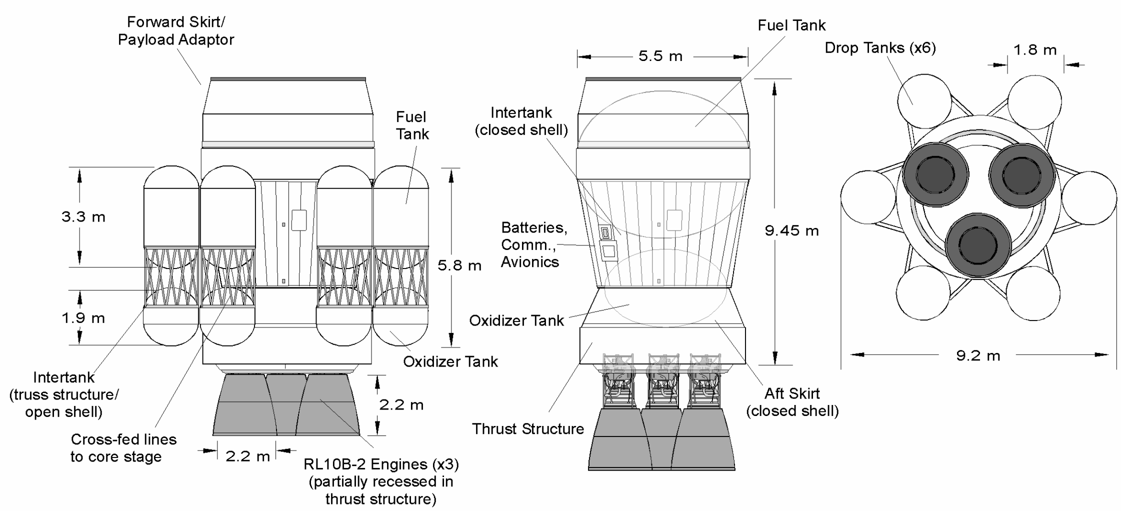

Figure 10: In-Space Propulsion Stage

D. In-Space Transfer Habitats

Separate inflatable transfer habitats are used to support the crew during the outbound and inbound Earth-Mars in-space transfers. The two TransHabs are identical except for the amount of crew consumables stored within each. The return TransHab is stocked with sufficient consumables to support the crew during the return leg of the mission, plus enough contingency supplies to sustain the crew while in LMO should a failure of the surface habitat require the crew to abandon the surface mission prematurely.

As seen in Figure 11, the outer walls of the TransHab are composed of an inflatable material. This material consists of Multi-Layer Insulation (MLI), micrometeorite protection, and redundant pressure bladders. Inflated, each TransHab provides approximately $60m^3$ of habitable volume. The tunnel down the rigid center of the habitat is 1m in diameter, bounded on either side by a circular hatch and docking mechanism. This tunnel allows the crew to move freely between the TransHab and the modules docked to it.

Power is provided to the TransHabs by four solar arrays.[12] There is sufficient redundancy in the solar arrays such that should one fail, the other three would be adequate to power the habitat. Thermal control inside the habitat is maintained by a two-fluid water/Freon active system. Water is pumped throughout the central core of the habitat and through coldplates attached to all of the major equipment. The water carries the heat to the top and bottom edges of the habitat, where it is transferred to the Freon loop via heat exchangers. The Freon is then pumped through radiators exposed to the vacuum of space, where the heat is rejected. The exterior of the habitat is coated with reflective paint and as mentioned previously lined with MLI. The MLI insulates the habitat while the coating reflects most of the sun's thermal energy. To ensure the safety of the crew, the major components of the power distribution and thermal control systems are redundant.

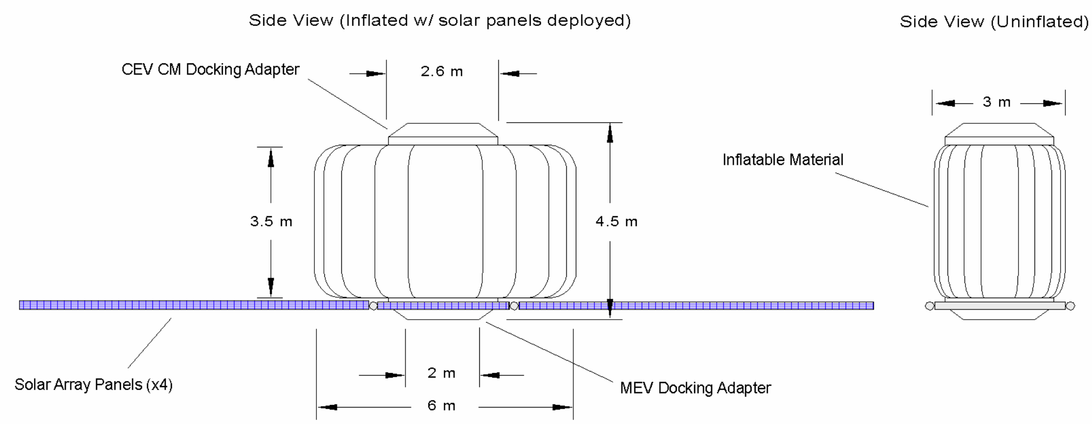

Figure 11: In-Space Transfer Habitat

Each TransHab has a closed water and oxygen life support system. Water is filtered using multifiltration and waste water is reclaimed through vapor compression distillation. Solid waste is vented into space. The CO2 generated by the crew is removed from the cabin atmosphere by a 4-bed molecular sieve. This CO2 is combined with tanked and generated H2 to produced CH4 and water in a sabatier reactor. The CH4 is vented into space, but the water is converted into O2 and H2 via electrolysis in an oxygen generation assembly. The H2 generated is fed back into the sabatier reactor, the O2 back into the cabin atmosphere. All of the major components of the ECLSS system are fully redundant.

