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Austere Human Missions to Mars

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The Design Reference Architecture 5 (DRA 5) is the most recent concept developed by NASA to send humans to Mars in the 2030 time frame using Constellation Program elements. DRA 5 is optimized to meet a specific set of requirements that would provide for a robust exploration program to deliver a new six-person crew at each biennial Mars opportunity and provide for power and infrastructure to maintain a highly capable continuing human presence on Mars. This paper examines an alternate architecture that is scaled back from DRA 5 and might offer lower development cost, lower flight cost, and lower development risk. It is recognized that a mission set using this approach would not meet all the current Constellation Mars mission requirements; however, this "austere" architecture may represent a minimum mission set that would be acceptable from a science and exploration standpoint. The austere approach is driven by a philosophy of minimizing high risk or high cost technology development and maximizing development and production commonality in order to achieve a program that could be sustained in a flat-funded budget environment. Key features that would enable a lower technology implementation are as follows: using a blunt-body entry vehicle having no deployable decelerators, utilizing aerobraking rather than aerocapture for placing the crewed element into low Mars orbit, avoiding the use of liquid hydrogen with its low temperature and large volume issues, using standard bipropellant propulsion for the landers and ascent vehicle, and using radioisotope surface power systems rather than a nuclear reactor or large area deployable solar arrays. Flat funding within the expected NASA budget for a sustained program could be facilitated by alternating cargo and crew launches for the biennial Mars opportunities. This would result in two assembled vehicles leaving Earth orbit for Mars per Mars opportunity. The first opportunity would send two cargo landers to the Mars surface to preposition a habitat, supplies, and exploration equipment. The next opportunity, two years later, would send to Mars orbit 1) a lander with a Mars Ascent Vehicle (MAV) and 2) a crewed Mars Transit Habitat with an Orion CEV for Earth return. The following opportunity, two years after the first crew, would go back to cargo-only launches. This alternation of cargo and crew opportunities results in a sustainable launch rate of six Ares V launches every two years. It is notable that four of the six launches per Mars opportunity are identical, build-to-print, Tran-Mars Injection stages. This type of production rate could lend itself well to a COTS-type service provider, and would make it feasible to have a live spare in place in the event of a single launch failure.

Nomenclature

CCM	=	Contingency Consumables Module
CEV	=	Crew Exploration Vehicle (Orion)

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COTS	=	Commercial Orbital Transportation Services
DAV	=	Descent/Ascent Vehicle
DIPS	=	Dynamic Isotope Power System
DRA	=	Design Reference Architecture
DRM	=	Design Reference Mission
EDL	=	Entry, Descent, and Landing
EDS	=	Earth Departure Stage
EVA	=	Extra Vehicular Activity
I_{SP}	=	Specific Impulse
ISS	=	International Space Station
ISRU	=	In-situ Resource Utilization
JPL	=	Jet Propulsion Laboratory
L2	=	Earth Lagrangian point 2
LEO	=	Low Earth Orbit
LCH_4	=	Liquid Methane
LH_2	=	Liquid Hydrogen
LOX	=	Liquid Oxygen
MAV	=	Mars Ascent Vehicle
MAWG	=	Mars Architecture Working Group
MOI	=	Mars Orbit Insertion
MSL	=	Mars Science Laboratory
NTR	=	Nuclear Thermal Rocket
Pu^{238}	=	Plutonium 238
SCRAM	=	Safety Control Rod Axe Man (jargon for emergency reactor shutdown)
SPLM	=	Surface Power and Logistics Module
SRP	=	Supersonic Retropropulsion
SurfHab	=	Mars Surface Habitat
T	=	Metric ton (1,000 kg)
TEI	=	Trans-Earth Injection
TMI	=	Trans-Mars Injection
TOP	=	Trajectory Optimization Program
TransHab	=	Mars Transit Habitat

I. Introduction

THE Design Reference Architecture 5 (DRA 5) [1] is the most recent concept developed by NASA to send humans to Mars in the 2030 time frame using Constellation Program elements. It was developed by a multi-center NASA design team, with most of the work performed in 2007. DRA 5 is optimized to meet a specific set of requirements that would provide for a robust exploration program to deliver a new six-person crew at each biennial Mars opportunity and provide for power and infrastructure to maintain a highly capable continuing human presence on Mars. DRA 5 was intended to be a reference point that was neither a minimum mission nor an overly ambitious one. This paper examines an alternate approach more toward the minimum end of the scale. Most of the elements of this paper were taken from the DRA 5 study. The DRA 5 study Addendum has a table of "Example Contingencies, Fallbacks, and Descope Options"[2], and many of them were exercised in the "austere" architecture described in this paper as an option for the human exploration of Mars. The impetus behind the austere architecture is to offer an approach that might have lower development cost, lower flight cost, and lower development risk. It is recognized that a mission set using this approach would not meet all the requirements assumed for DRA 5; however, it may represent a mission set that would be acceptable from a science and exploration standpoint.

II. Goals and Requirements

THE austere architecture has a goal of meeting the basic science requirements of DRA 5, but with a reduced crew size and reduced frequency. A crew of four would be landed on Mars every four years, based on launching a crew at every other biennial Mars opportunity.

The most important programmatic requirement assumed is to be able to implement the development phase and a sustaining flight phase with an annual budget that would be flat-funded, although adjusted for inflation to maintain the same buying power from year to year. The goal would be to have this funded at a rate not higher than is currently in the budget for human space flight (~$8 B). A secondary goal would be for the entire development phase and first human mission to not cost more than what was spent for the International Space Station (~$100 B). Another goal is to implement the program on the shortest schedule possible. This would be to reduce total cost and to maintain interest in the program over the lifetimes of the taxpayers funding it.

Having the lowest possible risk in both development (programmatic risk) and in flight (mission risk) is extremely important. This, combined with the previously mentioned programmatic requirements, leads to the derived policy of taking a very conservative technical approach, minimizing high risk or high cost technology development. This would translate to avoiding development of any new technology if it is not absolutely needed. Reducing technical risk and cost risk was given a higher priority than reducing mass or offering higher performance.

A final major goal would be to maximize development commonality and production commonality. For the austere architecture, this means a common basic design for both crewed and cargo landers and a common Earth Departure Stage (EDS) that would be utilized for all launches to Mars. This minimizes development and flight testing and maximizes production efficiency for the sustained program.

III. Conceptual Architecture

THE Mars Architecture Working Group (MAWG) that developed DRA 5 assessed a comprehensive trade space that was wide enough to consider all reasonable options, building from previous studies, but limited enough to enable a meaningful evaluation. Fig. 1 shows the austere architecture mapped onto the MAWG trade space, following the thick red lines. The DRA 5 trade tree branches are indicated with the number 14 markers (see legend). Different branches are indicated for the DRA 5 cargo and crew elements. For the austere architecture, using the conjunction-class, long-stay mission opportunities is the same approach taken by DRA 5, for the reasons of increasing performance, reducing crew time in zero g, and avoiding a potentially risky Venus flyby. With regard to Mars Orbit Insertion (MOI), DRA 5 uses aerocapture for the landers and propulsive MOI for the crewed Transit Habitat (TransHab). The same approach is taken for the austere architecture, but with some significant variations.

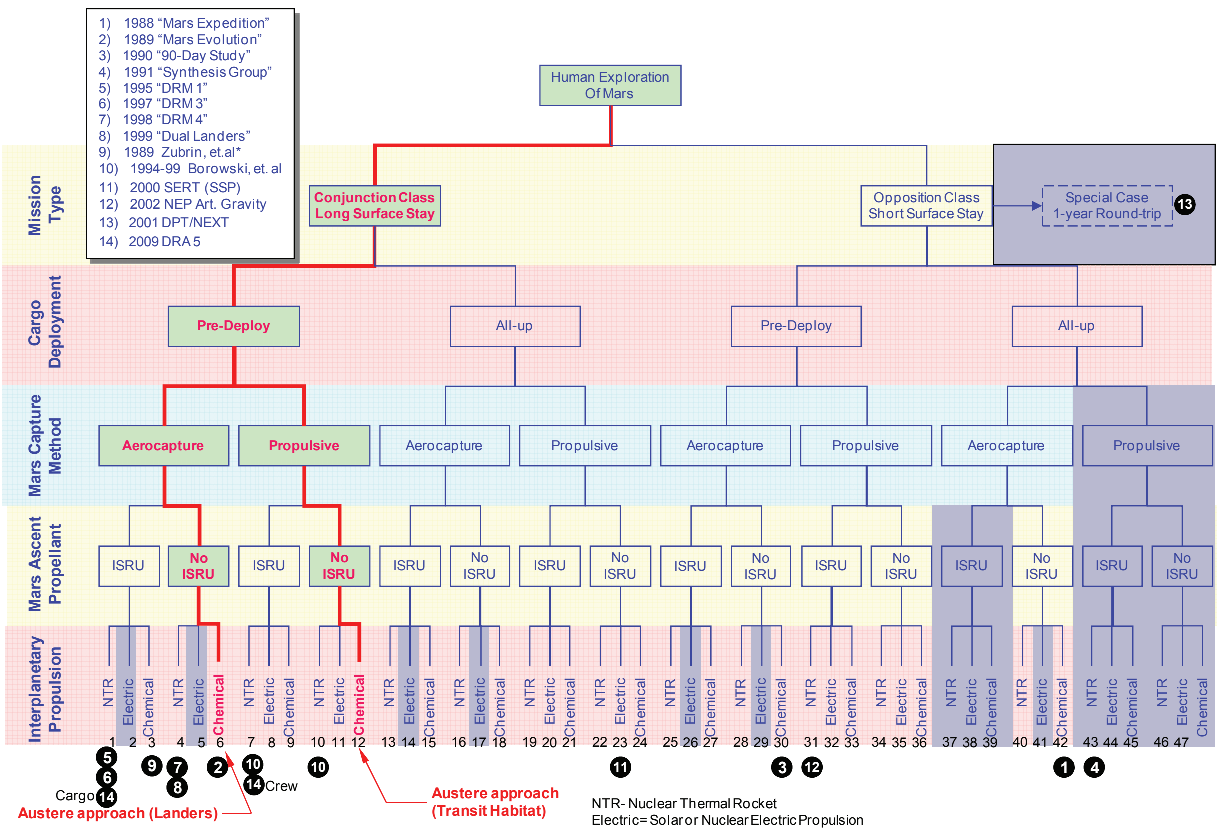

Figure 1. Trade tree for Mars architectures.

