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Affordable Human Moon and Mars Exploration through Hardware Commonality

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The Vision for Space Exploration calls for the safe, affordable, and sustainable human exploration of the Moon and Mars. In order to achieve exploration affordability and sustainability, the Mars-back approach was developed. The fundamental principle of the Mars-back approach is that the system elements used for lunar exploration are a subset (in terms of design) of those used for Mars exploration; thereby the lunar exploration hardware is directly relevant to the exploration of Mars. After systematic qualitative and quantitative analysis of over 1000 lunar and over 1000 Mars exploration architectures using a discrete event simulation tool, two architectures were chosen for further analysis based on overall mission mass, mission risk, and cost: a direct return architecture for lunar exploration, and an architecture similar in concept to the 1993 NASA Mars Design Reference Mission for conjunction-class Mars exploration. Employing the Mars-back approach, the Mars exploration hardware can enable a crewed lunar direct return architecture along with one-way cargo-delivery capability, such as for a long-duration surface habitat, without the need for additional hardware development. Common system elements include the CEV for short-term habitation and Earth entry, long-term in-space and planetary surface habitats, and propulsion stages for Earth departure, deep-space maneuvers, and planetary descent / ascent. Commonality was introduced through design for the most stressing case, typically Mars, when requirements were similar, and through modular, extensible solutions when requirements differed more widely. Based on the commonality concept, a hardware development roadmap was laid out for phased development of the hardware; each phase provides increasing mission capability. Hardware development with commonality eliminates the need for any significant "development gap" between lunar and Mars exploration missions. The development approach ensures that technology and hardware development for lunar missions is directly relevant to Mars exploration. Also, extensive testing of Mars hardware can be carried out during long-stay lunar missions, thereby increasing operational experience with the equipment to be used for Mars missions and reducing Mars mission risk. As identical hardware is used, lunar missions could still be executed during Mars exploration because the production and assembly lines would still be running. Most importantly, the overall lifecycle cost for exploration of the Moon and Mars is significantly reduced by limiting the amount of hardware that must be developed. The drawback of Mars-back commonality is a certain non-optimality in the common system design which leads to increased system dry and wet mass and therefore to a potential increase in recurring cost, mainly in launch and production. Quantitative analysis of this commonality penalty shows a modest growth of Initial Mass in LEO, which appears acceptable when set against the significant savings in overall lifecycle cost that would be achieved.

I. Introduction

THE Vision for Space Exploration, announced by President George W. Bush on January 14, 2004, represents a major redirection of US space policy following upon the loss of Space Shuttle Columbia in February 2003. The Vision provides clear and ambitious goals for US manned and unmanned space flight programs over the next

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decades. For the human spaceflight program, in addition to retiring the Space Shuttle, introducing the Crew Exploration Vehicle (CEV), and the completing ISS assembly and utilization operations, major milestones include short- and long-duration missions to the lunar surface, followed by missions to Mars. The first lunar landing mission is planned between 2015 and 2020, with the first human Mars mission during the subsequent decade. Affordability and sustainability are program drivers, because after an initial modest growth in the years 2005 to 2009, the NASA budget and therefore the resources available for exploration are intended to stay constant in buying power over the next decades. In order to assure affordability and sustainability given this resource constraint, a stepping-stone approach is envisioned where each mission is based on the previous one, adding new capabilities and building blocks over time without overstraining available resources. In line with this stepping-stone approach, Human Lunar Exploration (HLE) is intended as a preparatory activity for Human Mars Exploration (HME) [1].

Experience from past and present human spaceflight programs (Apollo, Space Shuttle, ISS) suggests that manned space system design is focused on highly optimized solutions (point designs) for the respective use-case (mission scenario and destination). The development process requires both significant time and resource allocation to be completed. The consequence of this design approach is that each new development program causes budget spikes that have to be compensated by either allocating additional resources and / or by adjusting the program schedule. Developing use-case specific systems for manned space exploration also potentially increases overall development risk because of the large number of coupled development projects; this risk eventually becomes apparent in schedule delays and / or development cost increase as well. Applying this point design approach to Human Moon & Mars Exploration as outlined in the Vision for Space Exploration is likely to lead to a lengthy development gap between Moon and Mars missions, and to high overall life-cycle cost, while also limiting the build up in the stepping-stone approach to operational experience and (potentially) enhanced subsystem capabilities. While this approach might lead to the successful completion of a lunar exploration program, the development gap and the renewed major resource commitment for Mars system development significantly reduce the affordability and sustainability of any following manned Mars program, and might prove to be prohibitive.

Repeated full system design cycles with their associated development gaps can be avoided using a development strategy that emphasizes high-level hardware commonality between Moon and Mars exploration systems, i.e. using the same propulsion stage, engine, crew compartment and / or habitat design for different destinations and applications. High-level commonality intentionally introduces a certain non-optimality for any one use-case when compared to a point-designed system for the sake of reduced overall lifecycle cost and shortened development gaps between Moon and Mars exploration. The non-optimality of the common system design usually manifests itself in the form of increased wet and dry mass, thereby potentially increasing life-cycle launch and recurring production cost; this cost increase, however, is offset by a reduction of life-cycle development and non-recurring production cost. As manned space system design is usually driven by non-recurring cost, the commonality approach potentially yields significant overall life-cycle savings.

II. The "Mars-Back" Approach to Moon and Mars Exploration

In the course of the NASA Concept Exploration & Refinement Study for the Crew Exploration Vehicle and Human Lunar Exploration, the MIT / Draper team developed an approach to architecture design called the "Mars-Back Approach": as Mars is the ultimate goal of the Vision for Space Exploration, the entire development effort should be oriented towards achieving manned Mars exploration. Figure 1 provides an overview how the introduction of high-level element commonality can contribute to that goal. The top arrow (the arrow represents time) represents a schematic development schedule for point-designed Moon and Mars exploration missions without high-level commonality relative to the Initial Operating Capability of the CEV. As the budget is assumed to be fixed (in accordance with the Vision's budget plan), changes in development cost due to commonality are translated directly into schedule changes. It is also assumed that the development of the lunar exploration system is started immediately after LEO CEV initial operating capability (IOC), and the development of the Mars exploration system immediately after lunar IOC. The arrows shown below visualize the effect of commonality decisions (approaches) for element pairs on the overall schedule. The commonality strategy for the entire Moon and Mars exploration system consists of such "atomic" commonality decisions for all element pairs.

Commonality approach (1) represents what could be called a "literal" Mars-Back approach: the system element in question is designed for Mars without regard to lunar exploration, but is then used for Moon missions; this way no lunar-dedicated version needs to be developed. Using Mars exploration system element designs for lunar missions will potentially increase the development resources required for lunar system development, thereby pushing the first lunar mission to the right, but will decrease the Mars development effort thereby enabling earlier Mars missions. More of the development work is now done before initial lunar missions. Mars exploration affordability and sustainability is increased for the cost of more expensive and later lunar operating capability. An additional benefit to this approach would be significantly increased experience with the common elements of the Mars exploration hardware.

Commonality approach (2) employs high-level element commonality within the Mars exploration system. Candidate elements for commonality would be long-duration habitats, propulsion stages, engines, landing gears, and aero-shells. The resulting reduction in Mars development cost decreases the lag between Moon and Mars missions and thus increases affordability and sustainability of Mars exploration. Lunar missions do not directly contribute hardware operating experience for the Mars hardware element.