The crew's food is stored dehydrated: it is hydrated before consumption, and the water consumed is recovered by the closed loop water system. The galley in the TransHabs includes a food warmer and sink. The habitats include aerobatic and resistive exercise equipment to combat muscle atrophy caused by the microgravity environment. Each TransHab is also stocked with ample medical supplies in case of an emergency. For hygiene, the habitats come equipped with a toilet and microgravity shower similar to those designed for the International Space Station. Both TransHabs contain a small water-walled radiation shelter to protect the crew from solar particle events.

The TransHabs were designed using HabSizer, a Microsoft Excel-based parametric sizing tool for crew habitats. The tool was developed by SEI to rapidly size habitation elements of human exploration architectures and perform trade studies on those elements. The primary inputs for HabSizer habitats are number of crew and mission duration, dimensions of rigid and inflatable components, and habitat location. From there, each subsystem is sized based on mass estimating relationships for its individual components.

The pressure vessel structure is sized based on the absolute pressure difference between the interior and exterior of the habitat. Internal support structure is added based on the habitat's interior surface area. Interfaces, such as doors and hatches, and flooring are included individually. Additional structures, including external supports and payload attachment points, are sized based on a percentage of the gross mass of the habitat.[13]

Mass and dimensions for active thermal control system components, including radiators, coldplates, pumps, valves, piping, and fluids, are determined based on the heat generated by the equipment in the habitat, and environmental heat transfer, including sunlight, based on the location of the habitat. The mass of the MLI and reflective paint is calculated using the exterior surface area of the habitat. [14]

The size of the power generation, distribution, and control systems are all based on the power level required within the habitat. The average and peak power required of each subsystem component is determined and a power budget is generated from the different operating modes of the habitat. The peak power required within the habitat can therefore be determined and used as the baseline power required of the power generation system. [14]

The individual components of the closed-loop ECLSS system, including the multifiltration and vapor compression distillation units for the water loop and the 4-bed molecular sieve, sabatier reactor, and oxygen generation assembly for the oxygen loop, are sized from the crew metabolic rates.[15] The rate of water, food, and oxygen consumption, solid waste, and carbon dioxide generation are determined for the number of crew based on published sources.[16] Equipment for atmospheric ventilation, temperature and humidity control, and dust and trace contaminant control is sized based on the habitable volume within the habitat. Water reclamation, waste removal, and fire detection and suppression equipment is also included. [14]

E. Mars Excursion Vehicle

The candidate Mars architecture includes two vehicles which enter the Martian atmosphere and land on the planet's surface. Each Mars Excursion Vehicle consists of a propulsive descent stage and a payload contained within an outer heatshield shell. The crewed MEV also includes an ascent stage and a pressurized rover, while the surface habitat MEV carries the fully-provisioned surface habitat complete with a nuclear surface power system (Figure 12).

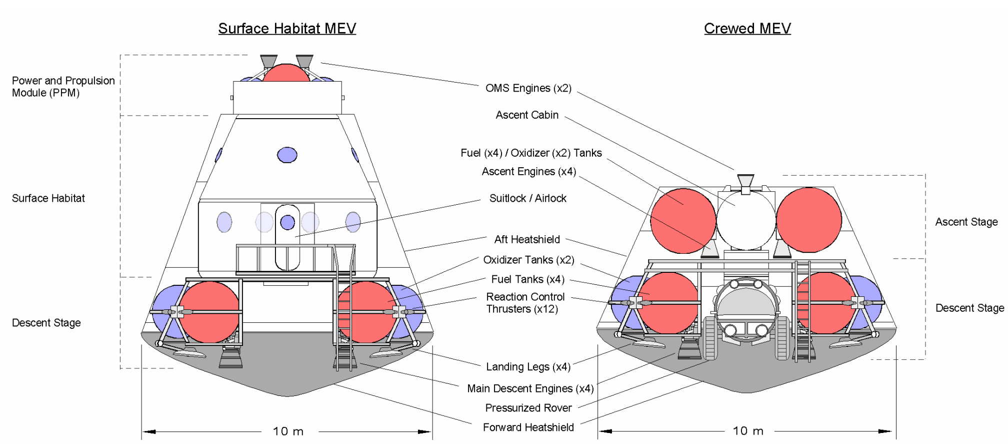

Figure 12: Crewed and Surface Habitat MEVs

1. Heatshield Sizing

In keeping with the historical design experience for Mars entry heatshields, the MEVs in this architecture incorporate a 70° conical forward shield with a spherical nose cap. A diameter of 10m was baselined in order to provide adequate aerocapture performance while maintaining compatibility with the launch vehicle shroud dimensions. The heatshield includes three component layers which are, from the outermost to the innermost, an ablative thermal protection material, a metallic honeycomb support structure, and an insulating blanket material. Selection and sizing of the layered materials of the heatshield were accomplished using Sentry[17], a conceptual level thermal protection system (TPS) sizing program developed by SpaceWorks Engineering, Inc. Sentry performs a one-dimensional (1-D) unsteady heat transfer analysis with convection, conduction, and radiative effects and an adiabatic backface condition. Both convective and radiative heat inputs were accounted for in this analysis. A trade study of several ablative material options led to the conclusion that Phenolic Impregnated Ceramic Ablator (PICA) offered the best combination of low technology risk and mass efficiency.

2. *Descent Stage Sizing*

Although the descent stages on the two MEVs are sized to accommodate a slightly different propellant load, there is a great deal of commonality between them. The stage is based on an open truss structure design which houses four common main rocket engines, propellant tanks, and subsystems. This layout also provides clearance for docking and carriage of the pressurized rover. A combination of LOX and LH2 is used to fuel four 54 kN-class (12 klb) main engines as the MEV stack descends the final 10 km altitude to the surface. These propellants are stored in six spherical tanks located around the circumference of the circular planform. During the Entry, Descent and Landing (EDL) mission segment, electrical power for the crewed MEV descent stage is provided by an Advanced Stirling Radioisotope Generator (ASRG) located on the ascent stage. Meanwhile, power for the surface habitat MEV descent stage during its EDL phase is stored onboard in a battery bank.