In order to significantly reduce the chemical propulsive ΔV for MOI, the TransHab would be inserted into a high elliptical orbit rather than a circular one. Then aerobraking would be used to reduce the orbit to the final desired low circular orbit. This is an effective approach that has been demonstrated successfully several times with robotic Mars missions. Since the stay time at Mars is quite long for the conjunction class missions, this was viewed as being a good trade-off, and it would reduce the consumables required for the surface mission since the surface stay time would be on the order of two months shorter. Since the Descent/Ascent Vehicle (DAV) must rendezvous and dock with the TransHab for crew transfer, the DAV would be aerocaptured into the same high elliptical orbit.

The cargo landers have no need to ever be in Mars orbit, and it is assumed that their landing site would be known prior to launch; therefore, they could use direct entry to land on Mars. This is an approach that has been well demonstrated in robotic missions, and the landing accuracy could be at least as good as with a descent from Mars orbit. This would result in a better mass performance for the cargo landers since they require no design features to support aerocapture or survival in Mars orbit.

The austere approach differs from DRA 5 with regard to propulsion technology, utilizing chemical propulsion rather than Nuclear Thermal Rocket (NTR) technology. While NTR offers significant improvements in Specific Impulse (I_{SP}), this would be offset by the greater dry mass required for the fission reactor. Although the net performance would be greater for NTR, the development cost and development risk (both technical and schedule) associated with NTR technology was the most important factor for not including it in the austere architecture. The in-flight risk would also be greater for NTR.

Liquid hydrogen propellant has been avoided for any portion of the mission after Trans-Mars Insertion (TMI). While the extreme low temperature maintenance requirement could be accommodated for a limited time in Low Earth Orbit (LEO) for the EDS, maintaining such low temperatures for a long Mars cruise was avoided to reduce both development risk and in-flight risk. Additionally, the larger tank volume required for liquid hydrogen propellant would be difficult to accommodate in configuration layouts that could fit within the Ares V payload fairing and within the mold line of the Mars lander entry vehicles. While the greater I_{SP} of liquid hydrogen propulsion is desirable, lower technology propellant storage approaches were taken in the austere architecture to reduce risk. The in-flight liquid hydrogen storage risk for the EDS occurs prior to commitment to TMI, so any failure would not result in loss of crew.

In-situ Resource Utilization (ISRU) for propellant production was not utilized in this architecture in order to reduce development cost and risk. The crewed lander carries all the propellant required for descent and ascent. While this presents a significant penalty in performance, it would definitely be a lower risk approach for the crew. The lander would not be required to meet up with any resource or asset on the Mars surface to enable a safe return to Mars orbit. The crewed lander could even perform an abort to orbit without landing, although this could only take place in the final seconds before touchdown to protect against a landing gear failure or a more hazardous than expected landing site.

The different elements of the architecture are shown in Fig. 2. The crew would be transported to Mars orbit and later returned to Earth in the TransHab. Docked to the TransHab are the Crew Exploration Vehicle (CEV) and a Contingency Consumables Module (CCM), described in Section IV A. The TransHab has a large propulsion module that would be utilized for MOI, Trans-Earth Injection (TEI), orbit adjust and course correction maneuvers, and attitude control. The TransHab stack would be injected toward Mars with two Earth Departure Stages that perform TMI as a two-stage system.

All landed elements utilize a common lander design that fits within a traditional blunt body entry vehicle in an integrated structural design with straightforward load paths. The most important lander type is the DAV, which transports the crew to the Mars surface and includes the Mars Ascent Vehicle (MAV) to return the crew to Mars orbit. Each lander would be injected toward Mars with a two-stage EDS system identical to that used for the TransHab.

IV. Conceptual Mission Scenario

THE mission scenario is depicted in several figures. Figure 3 shows the Earth-to-Mars leg of the DAV and its aerocapture into a high elliptical Mars orbit. The cargo landers follow an identical profile, except that direct entry and landing would be used instead of aerocapture into orbit.

Figure 4 shows the Earth-to-Mars leg of the TransHab and its propulsive capture in the same high elliptical orbit as the DAV.

Figure 5 depicts the rendezvous and docking of the TransHab and the DAV. At this point the crew could utilize the combined habitable volume of the two vehicles. Aerobraking passes are indicated to progressively lower the orbit over a period of one to three months, to be determined by the development program. Note that the heatshield of the DAV could be utilized as the leading surface for the aerobraking passes to significantly reduce heating effects on the stack. After the final orbit is achieved, the crew would transfer to the DAV and initiate the descent to the Martian surface.

Figure 6 depicts the EDL phase of the DAV. This would be identical to the EDL approach used for the cargo landers, except that the entry would be from low Mars orbit rather than direct entry. This is described in more detail in Section V.

Figure 7 depicts the return of the crew to Mars orbit in the MAV, docking with the TransHab, and transfer of the crew. Prior to the TEI burn, the MAV and the CCM would be jettisoned to eliminate the unneeded mass. Transit to the Earth and return of the crew in the CEV is that same as for DRA 5.

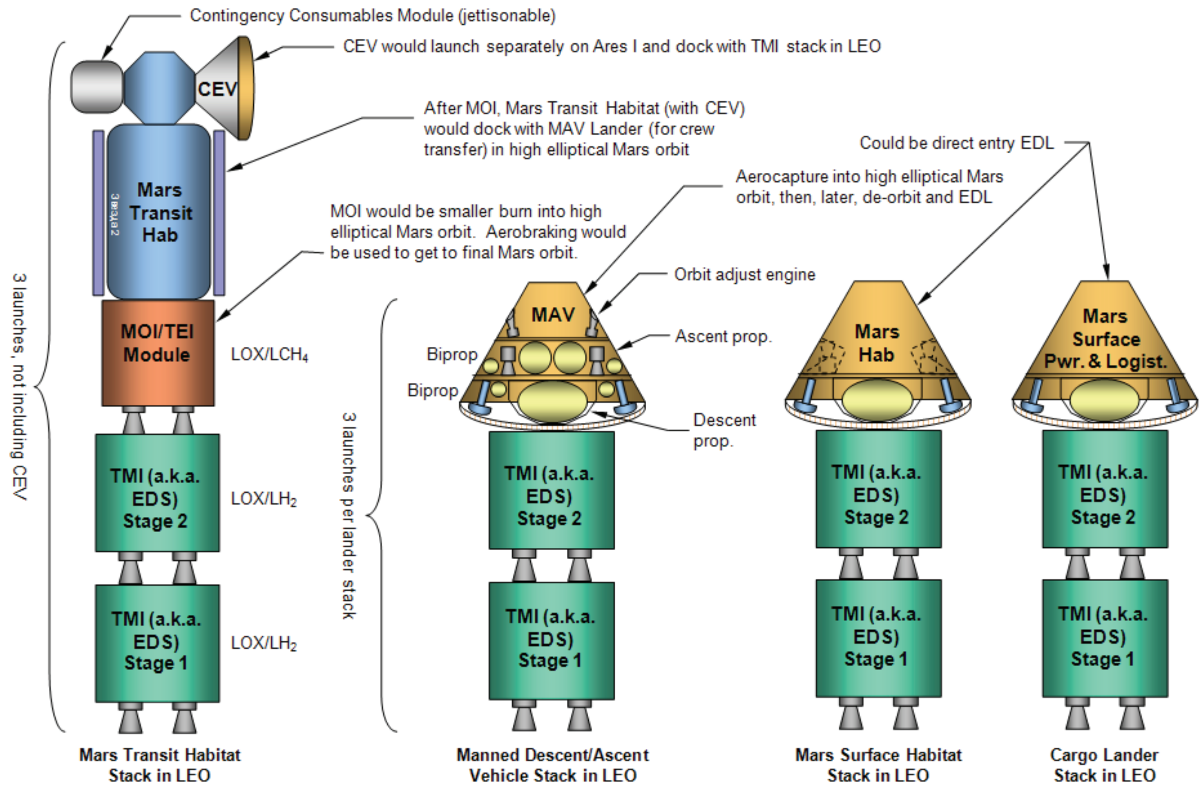

Figure 2. Austere architecture mission elements in LEO stack configurations.