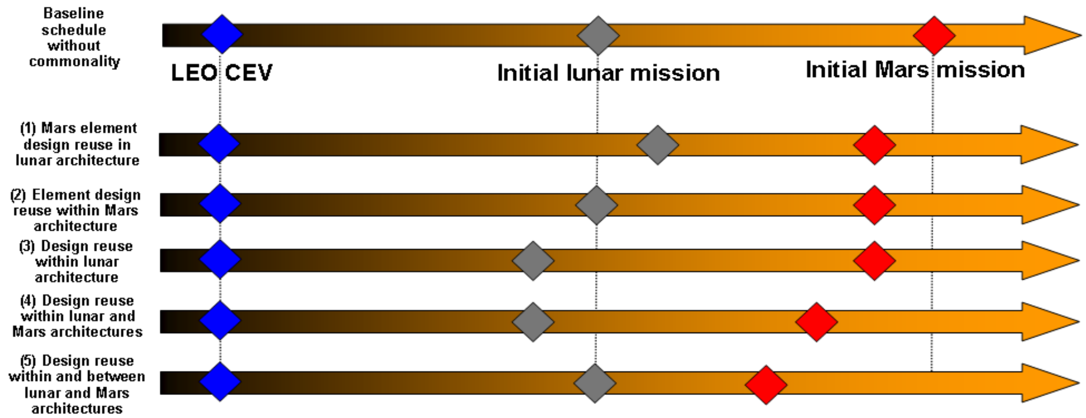

Figure 1. Simplified view of the impact of element commonality decisions on the Initial Operating Capability (IOC) of lunar and Mars exploration systems. *Design reuse within an architecture implies reuse of the same element design on different vehicles or the same vehicle within the same architecture; design reuse between architectures implies reuse of an element design in vehicles used in different architectures.*

Commonality approach (3) is identical to approach (2) except that commonality is introduced between elements of the lunar exploration architecture instead of the Mars architecture. On first sight, only the lunar architecture benefits from this commonality decision. As the lunar system development is completed earlier, however, Mars IOC is also accelerated even for a customized Mars system design because the development resources are available earlier.

Commonality approach (4) features commonality between elements within the lunar architecture and within the Mars architecture, but not across the Moon and Mars architectures; i.e., it represents the sequential application of commonality approaches (2) and (3). The order of application is irrelevant; the application of the approaches is commutative. The Mars development lag is significantly reduced for two reasons: as the development time for the lunar exploration system decreases, both the lunar IOC and the Mars IOC move left, even for a completely point-designed Mars system. This is again due to the fact that the Mars development resources are available earlier. Introduction of element commonality into the Mars system then further decreases the development time for Mars.

Commonality approach (5) consists of the sequential application of approaches (1), (2), and (3) for element pairs within and between Moon and Mars architectures. Approach (1) delays lunar IOC and speeds up Mars IOC. Approach (2) speeds up the Mars IOC further without impacting lunar IOC, and approach (3) moves both the lunar and Mars IOC to the left. The net effect is an unaltered lunar IOC and a significantly decreased lag between Moon and Mars IOC.

For some element pairings within and between Moon and Mars architectures, the penalty on recurring cost introduced by commonality can be prohibitive; in these cases customized element designs should be employed as shown for the baseline approach. The commonality strategy described in this paper is based on a multitude of decisions employing a varied combination of the approaches described above. In terms of the effect on the overall system, approaches (4) and (5) have the most desirable outcome: approach (4) leads to acceleration of both the Moon and Mars programs, approach (5) significantly decreases the Moon – Mars lag while keeping the baseline lunar IOC schedule; when feasible, the commonality strategy was therefore governed by approaches (4) and (5).

It should be noted that the description of high-level commonality effects presented here is not a complete and rigorous model of commonality effects on manned Moon and Mars exploration because it assumes the effects of all the commonality decisions to be identical for commonality within and between Moon and Mars architectures; in reality these effects would have to weighted and could only be calculated taking into account uncertainty. Also, there might be other high-level commonality approaches not described here. The scheme presented in Fig. 1 is therefore used only as a tool to qualitatively understand the effects of basic commonality decisions.

III. Moon and Mars Transportation Architectures

In order to carry out analysis of high-level commonality options for lunar and Mars exploration systems, specific system architectures need to be chosen. The process used to identify the architecture pair consisted of four steps:

- Enumeration of a large number (1162 for Moon and Mars each) of feasible transportation architectures
- Quantitative analysis based on crew compartment, long-term habitat, and propulsion stage models
- Screening of the most mass effective architectures for both the lunar analysis and Mars analysis
- Selection of the lunar and Mars reference architectures based on development cost & risk, crew safety

The individual steps of this process are described in more detail in the following subsections.

A. Systematic Generation of Transportation Architectures Using OPN

In order to determine the right lunar and Mars architectures, a systematic qualitative analysis of trasportation architectures was carried out by the MIT / Draper team during the base period of the NASA CE&R study. A graphical programming language tool called Object-Process-Network (OPN) [2] provided the basis for a generic human transportation model for missions to planetary surfaces. The transportation architectures were characterized by the number and kind of habitation and propulsion elements used on specific “flights” in the mission scenario. A flight was characterized by specific start and end points, and by being either manned or unmanned. Using OPN to model the transportation process from LEO to the lunar surface, and incorporating decision logic on the feasibility of flights into the OPN-based model, 1162 distinct and feasible transportation architectures were generated for each of the Moon and Mars missions. The composition of each transportation architecture was provided in a matrix format which served as primary output to a quantitative analysis tool. For a more detailed description and reference of the qualitative transportation architecture analysis, please refer to [3]. Although lunar mission mode analysis was given considerable attention during the early development phase of the Apollo program [4], and although multiple Mars mission modes have been proposed in studies[5-8] the systematic analysis based on OPN generating 1162 uniquely feasible architectures for both Moon and Mars is the most comprehensive one known to the authors.

B. Quantitative Analysis of Transportation Architectures

Based on the architecture matrix output from the OPN-based model, the 1162 transportation architectures for Moon and Mars sorties each were analyzed quantitatively with a Matlab-based integration tool. The tool enabled analysis of the architectures for different mission types and durations, e.g. opposition class or conjunction class Mars missions, short lunar missions (surface stay on the order of days), and long lunar surface stays (on the order of months). Also, different technology choices were investigated, including advanced propulsion (nuclear thermal propulsion, nuclear / solar electric propulsion), as well as In-Situ Propellant Production (ISPP) both for lunar and Mars missions. The integration tool allowed for ranking and visualization of the quantitative results.

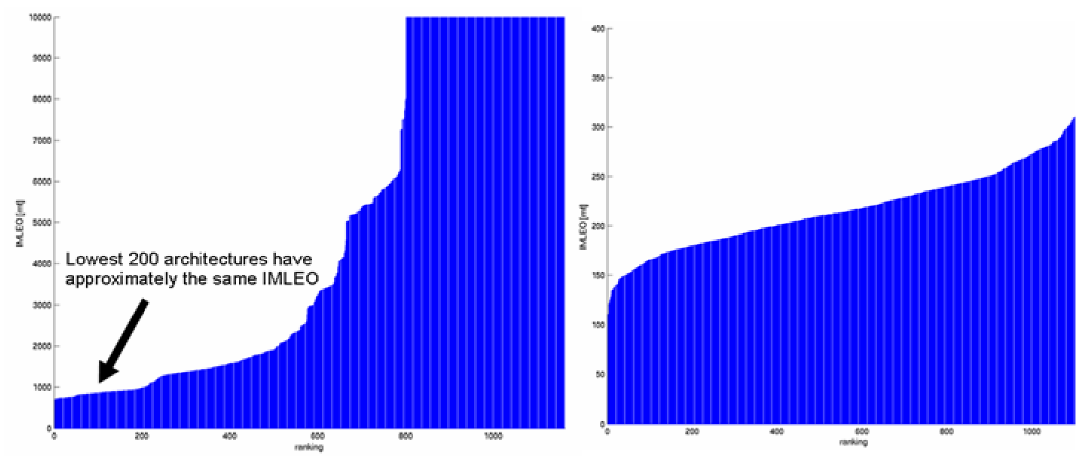

Figure 2. Ranked Injected Mass in LEO (IMLEO) results for 1162 Mars architectures (left), and lunar architectures (right). *x-axis: architectures; y-axis: IMLEO; analysis for chemical propulsion, conjunction class Mars mission and short-stay lunar mission (7-day surface stay).*

Figure 2 provides an overview of the ranked results for Mars and Moon architectures. The first several hundred Mars architectures (left) do not show a major increase in IMLEO, whereas among the first several hundred lunar architectures (right), there is a pronounced increase in IMLEO. For a more detailed description of the integration tool and Moon and Mars architecture results, please see Ref. 9.