3. *Ascent Stage Sizing*

The ascent stage is designed to transport the crew and a small return payload from the surface of Mars to low Mars orbit. The crew is accommodated in a pressurized cylinder which provides a close-fitting environment akin to an aircraft cockpit. Two hatches located on the top and bottom of the cylinder provide access to the TransHab and pressurized rover, respectively. Four 33 kN-class (7.5 klb) main rocket engines, two 18 kN-class (4 klb) orbital maneuvering system (OMS) engines, and an array of reaction control thrusters comprise the propulsion system on the MEV ascent stage. It is necessary to include separate OMS engines oriented away from the forward heatshield due to the fact that the main engines are not exposed prior to final atmospheric entry. LOX and LH2 propellants are stored in six spherical tanks situated around the central crew cabin. Propellants for the OMS are cross-fed from the descent stage to perform post-aerocapture orbit circularization and entry initiation burns. Electrical power is supplied by an onboard ASRG. The ascent stage is equipped with supplies and consumables sufficient to support three crew members for up to seven days. This duration enables some contingency beyond the nominal time required for ascent and rendezvous.

4. *Surface Habitat MEV Power and Propulsion Module Sizing*

Unlike the crewed MEV, the surface habitat MEV is not equipped with an ascent stage, nor does it travel to Mars mated to a TransHab. In order to maintain the quiescent power level required by the surface habitat and to preserve the capability to perform post-aerocapture orbit circularization and entry initiation maneuvers, it is necessary to mate a power and propulsion module (PPM) to the Cargo MEV stack. The PPM is a simple stage comprised of a light structure, two 18 kN-class (4 klb) liquid rocket engines, propellant tanks, deployable solar arrays, and associated power management and distribution hardware.

5. *Trajectory Simulation: Entry, Descent, and Landing and Mars Ascent*

The EDL segment of the mission was simulated using POST-II. The EDL phase was considered to begin at the first interface of the MEV spacecraft with the Martian atmosphere. As indicated in the concept of operations for this architecture, two MEVs are deployed to Mars during each mission: a crewed MEV and a surface habitat MEV. These spacecraft perform a sequence of maneuvers upon arrival including, aerocapture into Mars orbit, aerobraking into low elliptical orbit, orbit circularization, final atmospheric entry, and propulsive descent to the surface.

The objective of the aerocapture maneuver is to use aerodynamic forces to slow the inbound MEV spacecraft from its arrival velocity of ~ 7 km/s to a desired Mars orbital velocity. At the periapsis point of this capture pass, the MEV dips into the upper atmosphere to an altitude of 61 km. Subsequent aerobraking is performed to further reduce the energy of the spacecraft's parking orbit.

Once the proper alignment of the orbital ground track and landing site is reached, the MEV initiates its final entry into the Martian atmosphere. The spacecraft decelerates aerodynamically to an altitude of 10 km, at which point the fore and aft heatshields are jettisoned and the liquid rocket engines on the descent stage are ignited. The terminal phase of the descent is then dominated by propulsive deceleration with some aerodynamic drag contribution. Touchdown occurs at a vertical velocity of 2 m/s.

When the end of the surface stay period is reached, the crew returns to the crewed MEV equipped with the ascent stage. The main engines on the ascent stage are ignited and the stage is boosted from the surface into a 100 by 400 km altitude orbit. A small circularization burn is performed at apoapsis in preparation for rendezvous with the orbiting ISPS, return TransHab, and ERV. Simulation of the ascent trajectory was also conducted using POST-II.

F. Mars Surface Habitat

The surface habitat supports the crew during their stay on the Martian surface. Access to the surface habitat is made possible via a hatch in the floor that connects to the pressurized rover. In addition, the surface habitat houses two suitlock airlocks to provide for local Extra-Vehicular Activity (EVA) to the surroundings. The surface habitat is a two level rigid pressure vessel that provides ~84 m^3 of habitable volume. The lower level of the habitat includes a galley, an airlock, a small science laboratory, a medical station and an area for hygiene with a sink and shower. The upper level is divided into three separate areas for personal crew quarters.

The surface habitat is powered by one of two ASRGs. Both ASRGs can individually provide the habitat with sufficient power for all the habitat operating modes. One ASRG is activated once the surface habitat is deployed on the surface; the second ASRG is kept in reserve. Should the first ASRG fail, the second can be brought online.

The surface habitat shares nearly identical thermal control and ECLSS systems with the TransHabs. As with the TransHabs, the surface habitat has a closed-loop water and oxygen life support system, with fully redundant equipment. Crew accommodations on the surface habitat are similar to those on the TransHab, but designed for the 0.3g Martian environment rather than microgravity. Because the crew will spend significant portions of time on EVA, no resistive exercise equipment is included. Furthermore, the Martian atmosphere provides sufficient protection from solar particle events, so a radiation shelter is not needed.[18]

Direct access to the Martian surface from the surface habitat is made possible by the suitlock airlock. In this airlock design, the rear entry EVA suits are docked to the sides of the interior of the airlock, allowing the crew to regularly access the Martian surface without fear of contaminating the internal habitat environment. Should the crew wish to bring scientific samples from the surface into the habitat, however, an equipment hatch also connects the interior of the airlock to the habitat interior. This hatch can also be used in an emergency to bring a damaged EVA suit or incapacitated crewmember into the habitat. The airlock has a large hatch opening on the exterior of the habitat; a platform and ladder outside the airlock provide access to Martian surface. Should the airlock fail, the suitlocks on the rover provides a back-up means of habitat entry.

A model of the surface habitat was generated in a similar fashion to those of the TransHabs using the HabSizer tool. The model for the ASRG power generation unit was produced in HabSizer based on literature references.[19] It was assumed that the EVA suits designed for this mission have a mass of 60 kg which is the Occupational Safety & Health Administration standard for carrying capacity of the average human in the Martian gravity environment.[14] The other subsystems: structures, thermal control, power distribution and management, avionics, life support, and crew accommodations, were sized in the same manner as those in the TransHabs. As with the TransHab, the major components of the power distribution and thermal control systems are redundant.