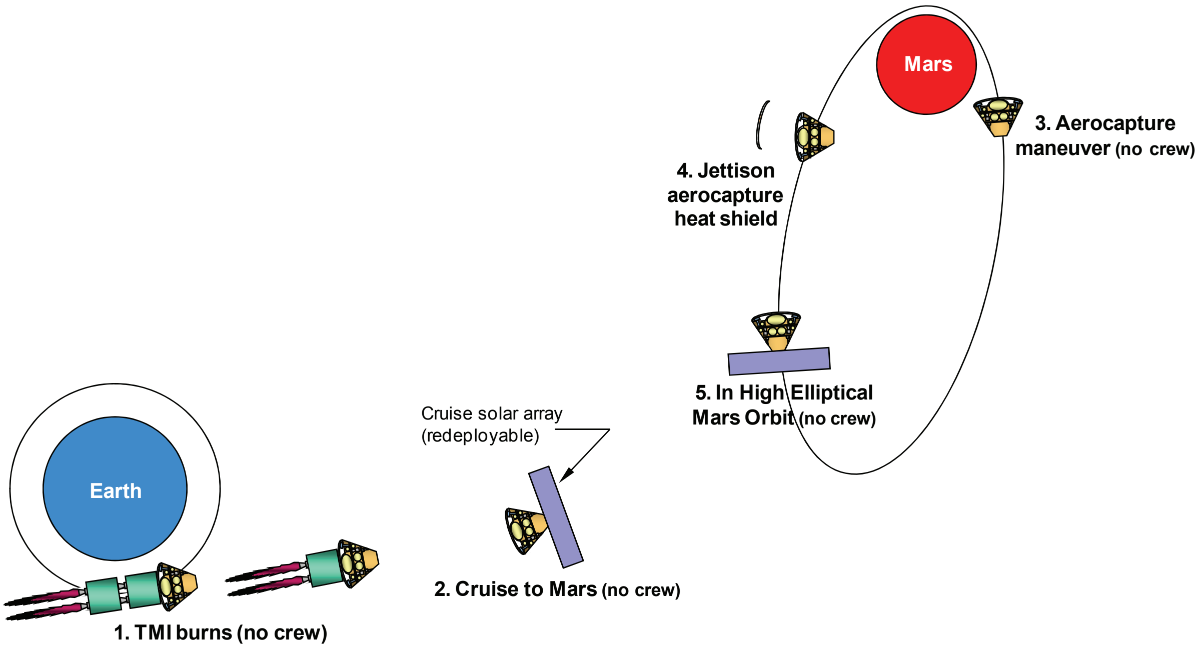

Figure 3. Earth-to-Mars leg of the DAV and its aerocapture into a high elliptical Mars orbit.

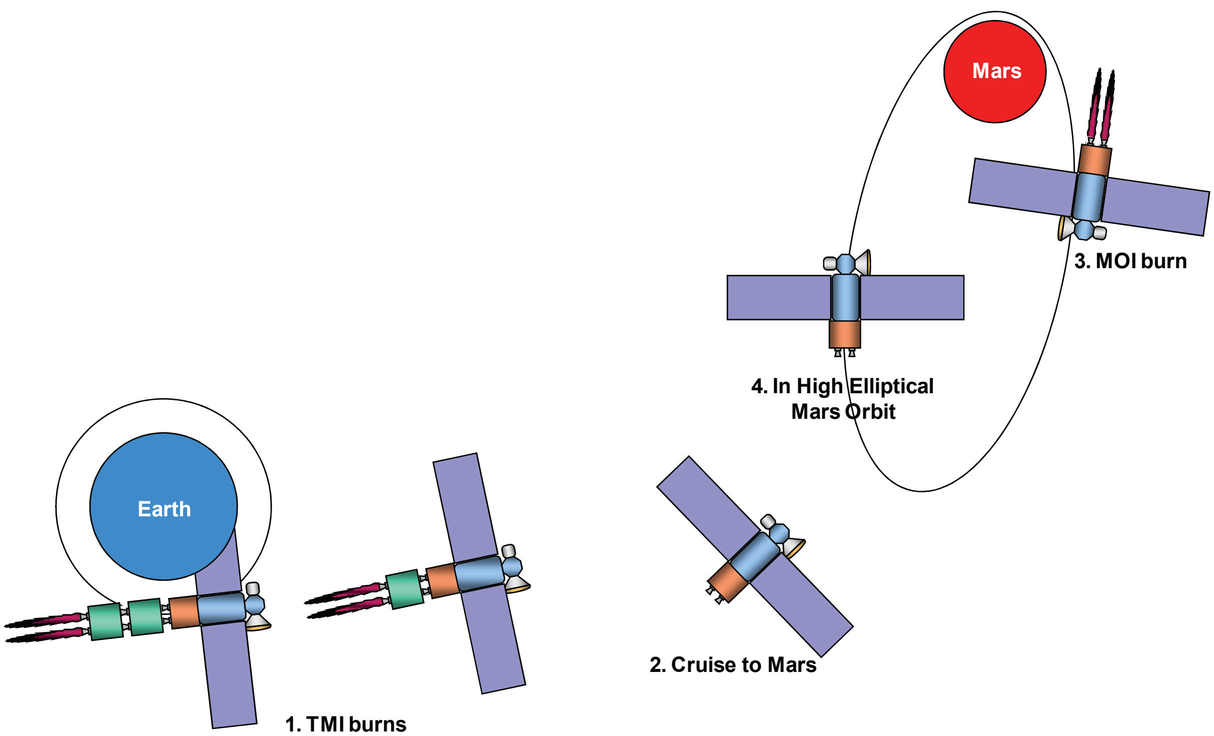

Figure 4. Earth-to-Mars leg of the TransHab and its propulsive capture in the same high elliptical orbit as the DAV.

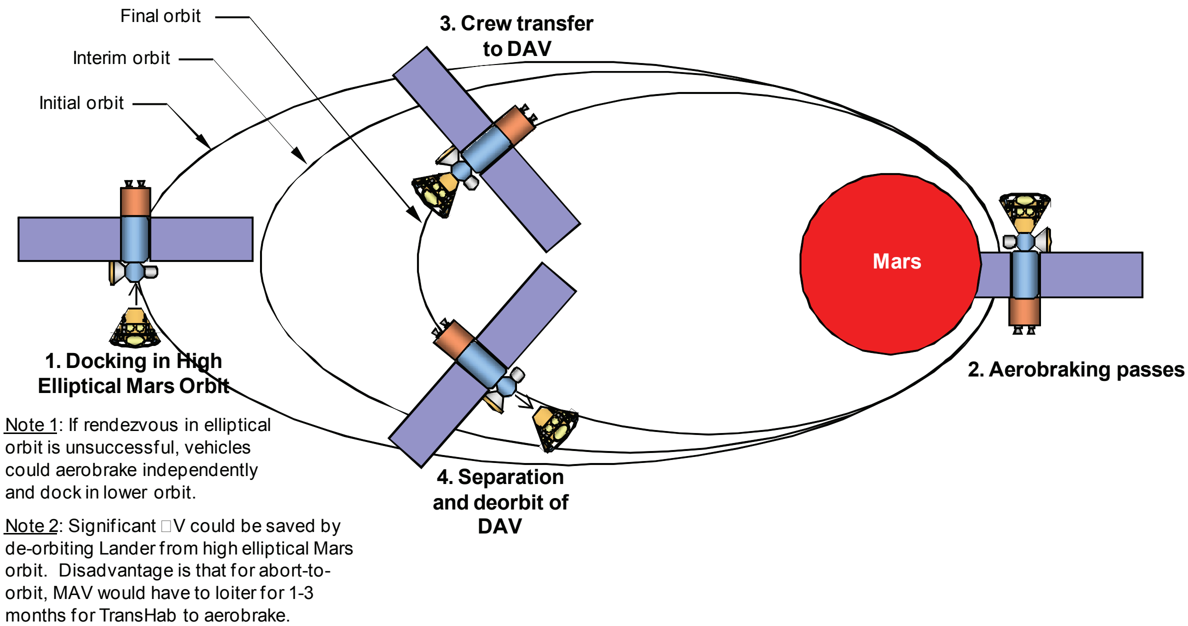

Figure 5. Rendezvous and docking of the TransHab and the DAV.

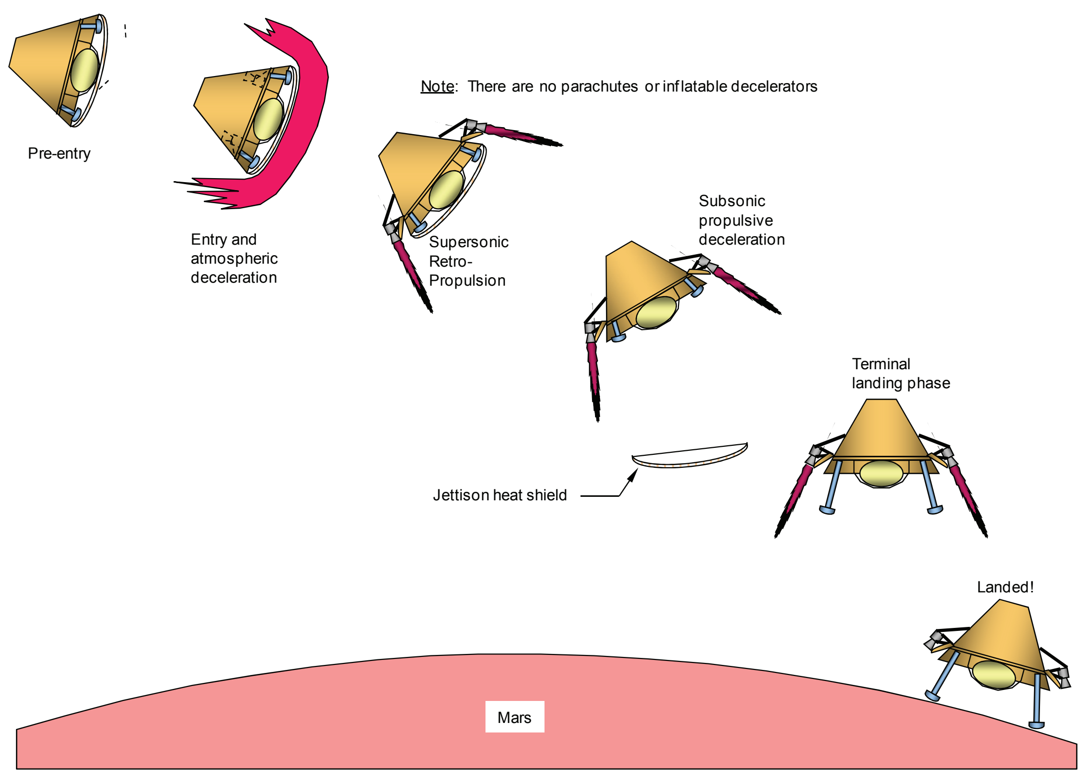

Figure 6. EDL phase of the DAV.