C. Identification of Candidate Lunar and Mars Architectures Using Screening Criteria

The 100 lowest IMLEO architectures for Moon and for Mars provided the basis for the selection of three candidate architectures each, which were investigated in more detail. Beyond architecture mass, the following screening criteria were used to select architectures:

- Crew safety and mission risk (number of mission critical events, rendezvous & docking, etc.)
- Operational simplicity (reconfiguration, number of vehicles involved, etc.)
- Perceived development risk
- Suitability for high-level commonality (propulsive elements, habitats, crew compartments)

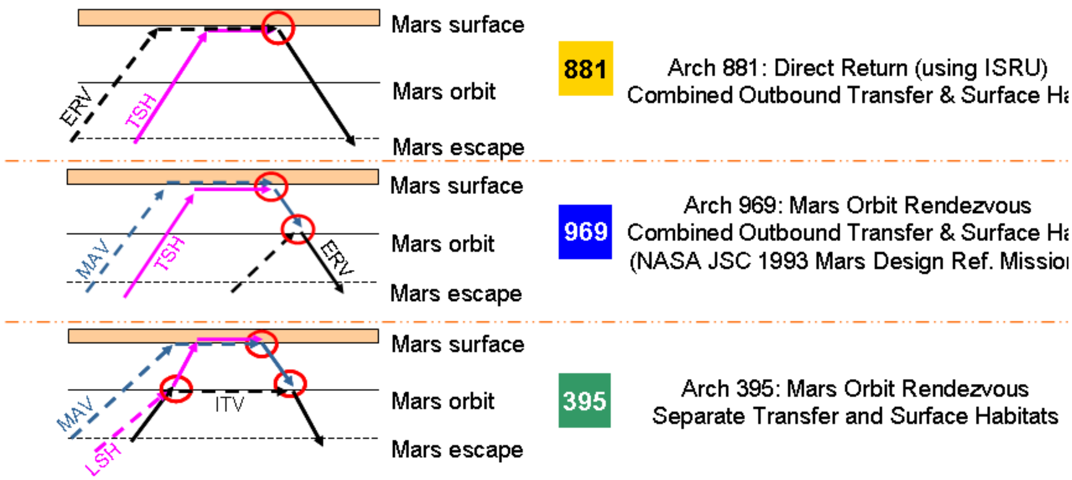

Figure 3. Candidate Mars architectures based on screening criteria. *Solid arrows represent vehicle occupation by crew, broken arrows unmanned vehicle flight. Red circles represent changes of vehicles for crew*[3,10].

Figure 3 provides an overview of the three chosen Mars architectures; the tracking numbers are the OPN-based architecture designations. Architecture 881 is similar to the “Mars Direct” architecture proposed by Zubrin et al.[8,11]. It employs two vehicles: the Transfer & Surface Habitat (TSH) is used by the crew during Earth-Mars cruise and during the Mars surface stay. The Earth Return Vehicle (ERV) is used for transfer from the Martian surface back to Earth, i.e., the crew changes vehicles at the end of the Mars surface stay. Due to the large propulsive requirements for lifting the ERV habitat out of the Martian gravity well, ISPP for the ERV ascent and TEI stage(s) is required for architecture 881. One surface rendezvous is required.

Architecture 969 is very similar to the NASA Mars DRM[5,6]: the TSH transports the crew from Earth to the Martian surface. At the end of the surface stay the crew switches to the Mars ascent vehicle (MAV), which is used for ascent to orbit and rendezvous & docking to the ERV. After the crew has switched to the ERV, they return to Earth. Architecture 969 can be carried out both using ISPP for the MAV ascent stage or not; ISPP is not an enabling factor for this architecture. Two vehicle switches are required: one docking in Mars orbit and one surface rendezvous.

Architecture 395 is a variant of architecture 969: the Interplanetary Transfer Vehicle (ITV) carries the crew to Mars orbit. The crew then switches to the Landing & Surface Habitat (LSH). At the end of the surface stay, the crew uses the MAV to ascend to Mars orbit and rendezvous & dock to the ITV. The ITV then brings the crew back to Earth. Three vehicle switches are required: two dockings in Mars orbit, and one surface rendezvous. Architecture 395 can be carried out both with ISPP and without.

Figure 4 shows the candidate lunar architectures chosen after applying the screening criteria. The architecture numbers are again those generated using the OPN-based model.

Architecture 1 represents a lunar direct return architecture: the crew utilizes the same crew compartment during the entire mission. This architecture is similar to that proposed in the NASA first lunar outpost study of the early 1990s.[12] For long-duration lunar surface mission, the crew would switch to a lunar surface habitat in order to not have to bring a large habitat back to Earth. In the short lunar mission case, architecture 1 does not require any rendezvous or docking. In this architecture, the lunar variant of the CEV would go to the lunar surface.

Architecture 12 is a derivative of architecture 1: it also employs only one crew compartment, but leaves the TEI propulsion stage in lunar orbit instead of binging it to the lunar surface. This enables significant IMLEO savings for the added “cost” of a required rendezvous maneuver in lunar orbit and the development and fabrication of an additional stage. For long-duration lunar surface stays an additional lunar surface habitat would be employed. In architecture 12, the lunar variant of the CEV would also go to the lunar surface.

Architecture 67 is the Lunar Orbit Rendezvous (LOR) architecture used in the Apollo program: the crew is transported to the Moon and back in a reentry capable crew compartment, but switches to a dedicated vehicle for descent to the lunar surface, the surface stay, and ascent to lunar orbit. After rendezvous and docking in lunar orbit, the crew returns to Earth. For long-duration lunar surface stays an additional lunar surface habitat would be employed. One rendezvous & docking is required for return to Earth, and the crew has to switch vehicles twice. Please note: for the analysis presented here it was assumed that the entire crew goes to the lunar surface; in the Apollo program one crewmember stayed in orbit while two other crewmembers went to the lunar surface. In architecture 67, the lunar variant of the CEV would go to lunar orbit.