G. Pressurized Rover

The payload of the crewed MEV includes a four-wheeled pressurized rover capable of transporting all three astronauts. In the nominal mission concept of operations the crew boards the pressurized rover after landing, and drives the vehicle to the surface habitat. At the conclusion of the mission, the crew makes a return trip aboard the rover to the crewed MEV. The pressurized rover is powered by an ASRG and is designed for 10 days of operation per driving excursion. Multiple excursions in addition to the round trip between the crewed MEV and surface habitat are assumed on each mission. Three suitlocks are mounted on the rear of the rover to enable EVA and to provide a contingency ingress/egress capability in the event that the topside docking system is unable to mate with the surface habitat or ascent stage.

V. Architecture Results

The following sections are intended to illustrate key findings and results determined from the assessment of the candidate Mars architecture proposed above. Several different types of architecture-level metrics are presented. Overall architecture element masses and how they are combined for launch on available ETO launch vehicles are shown. Affordability metrics are also presented. These include element-level and overall architecture cost metrics such as Life Cycle Cost (LCC), Design Development Testing & Evaluation (DDT&E), and Theoretical First Unit (TFU) costs. Finally, reliability metrics such as Loss of Mission (LOM) and Loss of Crew (LOC) percentages will be provided.

A. Architecture Element Masses

Each architecture element was designed and sized using a combination of physics-based relationships and industry standard mass estimating relationships. Detailed sizing was conducted at the subsystem level for every component. Subsystem masses from complete 2-level weight breakdowns were then rolled-up to get architecture element total masses. The sizing models for each individual element, along with launch vehicle performance data, and in-space trajectory simulations were fully integrated into the MSAT design environment. MSAT allowed for rapid 'closure' of the architecture from a performance standpoint. Table 1 shows the inert and total mass for various elements of the proposed Mars architecture. For brevity, the full multi-level weight breakdown for each element is not shown.

Table 1: Architecture Element Inert and Total Masses

Architecture Element	*Inert Mass (t)*	*Total Mass (t)*	*Stage Inert Mass Fraction*
TMI Stage	10.53	123.19	0.09
In-Space Propulsion Stage	5.78	53.26	0.11
Crewed MEV Ascent Stage	3.94	14.43	0.27
Crewed MEV Descent Stage	4.21	9.91	0.43
Pressurized Rover	3.79	4.05	0.93
Surface Habitat	15.86	20.33	0.78
Surface Habitat MEV Descent Stage	4.43	8.38	0.53
Outbound TransHab	8.89	10.67	0.83
Return TransHab	8.79	10.92	0.80
Earth Return Vehicle	-	6.00	-

The architecture level inputs used in MSAT for the final design were chosen to provide adequate launch windows for all Earth-Mars mission opportunities between 2030 and 2040. Therefore, the in-space propulsion stages are sized to provide the maximum required delta-V needed across all opportunities, while the TransHabs and surface habitat are able to carry the maximum amount of crew consumables needed for the longest transit and surface stay times encountered. Furthermore, payload mass limits for each Ares V launch were monitored to assure architecture closure. For the cargo mission where both Ares V launches deliver their payload to LEO, a maximum payload constraint of 130 t was imposed. Figure 13 shows that both the TMI stage launch (Cargo ETO Launch 1) and the ISPS/TransHab/ERV launch (Cargo ETO Launch 2) are under this limit. While the TMI stage approaches the 130 t limit, the ISPS/TransHab/ERV stack is well below the 130 t maximum (~72 t total mass). Since this 72 t represents the payload of the TMI stage after LEO rendezvous, the 130 t limit on the Cargo ETO Launch 1 effectively limits the maximum weight of the ISPS/TransHab/ERV combination to 72 t or less. There may be Earth-Mars mission opportunities with less demanding departure and arrival C3's which may allow for a larger ISPS/TransHab/ERV stack mass. However, the Cargo ETO Launch 2 will always have excess LEO payload capability which could be used to place secondary unrelated payloads into LEO.

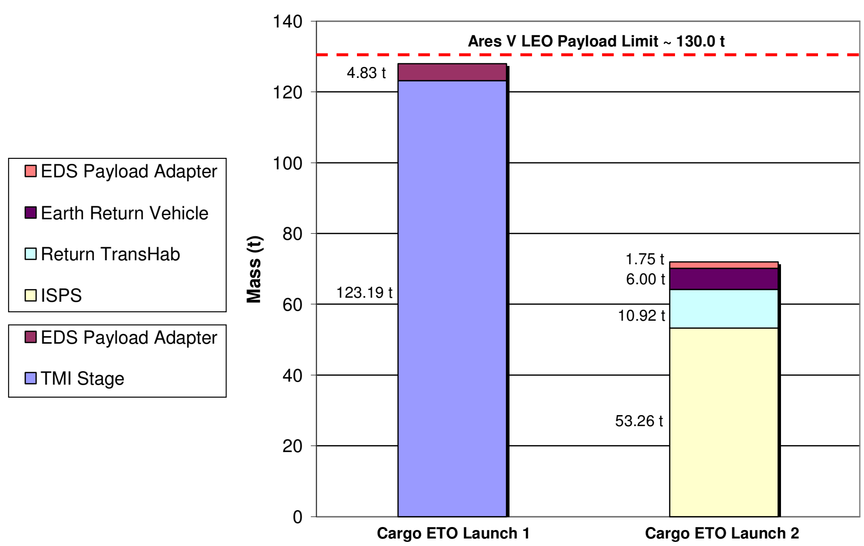

Figure 13: Ares V Cargo ETO Launches

The maximum payload for the two Ares V launches needed for each crew delivery is much lower than the maximum Ares V LEO payload. This is because these Ares V launches must place their payload on Mars transfer trajectories. The first Ares V launch places the surface habitat MEV on a direct Mars transfer trajectory while the second places the crewed MEV into LEO to rendezvous with the CEV launched on an Ares I. After the crew transfer from the CEV, the crewed MEV is placed on a Mars transfer trajectory using the propellant remaining in the Ares V EDS. The maximum weight limit for each MEV was 42.9 t. An Ares V is able to place this payload mass on an interplanetary trajectory with a C3 of ~19.39 km^2/s^2. This represents the maximum required Earth departure C3 for any crewed opportunity Earth departure window between 2033 and 2040. Figure 14 shows that both MEV launches are under the 42.9 t limit