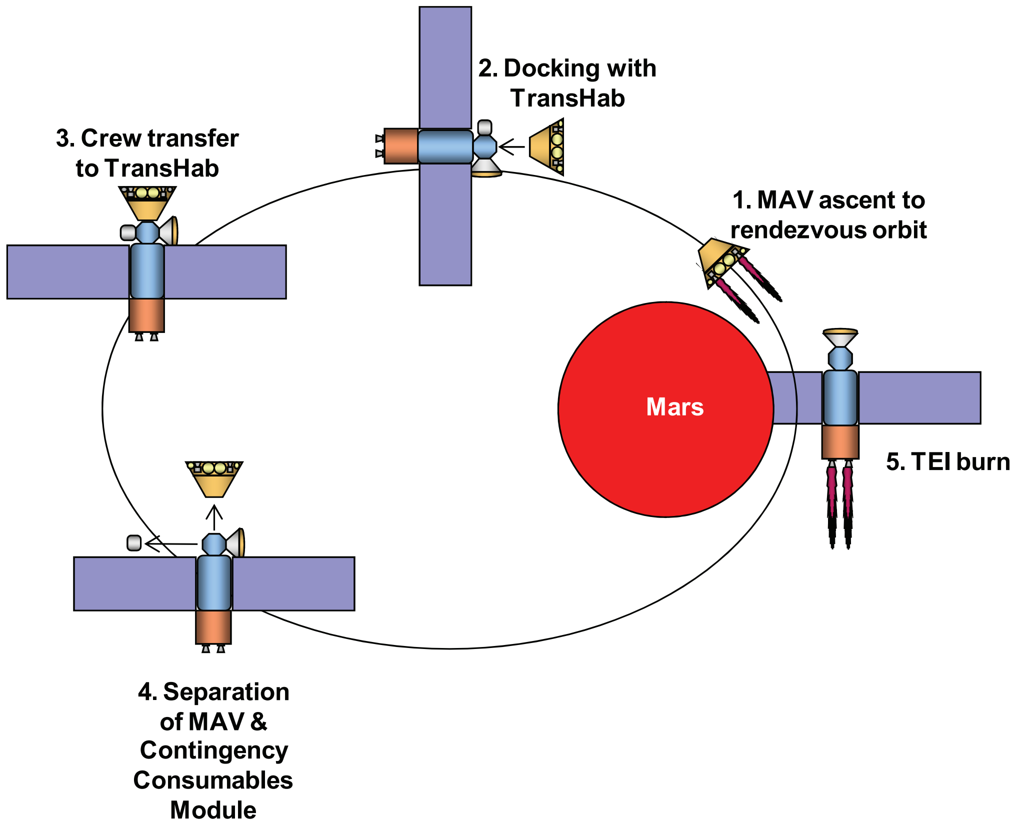

Figure 7. Return of the crew to Mars orbit in the MAV, docking with the TransHab, and transfer of the crew.

A. Key Features of the TransHab and the Earth Departure Stages

The TransHab is envisioned as being similar to the Zvyezda module on the International Space Station (ISS), but somewhat larger to support a crew of four for almost three years, along with all of the necessary logistical supplies and consumables. It would be solar powered and have multiple docking ports to connect to the DAV, the CEV, and the CCM. A fourth port would be available to allow for contingency Extra Vehicular Activities (EVAs), if needed. To allow for rendezvous and docking along the centerline of the long axis of the stack, and to manage center of mass of the stack, docked modules would need to be moved from one port to another port in-flight. This could be accomplished with a small robotic arm designed for that purpose and is a capability that was demonstrated by the Lyappa arms on the Mir space station complex[3]. For example, the CEV would rendezvous and dock with the TransHab in LEO at the end port. Before the TransHab can rendezvous and dock with the DAV in Mars orbit, the CEV would have to be moved to one of the side ports to free up the end port for the DAV.

If for some reason the crew cannot land on Mars, or if they must prematurely abort to Mars orbit, the TransHab needs to have contingency consumables to enable the crew to spend their entire time at Mars in the TransHab. These supplies would be stored in the CCM, and hopefully would never need to be utilized. Just prior to TEI, the CCM would be jettisoned and left in Mars orbit to reduce mass for the Earth return leg of the mission.

The MOI/TEI propulsion module employed for this architecture uses Liquid Oxygen/Liquid Methane (LOX/LCH_4) propellants. This has a lower I_{SP} than Liquid Oxygen/Liquid Hydrogen (LOX/LH_2), but has lower volume and presents less risk for thermal storage and propellant loss. It would be permanently attached to the TransHab, as it would be needed for course correction and attitude control all the way back to Earth.

The TransHab, with MOI/TEI module and CCM, would be initially delivered to LEO by a single Ares V launch. The CCM would then need to be transferred from the end port to a side port. The EDS, which would serve as stage 2 for TMI, would be launched on a second Ares V to LEO for rendezvous and docking with the TransHab. An identical EDS, which would serve as TMI stage 1, would be launched on a third Ares V to LEO for rendezvous and docking with the TransHab stack. The CEV would be launched separately on an Ares I vehicle to dock with the stack and transfer crew. At that point, the complete TransHab stack would be ready for TMI. The stack is depicted in Fig. 2. An important feature is that, in a contingency, this TransHab stack could safely go to Mars, enter orbit, and then return to Earth even if it is unsuccessful in ever meeting up with any of the other Mars mission elements. As a side note, this design for the TransHab stack could also be used to support other types of crewed interplanetary missions such as to an asteroid.

In the event that LOX and LCH_4 cryogenic propellant storage proves to be a difficult development or presents too great of an in-flight risk, there could be a fallback option to perform the TransHab mission with traditional bipropellant: hydrazine and nitrogen tetroxide. An analysis with the MassTracker tool[4] has shown that such an architecture closes if separate MOI and TEI stages are used for the TransHab (rather than a single-stage propulsion module). This noncryogenic architecture would require a 180 T to LEO capability for the Ares V. This may be feasible, since the current LEO capability for the 51.00.48 Ares V design is projected to be 187.7 T[5].

B. Key Features of the Landers

The common lander design for the austere architecture employs a traditional blunt body entry vehicle with a diameter in the range of 13 m. It uses an ablative heat shield and has attitude control and ΔV thrusters that can fire through ports in the backshell similar to thrusters used in the Mercury, Gemini, Apollo, Shuttle, and CEV vehicles. A deployable and refurlable solar array would be needed to provide power in LEO and during cruise to Mars. For the DAV version, the array must be retained by the MAV to provide power after return to Mars orbit from the surface.

The lander uses aerodynamic deceleration to reduce its speed as it enters through the thin Martian atmosphere. As described in Section V, no deployable decelerators are employed, but high thrust rocket engines are utilized to provide Supersonic Retropropulsion (SRP) and perform final steering and braking for soft landing on the Martian surface. This is an approach that has been assessed in papers by Korzun, et al.[6] and Christian, et al.[7]. Once the lander velocity becomes subsonic, the heat shield would be jettisoned to reduce mass and allow for the landing gear to deploy. The common lander in this study uses traditional bipropellant engines, hydrazine and nitrogen tetroxide, to reduce both development and in-flight risk and minimize volume and cooling constraints for fitting within the moldline of the vehicle. The Entry, Descent, and Landing (EDL) approach is described in more detail in Section V.