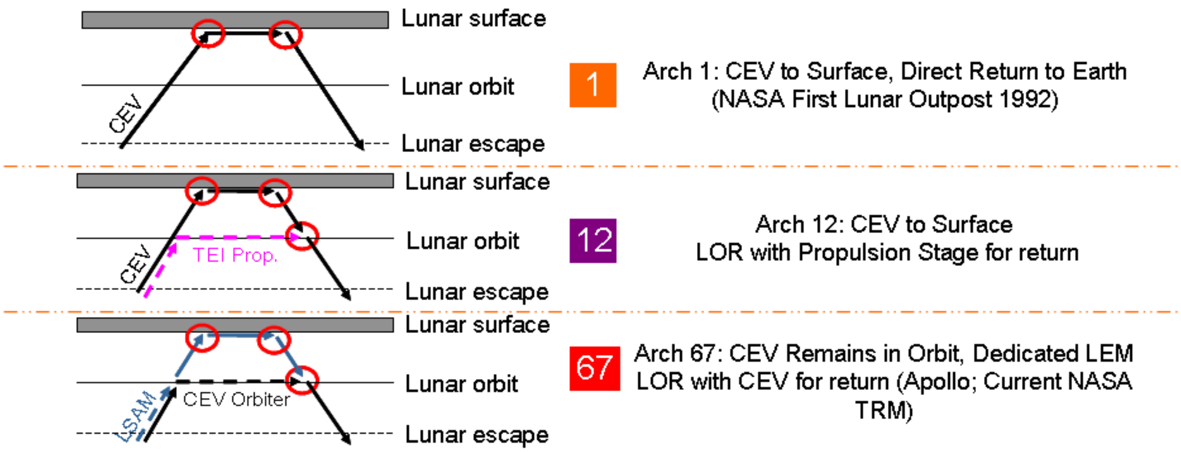

Figure 4. Candidate lunar architectures based on screening criteria. *Solid arrows represent crewed vehicle phases, broken arrows unmanned vehicle phases. Red circles represent crew transfers between vehicles*[3,11].

D. Selection of Lunar and Mars Reference Architectures

Of the candidate lunar and Mars architectures, the following two were selected for an in-depth commonality analysis. The reasons for this selection are presented below. It should be noted, however, that a similar analysis of high-level commonality could and should be performed for all candidate Moon and Mars architecture pairs.

Selected Mars architecture: 969

- Lowest IMLEO of the candidate architectures
- No ISPP required to carry out Mars missions (required for 881), ISPP and In-Situ Resource Utilization (ISRU) could however be introduced to enhance the architecture
- One surface rendezvous required
- Only one rendezvous and docking in Mars orbit required, after the surface stay
- No rendezvous & docking required after aerocapture into Mars orbit (required for 395)

Selected lunar architecture: 1 (lunar direct return)

- Anytime return from any landing site on the lunar surface possible without additional plane changes (plane changes required for 12 and 67 because of orbiting elements)
- Lowest number of mission critical events

- Straightforward operational approach
- No rendezvous & docking required during the mission (required for 12 and 67)
- Only one crew compartment design required (2 for 67 / LOR)

For a more detailed discussion of the benefits of architectures with the CEV to the lunar surface, and of the rationale for lunar direct return, please refer to[13].

IV. Exploration System Commonality Using the Mars-Back Approach

As stated above, the objective of the Mars-Back approach is to employ high-level commonality between and within the lunar and Mars exploration architectures in order to decrease the development time between Moon IOC and Mars IOC without inflicting a significant penalty on lunar exploration system development.

For the sizing of system elements (crew compartments, long-duration habitats, heat shields, Mars aero-shells, and propulsion stages) a parametric approach was used. The element models were based on previous designs[7] or designs studies[14-16,5-6] as well as on parametric subsystem models described in human spacecraft design reference literature[17-21,13]. Table 1, Table 2, Table 3, and Table 4 in the appendix provide an overview of the resulting structural mass fractions and element masses, as well as the specific impulses and delta-v values used for the common system design.

A. Common Vehicle Elements and Vehicle Stacks

Figure 5 provides an overview of the vehicle stacks before departure from Earth orbit. All the vehicle stacks shown in are comprised of a set of common elements. This set consists of:

- An Earth departure stage using LH2 / LOX propellants (shown in red at the bottom of the stacks). The Earth departure stage has a dry mass of about 11 mt, and a maximum wet mass of 112 mt. It can be used with varying initial propellant fillings and so accommodate different mission requirements. The common Earth departure stage is sized by trans-Mars injection (TMI) of the TSH. The Earth departure stage includes up to 5 engines in order to accommodate the loss of one engine during Earth departure burns.
- A core propulsion stage used as the basis for all maneuvers in vicinity of the destination (Moon / Mars). This core stage utilizes LCH4 / LOX propellants and has a maximum wet mass of about 19 mt. The core propulsion stage can also be used both partially and completely filled with propellant.
- A set of four common-bulkhead LCH4 / LOX extension tanks and associated structure (including plumbing, thermal control, and interfaces) to provide more propellant capacity to the core stage. The set of extension tanks are sized by the lunar descent of the crew transportation system, and can be used both completely filled and partially filled.
- A set of XL common bulkhead extension tanks (LCH4 / LOX) and associated structure to provide additional propellant to the core stage for the ERV TEI maneuver, which also sizes the tanks. Both sets of extension tanks (strap-on tanks) are arranged along the circumference of the core stage.
- A short-duration crew compartment that consists of a pressurized capsule of 27 cubic meters capable of hyperbolic entry both from Moon and Mars. In addition, the crew compartment features a small cylindrical element called the Integrated Power Unit (IPU) that contains additional consumables and equipment, and is jettisoned from the capsule before reentry. The capsule and IPU together are the Crew Exploration Vehicle that would initially be used for missions to LEO and to ISS. The only customization of the capsules concerning Moon and Mars missions is that for lunar missions, no docking equipment is required, and hence not installed to save mass. Structural scars for adding in the docking tunnel and ring for the Mars CEV are provided to avoid structural customization of the capsule. The MAV CEV is docked to the ERV before TEI to enable crew transfer to the ERV. This CEV is brought back to Earth and used for direct Earth entry of the crew at the end of the mission.
- An extended power unit that provides additional reactants to the CEV fuel cells housed in the IPU. This power pack is used only for the lunar crew transportation system, and is left on the lunar surface. The extended power unit has a wet mass of approximately 1.6 mt and provides an additional 12 days of full CEV power (7.2 kW).
- A core habitat that has 200 cubic meters of pressurized volume, and can carry enough consumables and equipment for a 810-day stay of up to 5 crew as required for the TSH.

- An inflatable pressurized surface tent for use on the lunar and Mars surface. The surface tent provides an additional 100 cubic meters of pressurized volume and weighs 3 mt including gas storage and stowage structure. It is only baselined for use on the Martian surface; it can, however, also be used on the lunar surface.
- A lunar landing gear and exoskeleton structure to which both the lunar descent propulsion stage and the payload on top of it is connected. The gear and exoskeleton structure accommodates different loading environments during Earth launch and planetary touchdown, and enables varying payload sizes and cg locations on top of the descent stage. The exoskeleton and gear structure is sized by the crewed lunar lander.
- A Mars landing gear and exoskeleton that serves the same function as the lunar landing gear and exoskeleton for the loading environments and payloads of the Mars mission. The Mars landing gear and exoskeleton is sized by the MAV descent stage and would likely be based upon the lunar landing gear and exoskeleton design.
- A common aeroshell that is used for aerocapture and aeroentry at Mars. The aeroshell is not only a passive structure, but includes also an RCS system for active attitude control during cruise and aerocapture / aeroentry. It also has a directional propulsion system for performing midcourse correction maneuvers during the Earth – Mars cruise. The aeroshell has its own photovoltaic power generation subsystem which provides the payload in the aeroshell with electrical energy. Avionics and communications systems are also included in the aeroshell system. The structure of the aeroshell is separated from the payload inside during aeroentry and Mars descent (if applicable). The aeroshell is sized by the ERV.
- A common non-throttleable, restartable, pressure-fed LCH4 / LOX engine design. This engine design is sized by the Mars ascent stage use case.
- A common throttleable, pressure-fed LCH4 / LOX engine design for descent to planetary surfaces. This engine design is sized by the MAV descent use case.