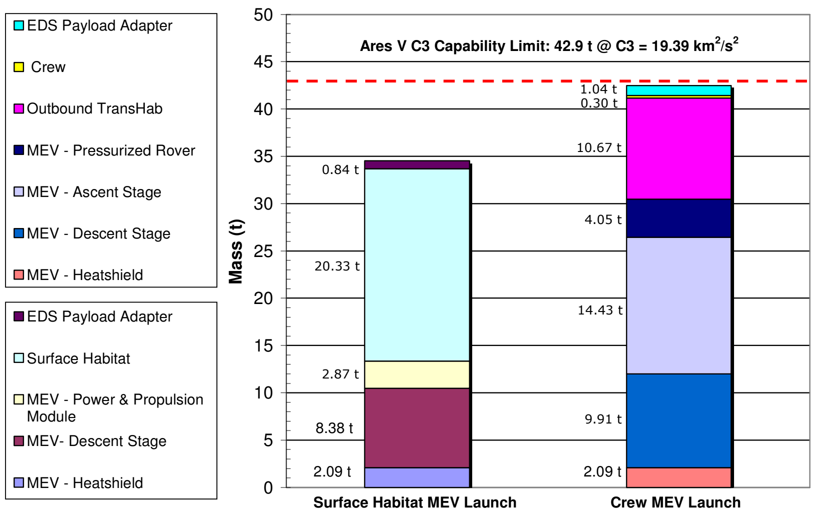

Figure 14: Ares V MEV Launches

B. Cost Analysis

A Life Cycle Cost (LCC) estimate has been developed for this notional human Mars exploration architecture. This estimate is developed using historical analogies, external cost estimates, and specific Cost Estimating Relationships (CERs). The ultimate objective of the LCC analysis is to determine the total program cost estimates, determine its main contributors, and develop the cost 'Sand Chart.' The Sand Chart is a budget chart that shows the elements of overall LCC over time. There has been an attempt to include all relevant Mars exploration costs (including test flights) into the generation of the LCC. The specific cost estimate developed here does not include any costs related to lunar exploration or International Space Station costs. This Mars exploration cost estimate includes the following elements for each campaign: MEV Ascent Stage, MEV Descent Stages (x2), ISPS, TMI Stage, Surface Habitat, TransHabs (x2), ERV, and the Pressurized Rover. Other costs included in this estimate include operations, facilities development, sustaining engineering, launch vehicles, technology development, robotic missions, overall NASA program integration, and program reserves. The estimate includes the development and acquisition cost of various hardware elements. Additionally, test flight costs are included. These test flights simulate the full up mission and include a vast majority of actual flight elements. The life cycle cost estimate is presented starting in the year 2025 and going up to the last year of the campaign (year 2040). Launch vehicles for this architecture, NASA's Ares I and Ares V are assumed to be ready for incorporation into the Mars human exploration architecture. Thus there is no development associated with the ETO launch vehicles, only a recurring cost for their acquisition for each mission. All cost estimates and figures are presented in constant year FY2007 dollars.

The cost estimate presented here is not developed in isolation. As of mid-2007, NASA's exploration plans include a human lunar return by the end of the second decade of the twentieth century. This cost estimate assumes that NASA will embark on such a lunar exploration campaign. For this Mars exploration campaign there is assumed to be some leveraging of development cost from the lunar campaign (particularly for elements such as the habitats). There is also some development cost reductions for the TransHabs based upon current mission and future plans from commercial companies such as Bigelow Aerospace. This is an assumption by the authors.

Figure 15 shows a notional illustration of the human Martian exploration campaign. For this assessment the first launch of the crew for a Martian landing is planned for fiscal year 2033, with subsequent missions every two years from then forward until fiscal year 2040, for a total of 4 missions over this 7 year period. The number of missions is assumed to be steady state with the same number of crew and cargo sent each opportunity. The first mission set of launches will be the cargo leg (with return TransHab and ERV). Approximately two calendar years later, the crew is launched to LEO on an Ares I, and rendezvous with the outbound TransHab and MEV that were delivered by an Ares V. Thus, in some years of the campaign, there will be several launches for different Mars opportunities.