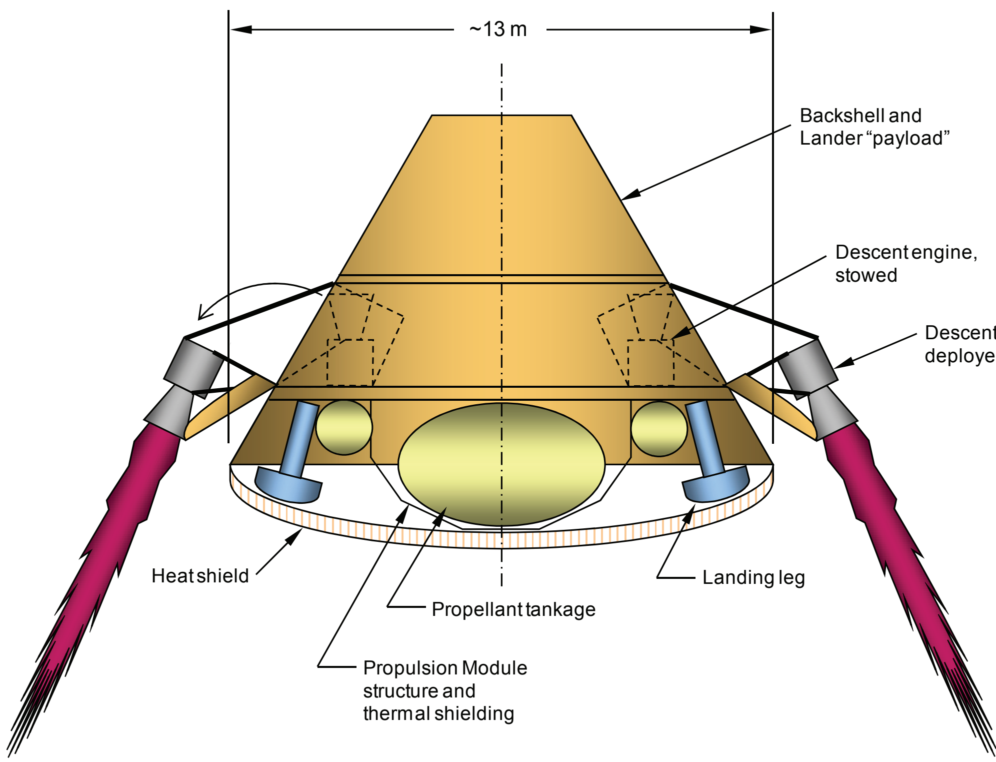

Figure 8. Common lander conceptual design.

The DAV includes a MAV within the moldline of the backshell. This contains a bipropellant ascent propulsion system and a small crew cabin, which would be extremely mass constrained in its design, similar to the cabin used in the Apollo Lunar Module. This is shown schematically in Fig. 2. The MAV would need to support a crew of four, Extra Vehicular Activity (EVA) gear, and consumables sufficient to support the crew for several weeks, to be determined by later analyses.

The SurfHab would be delivered separately to the surface, prior to the DAV, and would serve as the primary living quarters for the crew for their approximately one-year stay on the surface of Mars. The living quarters must fit within the moldline of the lander backshell. While a rigid pressurized volume within these confines might be adequate for a crew of four, using an inflatable habitat may well be worth the technology development involved in order to provide more spacious and comfortable quarters. A concept for this is depicted in Fig. 9. For this study, a Dynamic Isotope Power System (DIPS)[8] was selected over solar or fission reactor alternatives to provide power for the SurfHab. This is discussed further in Section VII.

The volume and mass constraints of the DAV and SurfHab landers make it difficult to include other needed logistical equipment for the surface mission. For this reason, the architecture considered here includes a Surface Power and Logistics Module (SPLM) to include such elements as additional DIPS power generators, one or more pressurized rovers, science equipment, and possibly additional consumables.

A high-level notional listing of the major elements of each lander type is shown in Table 1.

If successive crewed missions were conducted to the same site, continuing use of the elements already in place (i.e., SurfHab and SPLM) would allow for specialized lander types to be sent in lieu of those being reused. This would keep the program within the constraint of only two cargo landers per mission cycle. They would have to fit within the standard lander moldline and mass constraints. These specialized landers could greatly enhance the science conducted on the surface. Some examples of specialized landers might be a deep drilling platform, a large long-range pressurized rover, or equipment to enable exploring water seeps in cliff walls or underground caverns.

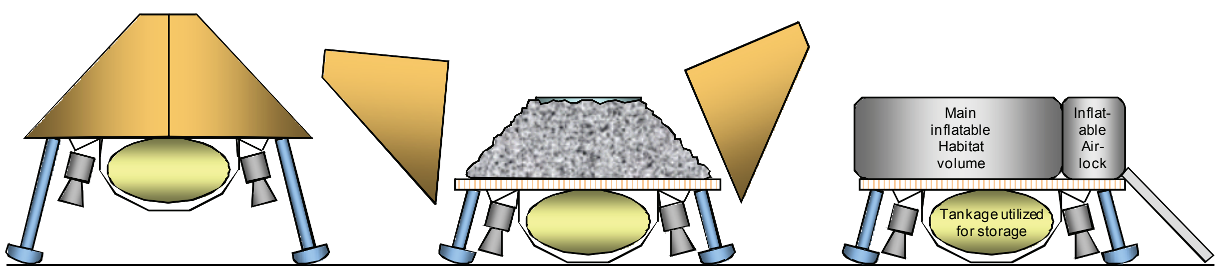

Figure 9. Concept for SurfHab with inflatable crew quarters.

Table 1. Listing of lander payload elements with mass assumptions.

		Payload Mass (T)
Descent/Ascent Vehicle		**46**
	Crew cabin	6
	Ascent Stage with propellant	40
Surface Habitat		**52**
	Pressurized Habitat with all required consumables	35
	Airlock with EVA suits	5
	Two 5 kWe radioisotope Stirling generators	1
	Small atmospheric ISRU oxygen generator	1
	Science equipment	10
Power/Logistics Module		**52**
	Two 2-man Small Pressurized Rovers, each with one 5 kWe radioisotope Stirling generator	20
	Two relocatable 5 kWe radioisotope Stirling generators (in addition to the rover units)	1
	Additional consumables	10
	Science equipment	21

V. Entry, Descent, and Landing Assessment

THE EDL approach chosen for this study relies as much as possible on blunt-body entry technology that has been employed successfully for the last 50 years. Due to the nature of Mars EDL for landers with masses much greater than the Mars Science Laboratory (MSL), it would be essentially impossible for a crewed lander to effectively utilize parachutes, as have the robotic landers to date[9]. Exotic deployable or inflatable decelerators might be effectively utilized, but it is not clear that the performance benefit would outweigh the cost and risk of the development and test program. Also, the in-flight mission risk to the crew for the possibility of a failure of the deployable decelerator(s) would have to be considered.

Slender-body entry vehicles, such as the ellipsled[10], may offer some significant performance benefits, but it is not clear that they outweigh the cost and risk of the development and test program that would be required to validate the technology. Of particular concern are the complexities of the structural load paths required and the complexity of the extraction of the lander from the slender-body aeroshell. Minimizing the complexity of the EDL events to increase crew safety has to be a major consideration in these trade studies.

For these reasons, the austere architecture has chosen a blunt-body entry vehicle of traditional design. There are no deployable decelerators. The final reduction of velocity would be performed by SRP, and these same rockets are also used for the final soft landing on the surface. A notional configuration for the lander is shown in Figs. 2, 6, and 8. In this configuration, the SRP engines are shown as being deployable to get them on the outside of the flowstream and to provide greater control authority for steering. This also avoids design issues involved with firing engines through plugs and holes in the ablative heatshield. Detailed modeling of the lander may indicate that such a feature is not required and that a configuration could be used like that described by Korzun, et al.[6], with non-deployable engines at the periphery of the heat shield, but within the vehicle moldline.

A representative EDL profile has been developed and analyzed for this lander design and is shown in Figure 10. For the DAV, the lander enters the Martian atmosphere at about 3.6 km/s and, over a period of about 6 minutes, would be aerodynamically decelerated to a point where it would be at an altitude of about 10 km with a velocity of 1.5 km/s. At this point, SRP would be initiated and, over the next 70 seconds, the vehicle would be decelerated by both the rocket engines and by atmospheric drag. At this point the vehicle becomes subsonic and the heatshield would be jettisoned (see Fig. 6) to reduce mass and allow for the landing gear to deploy. The rockets continue to decelerate the vehicle over the next 20 seconds and perform a traditional soft landing on the surface similar to Surveyor, Apollo, Luna, Viking, and Phoenix.

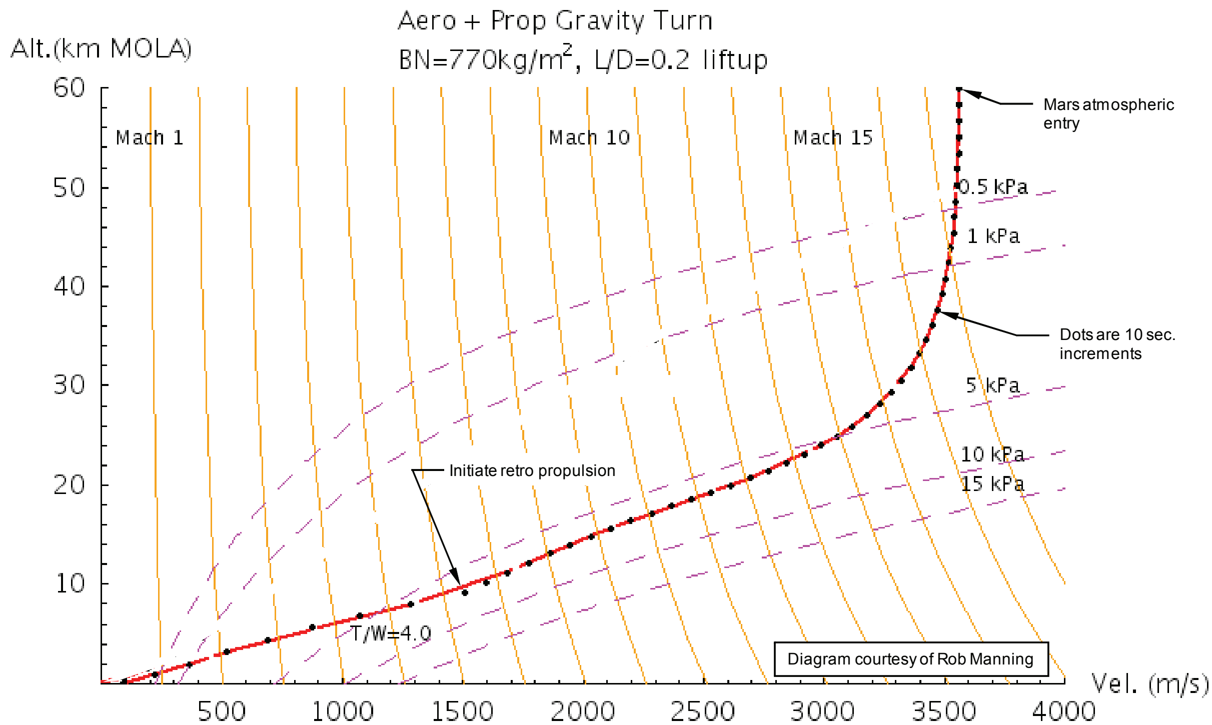

Figure 10. Representative EDL altitude and velocity timeline with mach numbers and dynamic pressure.