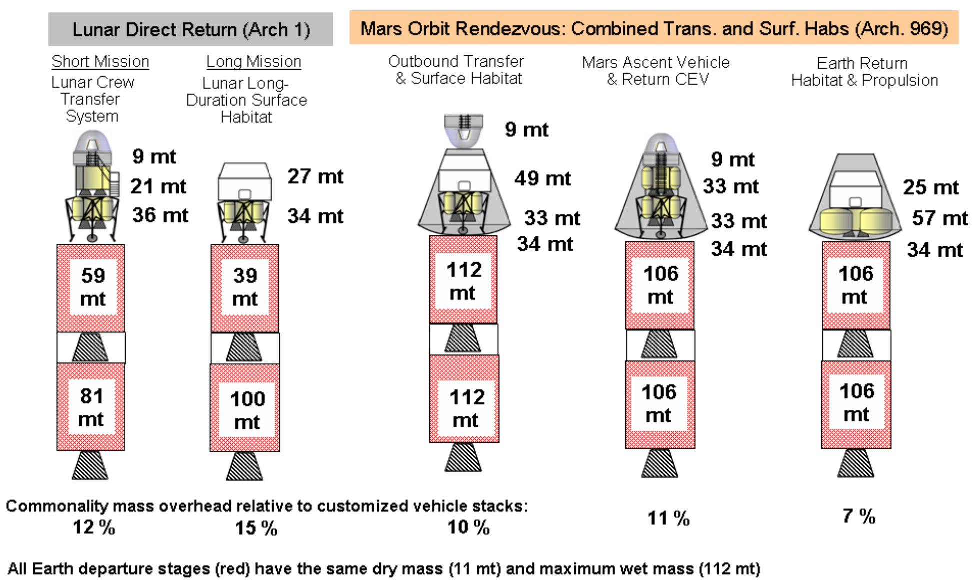

Figure 5. Common vehicle elements and vehicle stacks in LEO. *The numbers represent the element masses in metric tons. The vehicle stacks are shown after launch and assembly in LEO. More detailed description of the stacks and common elements is included in the text.*

Figure 5 shows how these common elements are arranged to form the individual vehicle stacks.

- The lunar crew transportation system stack for short-duration surface missions includes the lunar CEV (no docking equipment), the core LCH4 / LOX propulsion stage with two non-throttleable engines, the extension power pack, a core stage with four regular extension tanks and the lunar landing gear and exoskeleton and two common throttleable engines as descent stage, and two partially filled and sequentially used Earth departure stages.

- The lunar surface habitat stack (used to enable long-duration stays on the lunar surface) consists of the core habitat (outfitted for 5 crew for 180 days), the same descent stage configuration as for the crewed lunar lander, and also two partially filled and sequentially used Earth departure stages.
- The TSH stack features a Mars CEV (delivers the crew to the TSH in LEO, is used for Earth entry in the case of propulsive abort during Earth-Mars cruise), the core habitat (outfitted for 5 crew for 810 days) with stowed tent and surface cargo, the core LCH4 / LOX stage with a set of regular extension tanks, the Mars landing gear and exoskeleton, and four common throttleable engines as the descent stage, the aeroshell, and two completely filled and sequentially used Earth departure stages.
- The MAV consists of the Mars CEV, the core stage with a set of extension tanks and with four non-throttleable, restartable engines serving as the ascent stage, surface cargo, the same descent stage configuration as the TSH, the aeroshell, and two partially filled and sequentially used Earth departure stages.
- The ERV stack consists of the core habitat (outfitted for 5 crew for 200 days plus 1250 crew-days of extra consumables for contingency situations), the core stage with XL extension tanks and four non-throttleable, restartable engines as TEI stage, the aeroshell, and two sequentially used and partially filled Earth departure stages.

Below the individual stacks, the IMLEO overhead is shown compared to a completely customized version of the stack. The IMLEO overhead for the commonality scheme presented here does not exceed 15 % of the customized stack's mass. This commonality penalty is more than balanced by the significant decrease in the number of unique engine, propulsion stage, landing gear, habitat, crew compartment, Earth departure stage and aeroshell designs required; in many cases a reduction by a factor of three or more was achieved.

Please note that, although the lunar descent stages, Mars descent stages, and the Mars ascent stage all have the same maximum propellant volume, these propulsion stages are filled differently for each use case, and therefore also have different masses. Also, in the different use cases the propulsion stages have different numbers of engines and different (or no) landing gear, which leads to differing dry masses.

B. Lunar Launch Strategy

With the Earth departure strategy and the in-space element configurations known, the launch strategy can be derived. Figure 6 provides an overview of the launch strategy for the lunar crew transportation system and the lunar surface habitat.

The baseline launch scenario for the lunar crew transportation system involves three launches: first, the 81 mt Earth departure stage is launched into LEO on a Shuttle-derived in-line HLLV of about 100 mt payload capacity to LEO. Next, the 59 mt Earth departure stage and the descent stage are lifted into LEO on the same launcher, and docked to the first Earth departure stage. The third launch carries the crew in the lunar CEV and the ascent stage and power pack into LEO on a 30 mt class launch vehicle based on a 5-segment SRB with an upper stage (so-called "single stick" launcher). The CEV and ascent stage dock to the two Earth departure stages and the descent stage in LEO, and the result is the stack shown in Fig. 5.