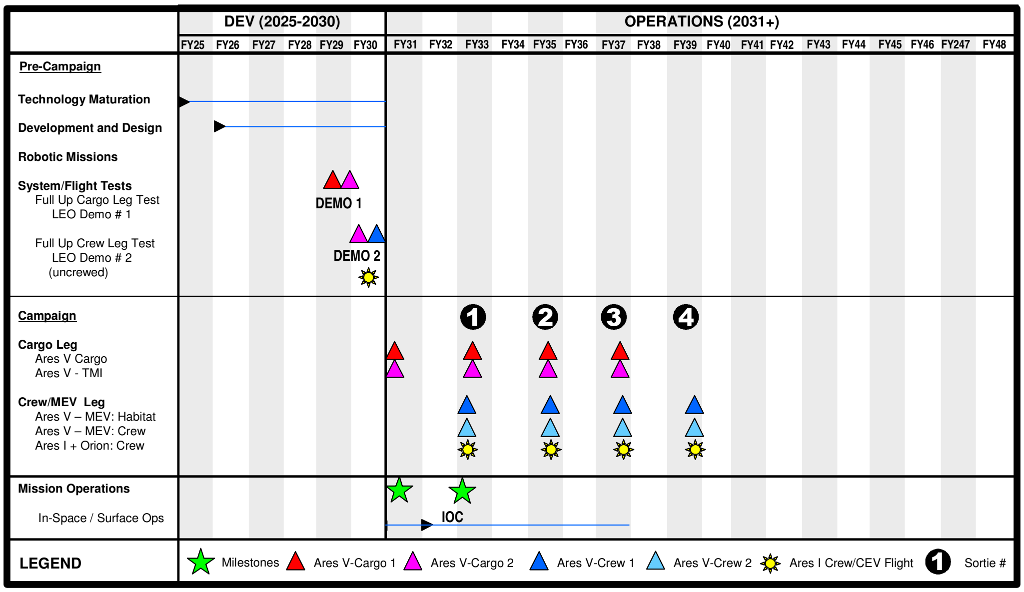

Figure 15: Notional Development Schedule and Operations Campaign

The deterministic total LCC (as seen in Table 2) from 2025 until 2040 for the human Mars exploration architecture is estimated to be US $96.82B in FY2007 dollars. This estimate is based upon the multiple assumptions described in previous sections including costs for pre-Phase A development (2-5% of Design and Development cost), hardware development/acquisition, flight tests, and sustaining engineering. This estimate is relatively complete in that it accounts for the all the major components that would be required in the Exploration Systems Mission Directorate (ESMD) enterprise at NASA related to exploration.

Table 2: Life Cycle Cost Estimate for Human Mars Exploration Architecture (2025-2040)

Cost Item	*LCC [FY2007]*	*% of Total LCC (Undiscounted)*
Trans-Mars Injection Stage	$4,630 M	4.8%
In-Space Propulsion Stage	$3,930 M	4.1%
Crewed MEV Ascent Stage	$7,580 M	7.8%
Crewed MEV Descent Stage	$4,450 M	4.6%
Pressurized Rover	$2,970 M	3.1%
Surface Habitat MEV Propulsion Module	$510 M	0.5%
Surface Habitat MEV Descent Stage	$4,220 M	4.4%
Surface Habitat	$9,840 M	10.2%
TransHabs	$6,930 M	7.2%
Earth Return Vehicle	$510 M	0.5%
Operations (Mission, Ground, EVA)	$9,430 M	9.7%
Facilities: Launch	$450 M	0.5%
Facilities: Mission Operations	$2,450 M	2.5%
Surface Systems	$1,130 M	1.2%
Launch Vehicles	$14,130 M	14.6%
Technology Development	*$1,000 M*	*1.0%*
Robotic Missions	*$1,000 M*	*1.0%*
Program Integration (Govt.)	$9,020 M	9.3%
Reserves	$12,630 M	13.0%
Total	**$96,810 M**	**100.0%**

As seen in Figure 16, a large investment is required during the initial years in the program for development of the hardware systems as well as the full-scale flight tests of all elements in 2029 and 2030. A notional budget line is also shown in Figure 16. This is the budget amount that NASA is projected to spend on human spaceflight and lunar exploration activities in FY2007 (space operations and exploration capabilities).[20] The notional Mars exploration architecture exceeds this limit in the years just prior to first operational launch. An optimized acquisition and development schedule may be able to determine a funding strategy that is under this budget cap.

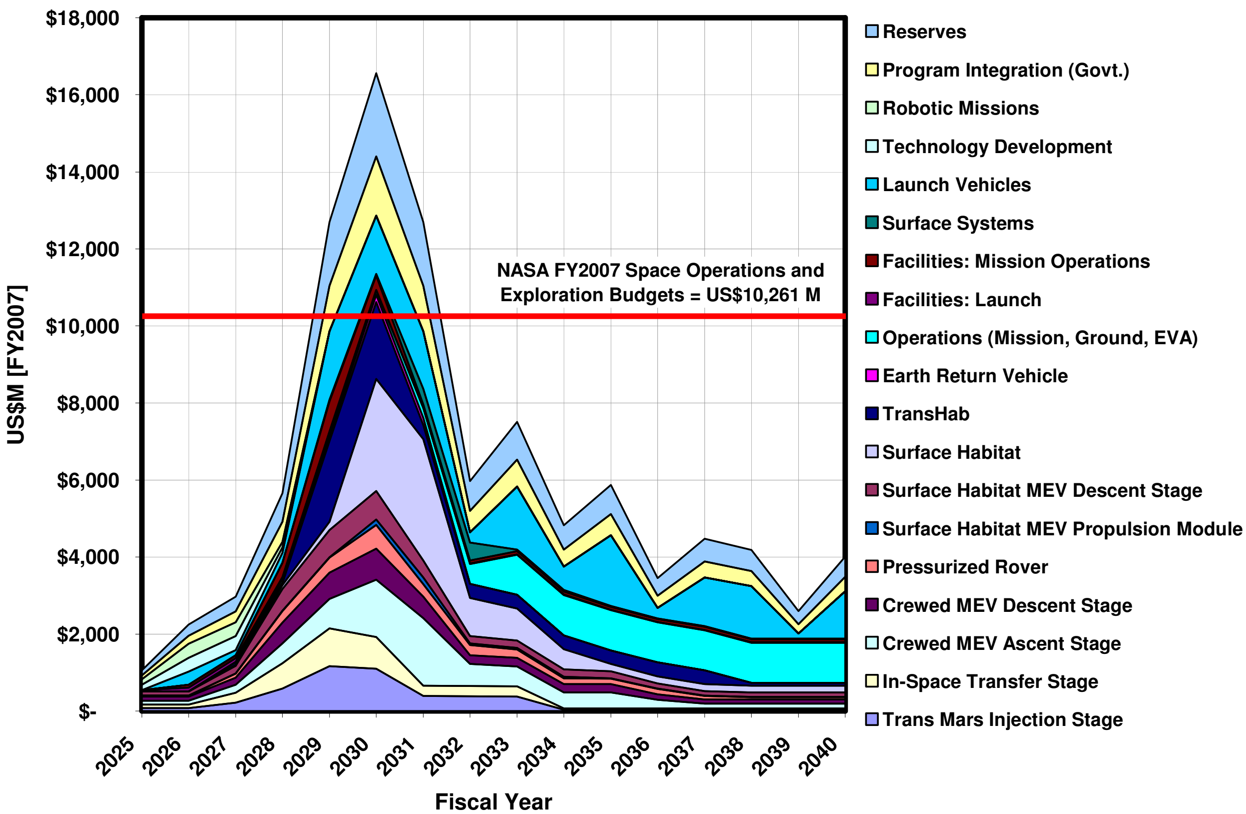

Figure 16: Annual Costs ('Sand Chart') for Human Mars Exploration Architecture (2025-2040)