Similar to Fig. 10, Fig. 11 displays entry profiles for three additional EDL scenarios: entering from parking orbits with apoapsis heights of 200; 1,000; and 30,000 km. This analysis was performed with The Aerospace Corporation's Trajectory Optimization Program (TOP) and provides good agreement with the Jet Propulsion Laboratory (JPL) tool used in Fig. 10. The simulation uses an exponential atmospheric model and a first-order Martian gravity model; therefore, the Mach and dynamic pressure lines are slightly different. No vehicle-banking maneuvers are included during atmospheric entry, and the thrust vector is controlled opposite velocity during terminal descent. Figure 11 shows the high dynamic pressure loads experienced by vehicles entering from highly elliptical orbits using this control scheme. Direct-entry profiles of the cargo vehicles would also have to withstand this type of loading. Maximum dynamic pressures for the three-entry scenarios are 5.9, 5.7, and 17.5 kPa, respectively. The lofting nature of the third profile allows additional time to aerodynamically slow the vehicle before terminal descent begins. All three profiles use a mass fraction near 0.75, which equates to approximately 41 MT of propellant used during the descent. Sensed acceleration profiles can be seen in Fig. 12.

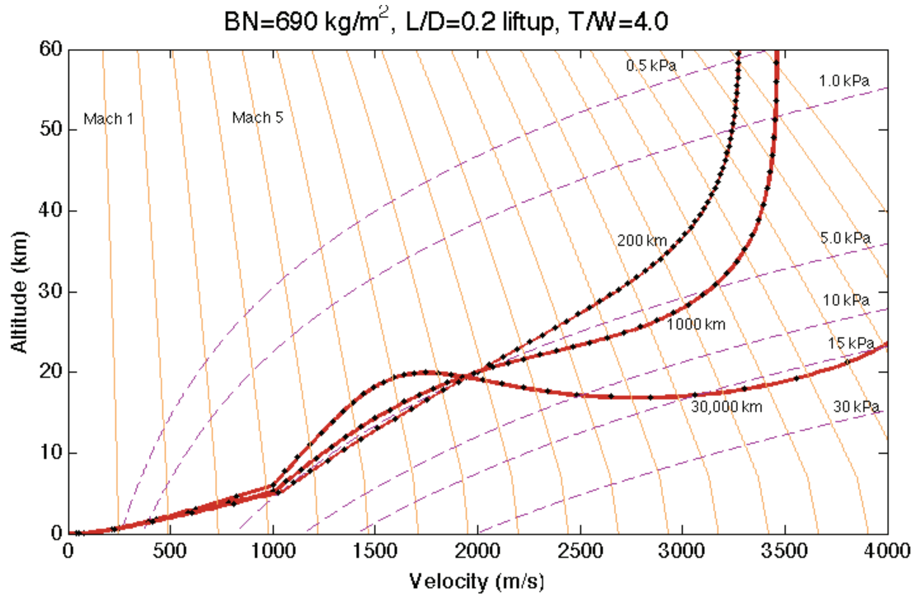

Figure 11. Entry profiles from orbit altitudes of 200 km; 1,000 km; and 30,000 km.

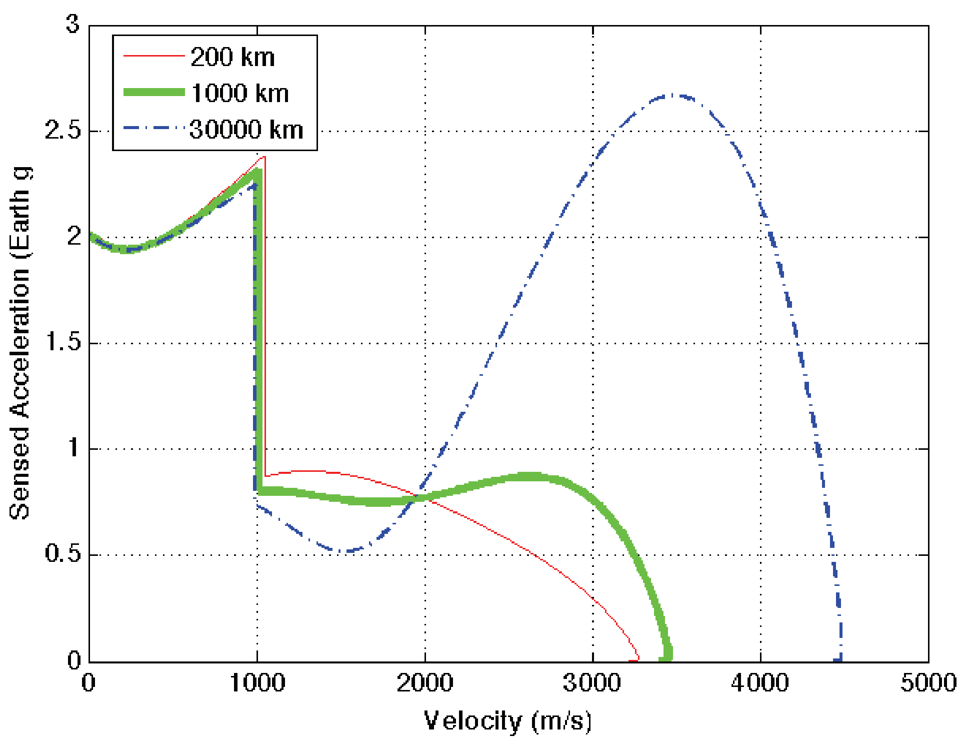

Figure 12. Sensed acceleration for three entry scenarios.

VI. Mass and Performance

THE dry mass estimates for the individual elements were mostly taken from the DRA 5 study or from earlier Design Reference Mission (DRM) studies. This applies to the non-propulsion and non-EDL dry masses. For many of the estimates, engineering judgment was used to revise them to be more conservative or to scale them down somewhat to be consistent with a crew of four rather than six. In some cases, analogy to historical spacecraft hardware was used as a sanity check, and estimates were revised based on those analogies. The masses are allocations, and would have to include required margins. The propulsion dry mass, EDL dry mass, and propellant estimates were based on gear ratios established by many prior studies of crewed Mars missions[11]. These gear ratios were checked and either validated or revised by analyses performed using MassTracker. The EDL gear ratio was also validated by independent runs with TOP.

The dry mass estimates, gear ratios, and wet mass estimates are summarized in Table 2. These estimates can only be considered notional. Table 2 also indicates the number of launch vehicles required to perform one mission cycle, broken down by mission element.

Table 2. Mass assumptions and gear ratios for major mission elements.

Element	Mass (T)	"Gear Ratio"	Prop. type	Ares V	Ares I	Comments
MAV Cabin	6.2					2.9 times Apollo Ascent Module dry mass
MAV Total	45.9	7.4	NTO/MMH			Includes ascent propulsion and structure
Lander Descent Stage	119.3	3.6	NTO/MMH			Includes separate aerocapture heat shield
Lander/MAV Total	**165.2**			**1**		
MAV EDS's	**330.3**	3.0	LOX/LH_2	**2**		Two stage assembly requiring two Ares V launches
Cargo Lander payload	52.0					Can be Habitat, or Surface Power and Logistics Module
Cargo Descent Stage	114.4	3.2	NTO/MMH			
Cargo Total	**166.4**			**1**		
Cargo EDS's	**332.8**	3.0	LOX/LH_2	**2**		Two stage assembly requiring two Ares V launches
CEV	**10.0**				**1**	Current Orion CM mass
Transit Habitat	35.0					For comparison, Mir Core Module mass = 21 T
Contingency Module	7.0					Emergency supplies for Mars abort to orbit (jettisonable)
Subtotal	52.0					
MOI/TEI Module	114.4	3.2	LOX/LCH_4			Assumes 1.2 km/s MOI followed by aerobraking
Subtotal (w/o CEV)	**156.4**			**1**		A single Ares V launches MOI/TEI module plus Habitat
EDS Stages	**332.8**	3.0	LOX/LH_2	**2**		Two stage assembly requiring two Ares V launches
Grand Total	1,983.1			12	1	Incl. 2 Cargo Landers (Surf. Hab., Power & Logistics)

VII. New Technology Development

THE austere architecture avoids new technology development. New technologies were considered only where needed to enable the mission, reduce cost, or reduce development or mission risk.

The key new technology is SRP for EDL. This would be required by any crewed Mars mission architecture, including DRA 5, and would be an enabling technology.