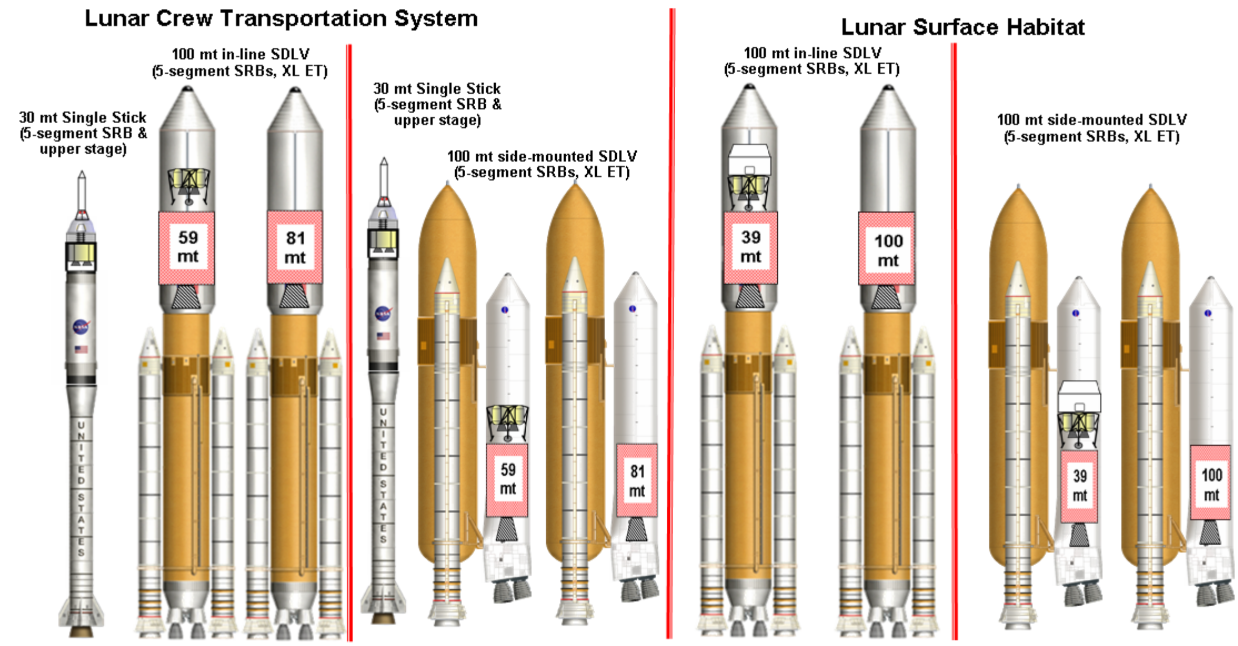

Figure 6. Lunar launch strategy options for crewed lunar lander and lunar surface habitat. *Both in-line and side-mounted Shuttle-derived heavy lift launch vehicles (SDHLLVs) are shown; the system elements were sized to fit into the 7.5 – 30 m payload bay of a side-mounted SDLV[16]. Man-rating of the SDHLLV would enable a lunar crew transportation mission with 2 SDHLLV launches only, similar to the lunar surface habitat launch strategy.*

The launch strategy for the lunar surface habitat involves two launches: the first launch delivers a 100 mt Earth departure stage to LEO on the same 100 mt class HLLV used for the lunar crew transportation system. The second launch carries the lunar surface habitat (core habitat and descent stage) and the second Earth departure stage into LEO. Docking with the first ED stage yields the stack from Fig. 5.

The lunar crew transportation system and lunar surface habitat launch strategies can both also be carried out using a side-mounted Shuttle-derived HLLV in the 100 mt class. The lunar lander propulsion stage design and the core habitat sizing allow for launching the lunar mission hardware inside a 7.5-meter diameter fairing of a side-mounted HLLV.

Also, man-rating of the HLLV enables elimination of the single stick launch for the crew transportation system because the CEV, the ascent stage and the power pack could be launched on the same HLLV that carries the descent stage. This would lead to a similar launch strategy as for the lunar surface habitat. Man-rating of the HLLV could in part be achieved in the same way as for the single stick: by using a launch escape system (LES) tower that accelerates the crew capsule away from the launcher in case of an abort.

C. Mars Launch Strategy

The Mars system architecture described above is used for conjunction class Mars missions with relatively short Earth-Mars and Mars-Earth transit times, and relatively long Mars surface stays. Each of the three stacks required for one Mars mission includes three packages with masses between 100 mt and 125 mt; two of these packages in each stack are Earth departure stages. As assembly and qualification of aerocapture and aeroentry heat-shields in LEO is assumed to be prohibitively complex, an HLLV with a capacity of at least 125 mt to LEO is required.

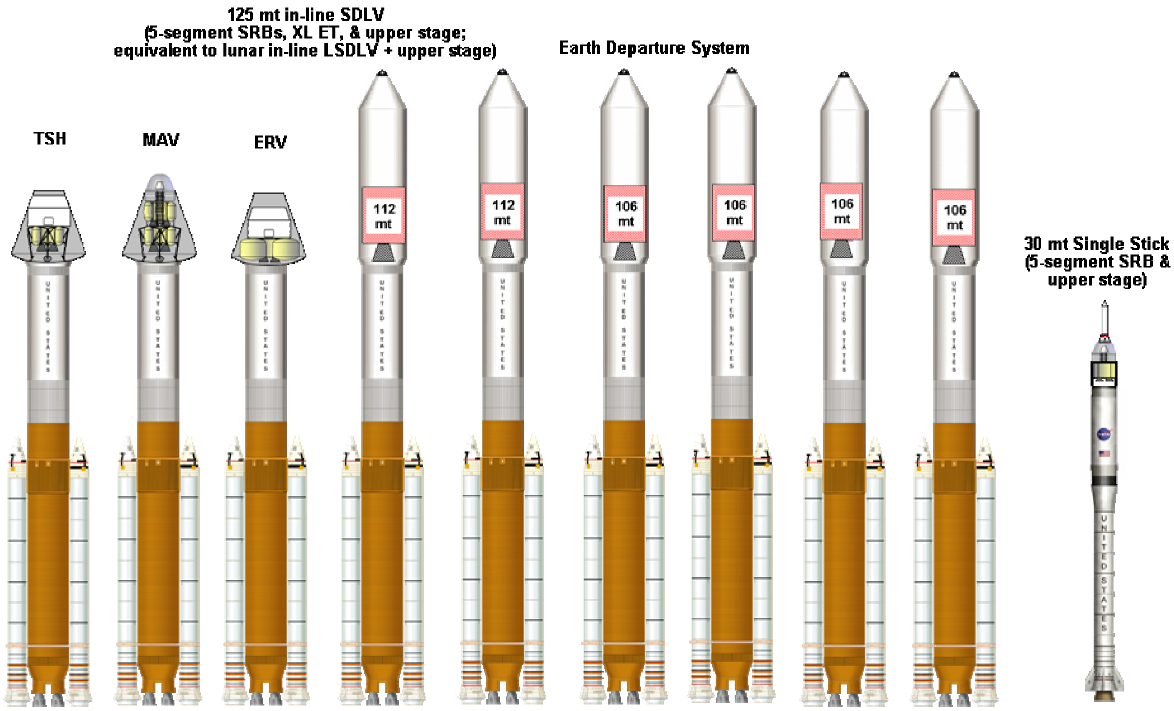

Figure 7. Mars mission launch strategy. *Shown is the steady-state launch strategy for one launch opportunity to Mars. As the unmanned MAV and the ERV for a mission are prepositioned one opportunity before the crew is sent to Mars, the launch manifest on the first opportunity used for manned Mars missions would only include the MAV, the ERV, and four launches of the associated 2 Earth departure stages (red stages) each. The launch manifest shown here therefore represents the TSH and crew for the current Mars mission, and the MAV and ERV for the subsequent mission.*

Figure 7 provides an overview of a Mars mission launch strategy based on a 125 mt class HLLV. This capacity could likely be realized with an in-line Shuttle-derived HLLV using 5-segment SRBs, a core stage based on a stretched version of the external tank, and an upper stage. Ten launches are required altogether to launch the elements required to carry out one conjunction class Mars mission: 3 launches with the HLLV to carry the TSH, the MAV, and the ERV to LEO in their aeroshells. Six HLLV launches are required to bring the associated Earth departure stages to LEO. One CEV launch on a single stick carries the crew to orbit. After being launched, the elements rendezvous and dock to form the stacks in Fig. 5.

As the MAV and ERV for a mission are actually prepositioned at Mars one opportunity before the crew leaves for Mars in the TSH, the launch manifest shown in Fig. 7 represents the hardware that would have to be launched in a steady state with manned Mars missions every opportunity. Although nine HLLV launches may seem difficult, this number is lower than the achieved launch rate of the Space Transportation System when spread out over the Earth-Mars synodic period (9 launches in 2.3 years equals approximately 4 launches per year).

The number of launches in a steady-state Mars exploration program can be reduced by introducing advanced propulsion technologies for Earth departure. Figure 8 shows a Mars steady state launch manifest for Earth departure stages employing nuclear thermal propulsion: the number of Earth departure stages and launches can be reduced from six to three. A similar Earth departure strategy was envisioned for the NASA Mars Design Reference Mission [5-6].