C. Reliability Analysis

Results of the reliability analysis for the candidate architecture yield a mean loss of mission probability of 38.6 percent and a mean loss of crew probability of 11.5 percent. Stated alternatively, LOM is expected to occur once every 2.6 missions and LOC is expected to occur once every 8.5 missions. The 90th percentile values for these end states are 39.6 percent for LOM and 12.4 percent for LOC. A conservative definition for LOM was adopted for this analysis, whereby the failure of any element during any mission phase was considered LOM. Under this philosophy, no distinction is made for when a failure occurs. For example, a failure which causes the crew to leave the surface on day 450 of a 500 day surface stay results in a LOM equivalent to a failure that occurs on day one. Therefore, all failures contribute to the LOM calculation even though some failures may not necessarily result in the complete loss of the mission. Figure 17 shows the resulting probability distributions for LOM and LOC.

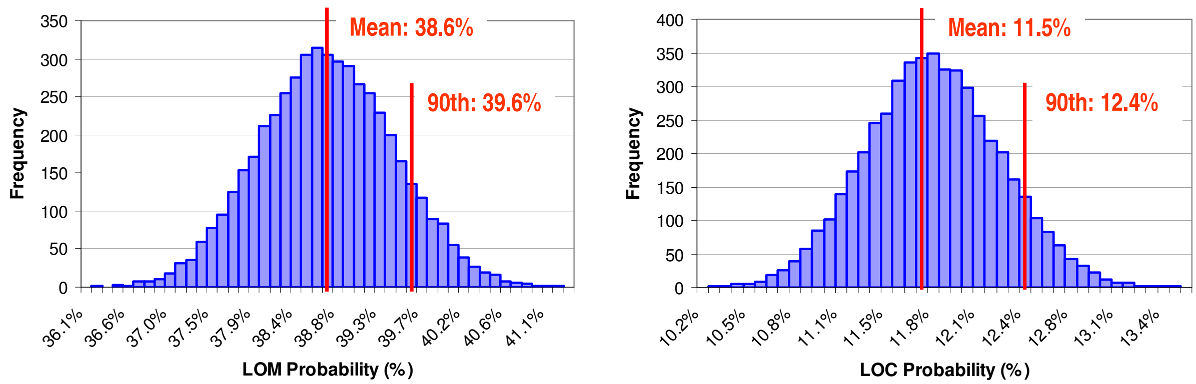

Figure 17: Loss of Mission and Loss of Crew Histograms (20,000 Monte Carlo Trials)

The relative contribution of each architecture element to these end states is shown in Figure 18. The high risk of catastrophic failure during Mars entry and long duration operation of the transit and surface habitats are the chief contributors to LOC. Recovery from a failure during the Mars entry mission phase also has a low probability due to few options for failure resolution over a relatively short period of time. Mars entry is also a chief contributor to LOM, but second to the Ares V launches. Each Ares V launch has over a 3 percent probability of failure and 4 launches are required, so the result is a significant total contribution to LOM by the Ares V. The crew is launched aboard an Ares I as opposed to an Ares V and thus only a small portion of Ares V failure probability (TMI by the EDS) has an implication for LOC. The TransHabs and surface habitat are also fairly significant contributors to LOM due to the lengthy operation time of these elements.

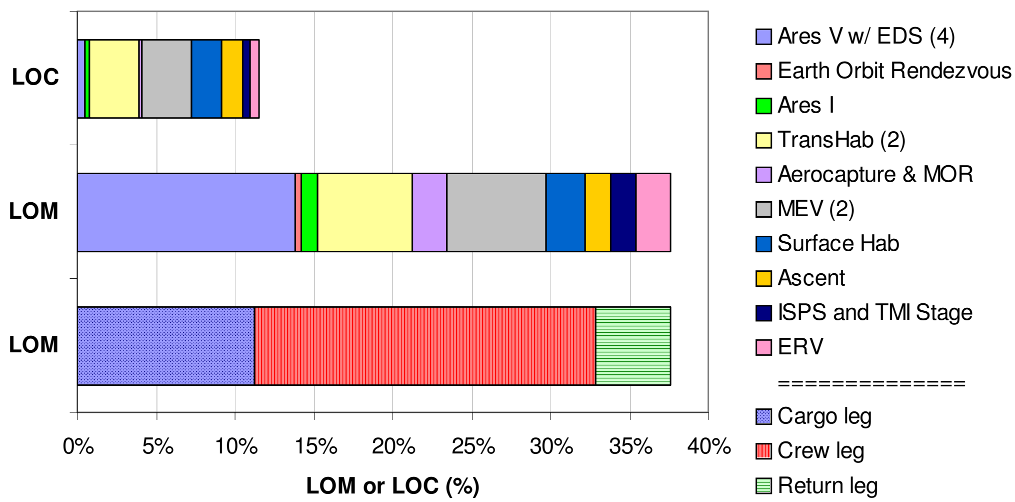

Figure 18: Contribution by Mission Elements to LOM and LOC

Reliability analysis was performed using a NASA standard fault tree and event sequence diagram approach as described in the NASA Probabilistic Risk Assessment Procedures Guide.[21] Fault tree basic events were predominantly at the subsystem level, and continuous operating time reliability functions ($R=1-e^{\lambda t}$) were applied for subsystems with significant operating times. LOC probability was calculated by applying a likelihood of failure resolution to each major loss of mission event in the event sequence diagrams. Subsystem redundancy and engine-out capability was modeled through the use of logical "AND" and "N of M" gates where appropriate. Data sources for subsystem failure rate values include NASA's Exploration Systems Architecture Study, commercial company publications, SEI internal models, and technical papers of the AIAA and other professional organizations. Probabilistic results were generated from 20,000 Monte Carlo trials with symmetric triangular distributions of plus or minus 20 percent on the input variables.