For surface-power technology, DIPS was chosen over solar and fission alternatives because of clear advantages in mass, volume, reliability, and ease of deployment. Although it is a new technology, it has high heritage and should be a low risk technical development. The biggest issue for DIPS would be the acquisition of the Plutonium 238 (Pu^{238}) needed to fuel the units. The architecture considered here generously assumed six DIPS for a total surface power capability of 30 kWe. This would require six times the amount of Pu^{238} used for the Cassini mission to Saturn. Detailed analysis of specific mission scenarios might show that a lower number of DIPS could be adequate.

A power architecture fallback would be to rely primarily on solar power with one or two DIPS available for emergency survival power. Although the large solar array development and deployment approach would require significant technology development, it may present lower cost and risk than a surface fission power source. Reliance entirely on solar or fission power presents significant risks for near total power loss in the event of severe dust storms or a nonrecoverable reactor SCRAM; therefore, DIPS would be a highly desirable, possibly enabling technology for crewed Mars missions. For these reasons, DRA 5 includes DIPS for emergency survival power.

LOX/LCH$_4$ propulsion would be highly desirable for the TransHab due its greater I_{SP} than the traditional bipropellant utilized in crewed and robotic missions today. LOX/kerosene could be a credible fallback with almost as high I_{SP}, but it still requires new thermal control technology to store cryogenic LOX for the duration of the TransHab mission. As mentioned in Section IV A, traditional bipropellant would be a credible fallback if Ares V can deliver greater than 180 T to LEO.

A small ISRU unit for extracting oxygen from the Martian atmosphere would be highly desirable to provide additional oxygen for life support. The mission would be designed to be executable without relying upon the ISRU unit, so it would not be mission-enabling; however, if it functions successfully, it could provide significant replacement oxygen, which would allow more operations of the airlocks in both the TransHab and pressurized rovers and, thereby, allow more EVAs for surface exploration.

An inflatable habitat would be desirable to provide a larger volume living quarters for the SurfHab, but it would not be an enabling technology. Radiation and leakage issues would have to be considered in the risk assessment to support the decision to use such technology.

VIII. Development Schedule

THE schedule and cost estimates developed in this study assume that Orion, Ares I, Ares V, and their supporting infrastructures are developed by the Constellation Program. The notional development schedule shown in Fig. 13 would require three Ares V rockets to be available to launch the DAV test flight in 2022.

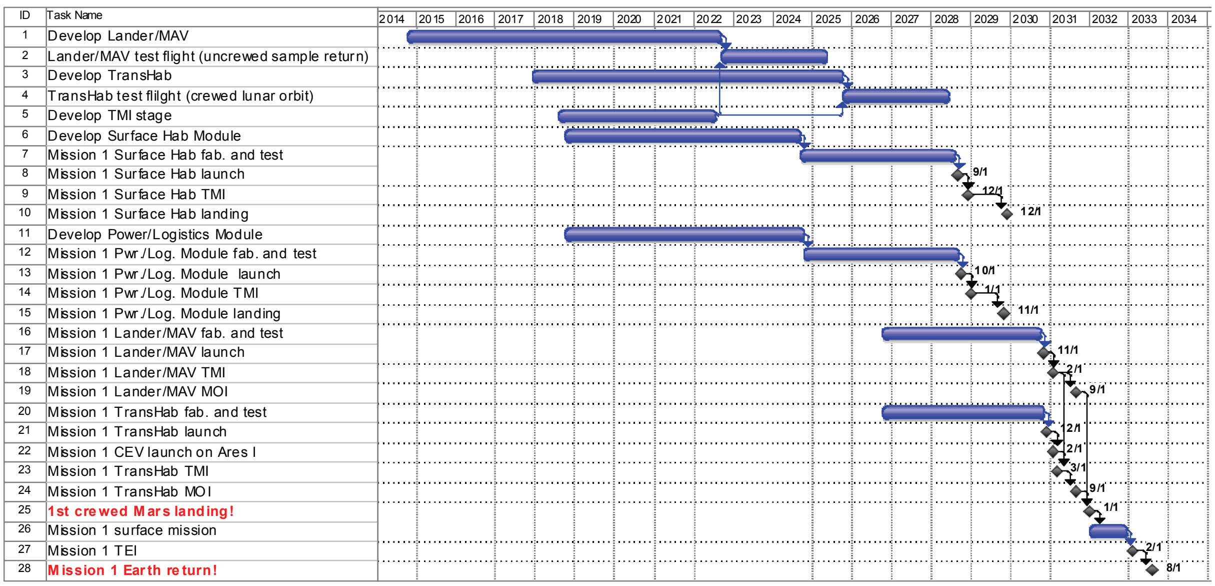

Figure 13. Development schedule.

IX. Flight Tests

BEFORE committing to crewed missions, the DAV would need to be completely tested out in the Mars environment with an uncrewed flight of the hardware and software. This is a key focal point of the development schedule in Fig. 13. The DAV test flight requires a full-up system with two EDS modules to launch the stack to Mars. This requires three Ares V launches. The flight would validate all phases of the DAV mission: LEO assembly, TMI, cruise, aerocapture into high Mars orbit, aerobraking to low Mars orbit and, most importantly, EDL. It would remain on the surface for the duration required by a crewed mission; then the MAV would be launched into Mars orbit. In order to utilize the test flight for scientific purposes, the MAV could deliver a Mars sample container to Mars orbit as part of a robotic Mars Sample Return mission.

The DAV test flight would also validate the EDL design for the cargo landers, since they utilize an identical mold line and identical EDL subsystem design. The cargo landers do employ direct entry rather than entry from low Mars orbit, so that difference would have to be validated by analysis.

The TransHab design could be validated in a relevant environment without having to travel to Mars. This could be achieved by a test flight in near-Earth space that could be crewed with abort-to-Earth capability in the event of problems. A three-year flight would fully validate the TransHab, and this could be conducted in LEO, Lunar orbit, near-Earth space such as L_2, or some combination of those regions. It could be fully crewed for the duration, crewed for only certain intervals in the test flight, or staffed by rotating crew teams. One EDS module would be desirable to validate interfaces and performance, so two Ares V launches would be needed to support the test flight.

X. Cost Estimates

THE cost estimates provided here for the austere architecture are notional and draw heavily upon the cost estimates performed for DRA 5. Costs are based upon NAFCOM[12] models, top-level historical analogies, and results from previous Mars mission studies, which were in turn based on NAFCOM results and historical analogies. The costs for the test flights and operational flights include the Ares V launches, Ares I launches, and the Orion crewed spacecraft. These estimates do not contain the sustaining costs for the Constellation Program nor the mission operations costs, all of which tend to be fixed annual costs that do not vary significantly with flight rate.

Table 3 presents cost estimates by year for each of the major elements of the development and flight program. Only development costs and Mission 1 costs are added into the bottom-line total. Mission 2 and beyond would carry roughly the same recurring costs as Mission 1 but would be phased in four-year increments. Cost reserves of 50% are included in the bottom line estimates. For this particular profile, funding peaks at $7.2 B in FY 2019 through FY 2021, then comes down to a sustaining rate of about $5 B per year for a continuing series of human missions to Mars, with a new crew launching every four years.

While Table 3 shows costs for an all-U.S. program, Table 4 presents costs to the U.S. for an international program scenario in which foreign partners would provide the TransHab along with its MOI/TEI propulsion module. The rows highlighted in yellow indicate where costs to the U.S. have been reduced or eliminated. In this particular case, contributions from foreign partners reduce the peak funding for the U.S. to about $6 B per year.

Table 5 presents a summary of estimated development costs with 50% cost reserves.

Table 6 presents a notional launch timeline for a four-year crewed Mars mission cycle.

Notional configurations for the different types of Ares V launches are depicted in Fig. 14. Indicated at the bottom of the figure is the number of launches required for each configuration for a four-year mission cycle, which would send a single, four-person crew to Mars and return them to Earth. Eight of the twelve launches are EDSs, and these are of identical configuration. This presents opportunities for increasing production efficiency and reducing costs, perhaps even using a Commercial Orbital Transportation Services (COTS) provider for this service. Additionally, significant insurance against a launch failure could be realized by providing an extra EDS and Ares V to be available on short notice to replace a failed launch.

Table 3. Notional cost profile for major program elements for an all-U.S. program.

		2015	2016	2017	2018	2019	2020	2021	2022	2023	2024	2025	2026	2027	2028	2029	2030	2031	2032	Total
Descent/Ascent Vehicle (DAV)	dvmt.	500	1,000	1,500	2,200	2,500	2,000	500												10,200
Earth Departure Stage (EDS)	dvmt.	200	300	400	600	500	400	200												2,600
Ares V upgrade	dvmt.	300	400	600	600	600	300	200												3,000
DAV test flight	test						500	1,500	1,390											3,390
TransHab Development	dvmt.				200	400	600	600	700	500	400	300								3,700
MOI/TEI Stage	dvmt.					200	300	400	600	600	400	200								2,700
CEV upgrade	dvmt.							200	200	200	200	200								1,000
TransHab test flight	test									504	1,008	504								2,015
SurfHab Development	dvmt.					300	300	700	750	800	750	750	350							4,700
Power/Logistics Module	dvmt.					200	400	500	700	750	550	400	300							3,800
SurfHab	Flt. 1												851	1,703	851					3,405
Power/Logisitics Module	Flt. 1												920	1,840	920					3,680
DAV	Flt. 1														848	1,695	848			3,390
TransHab/Crew	Flt. 1														633	1,265	633			2,530
SurfHab	Flt. 2																851	1,703	851	
Power/Logisitics Module	Flt. 2																920	1,840	920	
DAV	Flt. 2																		848	
TransHab/Crew	Flt. 2																		633	
Reserves/margin (50%)		500	850	1,250	1,800	2,350	2,400	2,400	2,170	1,677	1,654	1,177	1,211	1,771	1,626	1,480	1,626	1,771	1,626	25,055
Total ($M)		**1,500**	**2,550**	**3,750**	**5,400**	**7,050**	**7,200**	**7,200**	**6,510**	**5,031**	**4,961**	**3,531**	**3,632**	**5,314**	**4,877**	**4,440**	**4,877**	**5,314**	**4,877**	**75,165**

Table 4. Notional cost profile for the U.S. for major program elements for an international program.