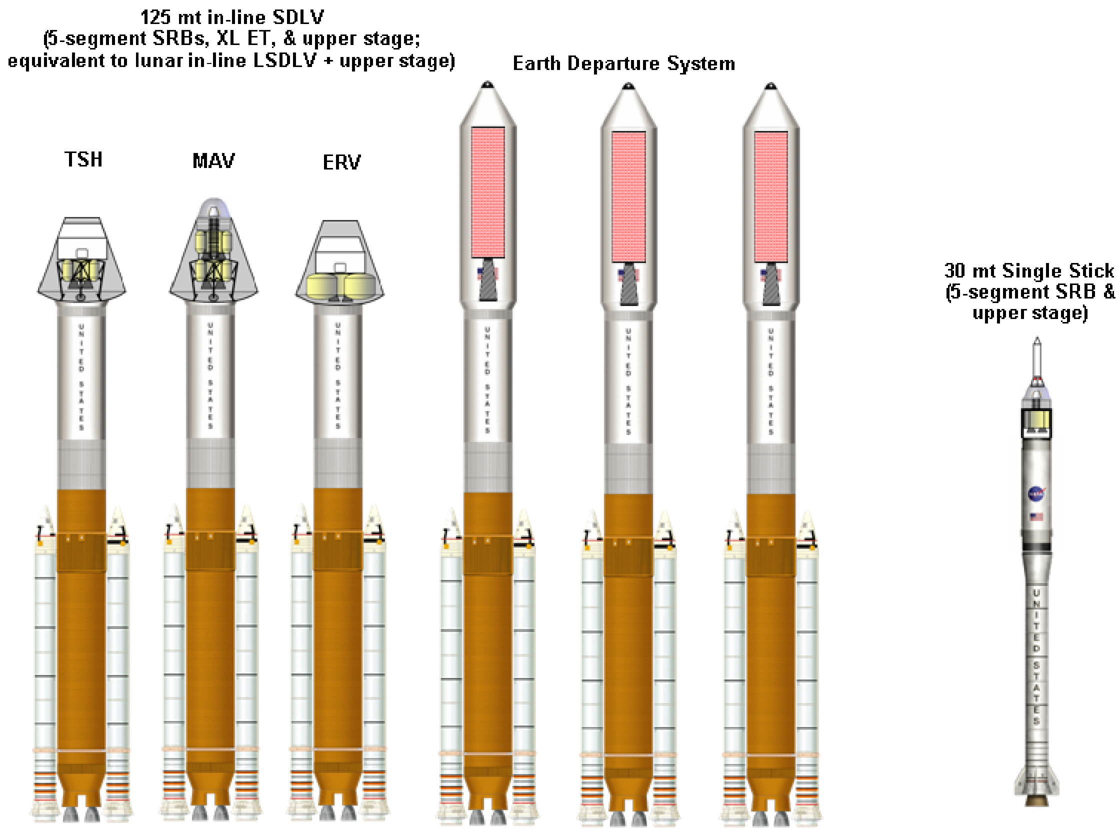

Figure 8. Mars launch strategy for the case of nuclear thermal propulsion (NTP) for Earth departure. *In the baseline case of chemical propulsion for Earth departure (see above), the Earth departure system accounts for 6 out of the 9 HLLV launches for. Using a more efficient NTP system for Earth departure can substantially reduce the number of launches required for Earth departure.*

D. Hardware Development Roadmap

The set of common hardware elements used for the Moon and Mars vehicle stacks presented above is significantly reduced compared to the number of element designs required for customized Moon and Mars exploration systems. This is due to commonality both within and between the Moon and Mars exploration systems. Not all of the common elements need to be developed at the same time: some elements are initially required for Earth orbital missions, some initially for lunar missions, and some only for Mars missions. Figure 9 shows a roadmap for the development of the hardware elements according to exploration phases.

The first phase of exploration involves missions to LEO and to the ISS. For these missions the CEV is required (potentially with a LEO-only heat shield, if development of the ablative heat shield for hyperbolic entry proves to require significant time), and the extended power pack. Also, a propulsion module is needed for orbit change, rendezvous & docking and deorbit maneuvers. By providing this propulsive capability with a separate module, it is ensured that the associated mass does not have to be carried on the lunar and Mars versions of the CEV. As described above in the lunar and Mars launch strategy, the CEV will require its own launch vehicle. For the development strategy presented here, it is assumed that this launch vehicle will be based on the so-called “single stick” configuration utilizing a 5-segment SRB and a LH2 / LOX upper stage. In order to ensure abort capability for the crew at all times during launch, an LES system needs to be developed than can pull the capsule away from the launcher in case of an emergency. This would enable an abort even before lift-off. Altogether, five distinct high-level elements would have to be developed for the first exploration phase, slated to begin soon after STS retirement in 2010. It should be noted that the development plan for the first exploration phase would look much the same for a development strategy with and without high-level commonality, because all the elements developed for this phase are essential for human exploration.