VI. Conclusions

While NASA's current focus is on Ares I / CEV programs and lunar exploration initiatives, it is important even at this early stage of development to see how resources produced by these programs translate to future missions. This paper has outlined a feasible Mars exploration architecture that is predicated on using multiple elements and subsystems that are expected to be developed and matured during NASA's lunar exploration campaign. The Mars architecture outlined in this study uses traditional LOX/LH2 propulsion systems along with Ares I and Ares V launch vehicles. The architecture was sized, using SpaceWorks Engineering, Inc.'s Mission Scenario Analysis Tool, to provide four sortie-style Mars exploration missions between 2030 and 2040. Detailed architecture element sizing was conducted along with a full life cycle cost analysis and mission reliability assessment. A limited set of new technologies is required for the Mars architecture outlined in this paper. These include cryocoolers or other hardware to eliminate cryogenic propellant boiloff in large LOX/LH2 propulsion stages, aerocapture and EDL systems for high mass entry vehicles, ASRG power systems, and inflatable transfer habitats.

References

[1] St. Germain, B., Charania, A., Olds, J. R., "A Stochastic Process for Prioritizing Lunar Exploration Technologies," AIAA-2005-6607, Space 2005, Long Beach, California, August 30 - September 1, 2005.

[2] Von Braun, W., *The Mars Project*, University of Illinois Press, Illinois, 1952, reprint 1991.

[3] Hoffman, S. J., and Kaplan, D. I. (ed.), "Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team," NASA Special Publication 6107, 1997.

[4] Drake, B. U. (ed.), "Reference Mission Version 3.0 – Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team," NASA Special Publication 6107-ADD EX13-98-036, 1998.

[5] Portree, D. S. F., "Humans to Mars: Fifty Years of Mission Planning, 1950 - 2000," NASA Monographs in Aerospace history Series, Number 21, 2001.

[6] Thomas, B., Vaughan, D., Woodcock, G., Drake, B., Johnson, L., and Griffin, B., "A Comparison of Transportation Systems for Human Missions to Mars," AIAA-2004-3834, 2004.

[7] Connolly, J. F., "Constellation Program Overview," *NASA Constellation Program* [online presentation charts], 2007 URL: http://www.nasa.gov/mission_pages/constellation/news/index.html [cited 14 August 2007].

[8] de la Fuente, H., Raboin, J. L., Spexarth, G. R., Valle, G. D., "TransHab: NASA's Large-Scale Inflatable Spacecraft," AIAA-2000-1822, 2000.

[9] Covault, C., "Bigelow's Gamble," *Aviation Week and Space Technology*, Vol. 161, No. 12, 27 Sep. 2004, p. 54.

[10] "National Aeronautics and Space Administration: Apollo 14 Mission Report", MSC-04112. 1971.

[11] Dumbacher, D. L., "A New Heavy-Lift Capability for Space Exploration: NASA's Ares V Cargo Launch Vehicle," AIAA-Space 2006, San Jose, CA, Sept. 19-21, 2006. [online presentation charts], URL: http://ntrs.nasa.gov/ [cited Aug. 14, 2007]

[12] "Space Solar Panels", Spectrolab Photovoltaic Products, [online data sheet], URL: http://spectrolab.com/DataSheets/Panel/panels.pdf [cited Aug. 1, 2007]

[13] Sarafin, T. P. and Larson, W. J., *Spacecraft Structures and Mechanisms*, Space Technology Series, Microcosm, Inc., El Segundo, CA, and Kluwer Academic Publishers, Dordrecht, The Netherlands, 1995, Chap. 15.

[14] Larson, W. J. and Pranke, L. K., *Human Spaceflight Mission Analysis and Design*, Space Technology Series, The McGraw-Hill Companies, Inc., 1999, Chaps. 16, 17, 18, 20, 22.

[15] Eckart, P., *Spacecraft Life Support and Biospherics*, Space Technology Series, Microcosm, Inc., El Segundo, CA, and Kluwer Academic Publishers, Dordrecht, The Netherlands, 1996, Chap. V.

[16] Wieland, P. O., "Designing for Human Presence in Space: An Introduction to Environmental Control and Life Support Systems", NASA RP-1324, 1994.

[17] Bradford, J. E., Olds, J. R., "Thermal Protection System Sizing and Selection for RLVs Using the Sentry Code," AIAA-2006-4605, 42nd AIAA/ASME/SAE/ASEE Joint Propulsion Conference and Exhibit, Sacramento, California, July 9-12, 2006.

[18] Simonsen, L. C. and Nealy, J. E., "Mars Surface Radiation Exposure for Solar Maximum Conditions and 1989 Solar Proton Events", NASA TP-3300, 1993.

[19] Chan, J., Wood, J. G., Schreiber, J. G., "Development of Advanced Stirling Radioisotope Generator for Space Exploration", NASA TM-2007-214805. 2007.

[20] FY 2008 Budget Request for FY 2007 President's Request, URL: http://www.nasa.gov/pdf/168653main_NASA_FY08_Budget_Summary.pdf [cited August 20, 2007]

[21] Stamatelatos, M., "Probabalistic Risk Assessment Procedures Guide for NASA Managers and Practitioners," NASA Office of Safety and Mission Assurance, NASA Headquarters, Washington, D.C., Version 1.1, Aug. 2002.