		2015	2016	2017	2018	2019	2020	2021	2022	2023	2024	2025	2026	2027	2028	2029	2030	2031	2032	Total
Descent/Ascent Vehicle (DAV)	dvmt.	500	1,000	1,500	2,200	2,500	2,000	500												10,200
Earth Departure Stage (EDS)	dvmt.	200	300	400	600	500	400	200												2,600
Ares V upgrade	dvmt.	300	400	600	600	600	300	200												3,000
DAV test flight	test						500	1,500	1,390											3,390
TransHab Development	dvmt.																			0
MOI/TEI Stage	dvmt.																			0
CEV upgrade	dvmt.							200	200	200	200	200								1,000
TransHab test flight	test									335	500	335								1,170
SurfHab Development	dvmt.					300	300	700	750	800	750	750	350							4,700
Power/Logistics Module	dvmt.					200	400	500	700	750	550	400	300							3,800
SurfHab	Flt. 1												851	1,703	851					3,405
Power/Logisitics Module	Flt. 1												920	1,840	920					3,680
DAV	Flt. 1														848	1,695	848			3,390
TransHab/Crew	Flt. 1														440	805	440			1,685
SurfHab	Flt. 2																851	1,703	851	
Power/Logisitics Module	Flt. 2																920	1,840	920	
DAV	Flt. 2																		848	
TransHab/Crew	Flt. 2																		440	
Reserves/margin (50%)		500	850	1,250	1,700	2,050	1,950	1,900	1,520	1,043	1,000	843	1,211	1,771	1,529	1,250	1,529	1,771	1,529	21,010
Total ($M)		**1,500**	**2,550**	**3,750**	**5,100**	**6,150**	**5,850**	**5,700**	**4,560**	**3,128**	**3,000**	**2,528**	**3,632**	**5,314**	**4,588**	**3,750**	**4,588**	**5,314**	**4,588**	**63,030**

Table 5. Notional development costs.

Development Item	Comments	Cost Basis or Analogy	Cost ('09 $B)
Earth Departure Stage (EDS)	Incl. rend. & docking system (ATV heritage)	Ares V EDS	3.9
Lander/MAV dvmt.	Incl. Supersonic Retro-Propulsion (SRP) dvmt.	Orion development	15.3
Test flight: Lander/MAV, unmanned	Might be part of an MSR mission		5.1
Mars Surface Habitat	Leverages off of earlier lunar surface habitiat	ISS module	7.1
Surface Power/Logistics Module	Assuming Stirling RTG's		5.7
CEV Block Upgrade for Mars			1.5
Trans-Mars Habitat	Incl. MOI/TEI prop. module	ISS module	9.6
Test flight: Trans-Hab & CEV	Manned flight in LEO or circumlunar		3.0
	Total		51.2
Notes:	Cost bogeys do not include mission or ground operations or facilities.		
	Costs include 50% margin over DRA 5/Aerospace Corp. estimates.		
	Lander/MAV test flight doesn't incl. any costs for an MSR mission.		
	This table doesn't include any Ares V upgrade costs.		

Table 6. Notional launch timeline.

Time	KSC Launch	LEO Launch	Vehicle	Comments
M-875 days	Mars Surface Habitat		Ares V	
M-870 days	Power/Logistics Module		Ares V	Isotope Stirling pwr.; small pressurized rover
M-825 days	Habitat EDS 1		Ares V	
M-820 days	Habitat EDS 2		Ares V	
M-815 days		Habitat TMI	EDS 1&2	Habitat is launched to Mars
M-790 days	Power EDS 1		Ares V	
M-785 days	Power EDS 2		Ares V	
M-780 days		Power TMI	EDS 1&2	Surface Power/Logistics Module is launched to Mars
M-95 days	Lander/MAV		Ares V	
M-90 days	Trans-Mars Habitat		Ares V	Based on Zvyezda-type module
M-45 days	Lander EDS 1		Ares V	
M-40 days	Lander EDS 2		Ares V	
M-35 days		Lander TMI	EDS 1&2	Lander/MAV is launched (uncrewed) to Mars
M-15 days	CEV		Ares I	
M-10 days	Trans Hab EDS 1		Ares V	
M-5 days	Trans Hab EDS 2		Ares V	
M		TransHab TMI	EDS 1&2	Crew is launched to Mars
Notes:	M = TMI time for crewed Trans-Mars Habitat with CEV			
	This is not necessarily the best timeline. It's just a representative example of one that might work.			

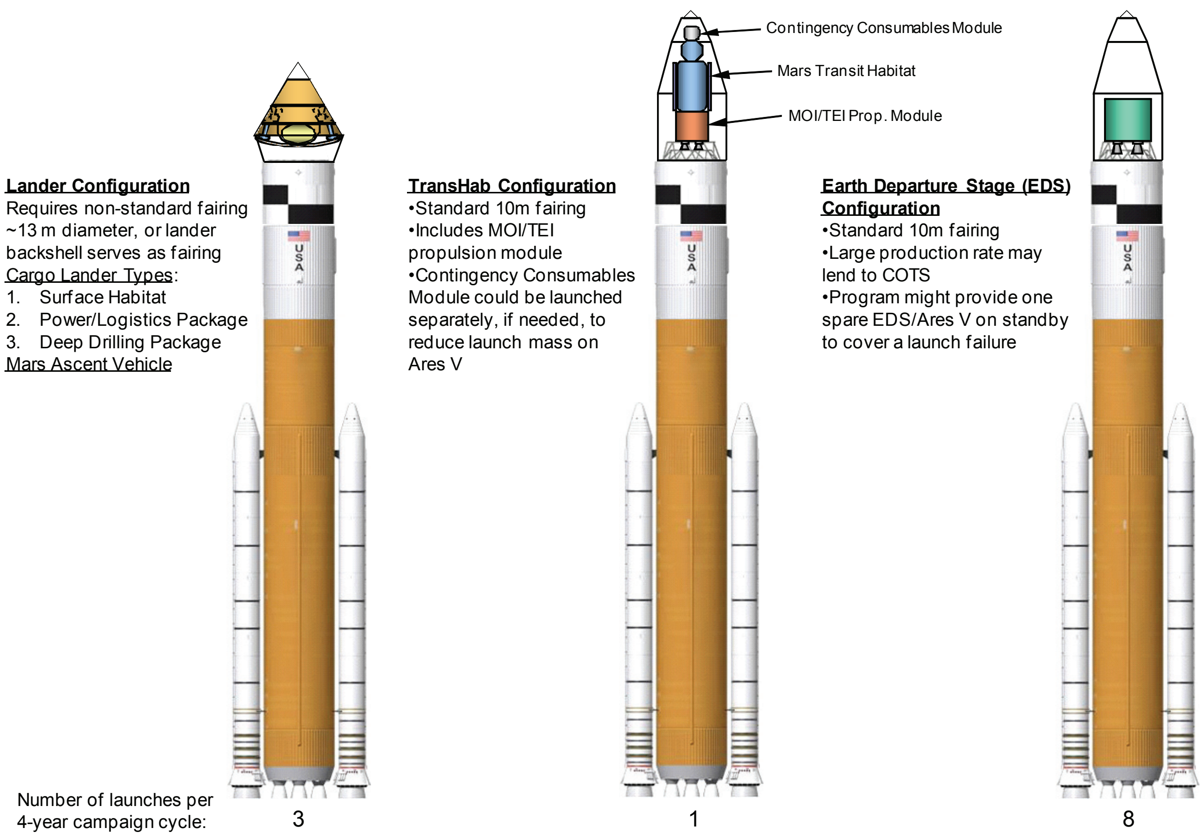

Figure 14. Ares V launch configurations and number of launches.

XI. Conclusions

CREWED missions to Mars could potentially be undertaken with very little new technology development, mainly SRP, LOX/methane propulsion, and a 5-kWe DIPS. The bulk of the development work would be straightforward engineering design, fabrication, and testing. By minimizing new technology, development risk could be low, with a program development cost and schedule similar to that of the ISS—about 18 years and $100 B.

Alternating crew and cargo launches at each biennial Mars opportunity would be an effective way to have a sustainable flat-funded program of human Mars exploration that minimizes the required Ares V launch rate. Although a continuous human presence on Mars would not be achieved, such a program would most likely be affordable without requiring a significant increase in the inflation-adjusted annual NASA budget. There are opportunities for international participation, which would both enhance the program and reduce the development and sustaining cost for the U.S.

Another conclusion of this paper is that keeping the launch rate low enough to have an affordable sustaining cost would require an Ares V LEO launch capability of about 170 T. This would also be needed to keep the launch rate down to something that would be reasonably achievable from the standpoint of turnaround time.

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