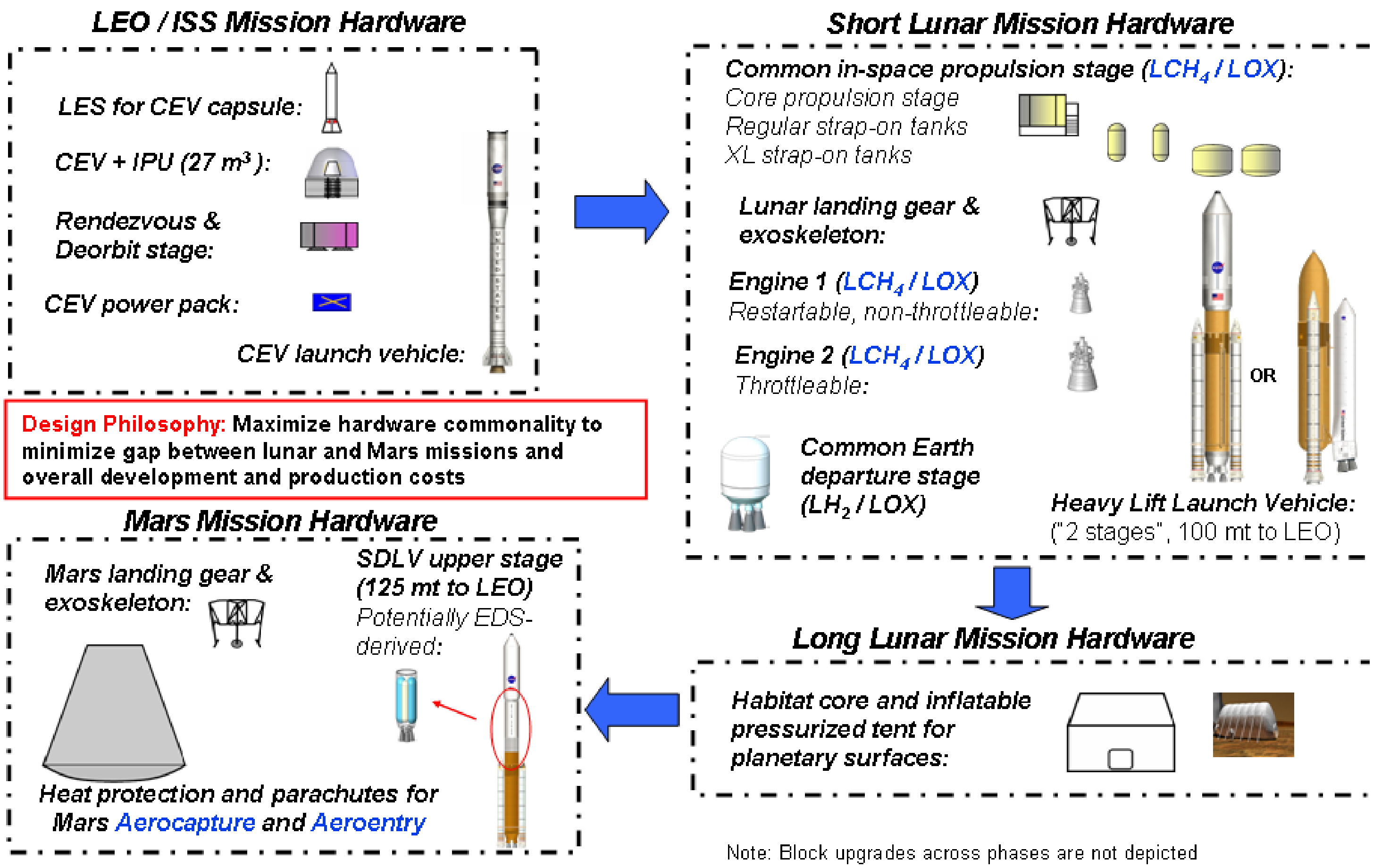

Figure 9 Hardware development roadmap for human Moon and Mars exploration systems employing extensive high-level commonality *The hardware development roadmap is organized in four distinct phases, corresponding to key objectives of the VSE* [1]*: (1) development of the CEV and associated propulsion, equipment, and launch vehicle for LEO and ISS missions; (2) development for short lunar missions (up to 7 days duration); (3) long lunar surface missions (months); (4) Manned Mars missions (conjunction class missions assumed). Please note: block upgrades (for example, upgrading the CEV heat shield from LEO entry to hyperbolic entry capability) are not shown in the development roadmap.*

The second phase of exploration adds the capability for manned missions to the Moon with surface stays of up to 7 days. For this exploration capability, a number of elements need to be developed (see Fig. 9):

- The Shuttle-derived HLLV for launching heavy payloads such as the Earth departure stages, the lunar descent stage, and (later on) the lunar surface habitat. The SDHLLV could be either in-line or side-mounted depending on the desired additional use cases and the development schedule: Mars missions will likely require an in-line SDHLLV because of the large payloads and the large aeroshell diameters for aerocapture; an in-line launch vehicle would therefore facilitate adapting the HLLV design for Mars missions. A side-mounted SDHLLV, however, is likely to be less expensive in development cost than an in-line vehicle, and could therefore be available earlier and potentially be used for ISS assembly. Also, a side-mounted HLLV requires fewer changes to existing ground infrastructure at KSC.
- The core stage with its two sets of extension tanks. The set of XL extension tanks would strictly not have to be developed before the Mars mission phase; it is assumed, however, that it would be more effective to develop all propellant related hardware with the same development team in the same project. Also, the availability of larger extension tanks might enable alternate missions in the Earth-Moon system without utilizing an Earth departure stage.
- Both the non-throttleable, restartable ascent and in-space engine, and the slightly higher thrust, throttleable descent engine need to be developed for short lunar missions; both engines are pressure-fed

LCH4 / LOX engines. It is conceivable that the descent engine design could also be used for the ascent and in-space engine in case it is desirable to avoid engine customization.

- The landing gear and exoskeleton for delivering payloads to the lunar surface. For the design of this element, top-level lunar surface habitat mass and cg properties also have to be taken into account.
- The common Earth departure stage utilized for TLI, LOI, and TMI.

Although the design elements will require significant development resources, it should be noted that a completely customized system design for short lunar missions would likely double the number of unique element designs required; high-level commonality within the lunar architecture is already beneficial at this stage.

The third phase of human exploration involves long-stay lunar surface missions in preparation for manned Mars missions. For this phase, only a small number of elements is required:

- The core habitat design, which is capable of housing up to 5 crew for up to 810 days both in-space and on planetary surfaces. This requires a designs that is independent of the gravity vector, and also largely independent of the exterior atmospheric and radiation environment. High-level analysis of long-term habitation technologies suggests that this is achievable[16-17, 19].
- The inflatable pressurized extension tent for planetary surfaces. This tent is strictly not required before the Mars mission phase; utilizing the tent to provide additional volume on the lunar surface could, however, provide both valuable experience with the hardware before sending it to Mars, and enhance the quality of lunar surface simulations of Mars missions.

It should be noted that with the hardware developed up to the third exploration phase, alternate missions to Near Earth Asteroids (NEAs), Mars flybys, and missions to the Martian moons Phobos and Deimos (employing propulsive capture into Mars orbit and subsequent aerobraking) are possible.

The fourth phase of exploration is dedicated to human exploration of the Martian surface. For this phase, the following elements are required in addition to the ones developed for previous phases:

- The common aeroshell utilized for the TSH, the MAV, and the ERV.
- The Mars landing gear and exoskeleton for surface access of the TSH and the MAV
- The HLLV upper stage to increase the SDHLLV capacity to the 125 mt to LEO required for Mars missions. If the original HLLV was a side-mounted vehicle, then the entire in-line HLLV plus upper stage need to be developed. This represents a key trade in terms of HLLV development: side-mounted HLLV for lunar missions enable faster and cheaper HLLV development for lunar missions and potentially alternate use scenarios such as ISS assembly, but results in higher overall lifecycle HLLV development cost because of the renewed major development effort for the in-line HLLV plus upper stage for Mars. An in-line HLLV for lunar missions will likely increase HLLV development cost and time for lunar missions, but lead to a reduced overall lifecycle HLLV cost.

In the fourth development phase, the benefit of the effect of the Mars-Back approach is especially pronounced: a customized Mars system design would require new habitat, propulsion stage, engine, and Earth departure stage designs, thereby necessitating a renewed major resource commitment and significant development time between lunar and Mars missions.

V. Conclusions, Recommendations, Future Work

The analysis of high-level element commonality between human lunar and Mars exploration systems presented above encompassed all mission phases: launch, in-space transportation, and surface operations. Although the analysis was based on parametric models, considerations for the actual structural and geometrical design, as well as for the powered mission phases, and crewed operations were taken into account in the sizing of architecture elements. Based on the analysis results, a number of conclusions can be drawn and recommendations derived:

- High-level element commonality within and between human lunar and Mars exploration architectures appears feasible from an operational point of view. Additional work regarding the feasibility of high-level element commonality strategy as presented here should be focused on more extensive (higher resolution) element design, which was beyond the scope of this initial study.
- In general, Moon and Mars architecture planning should be focused on the development of capabilities rather than the development of solutions for one design point. These include (among others) propulsive and habitation (long-duration, short-duration) capabilities.
- Options for high-level element commonality are not unique to the architecture pair of lunar direct return and Mars architecture 969; high-level commonality options for engines, propulsion stages, tanks, habitats, crew compartments, etc., potentially exist for any architecture pair and within any lunar and Mars architecture chosen for the VSE; these options need to be analyzed and the ensuing benefits in

development cost weighed against any penalties in recurring cost and the resultant development scheduling. Analyzing these commonality options is crucial for avoiding customized lunar and Mars exploration system design that might lead to excessive life-cycle cost and necessitate a prohibitively long Moon – Mars development gap. An example for commonality considerations within a lunar orbit rendezvous architecture could be the reuse of the crewed descent stage design for delivering a lunar surface habitat and other surface cargo.

- Commonality between Moon and Mars exploration elements provides valuable operating experience for the common elements. As the Mars missions in most cases imposes the more strenuous requirements (especially concerning equipment lifetime and operating time), this operational experience can potentially substitute dedicated test flights of Mars mission hardware.
- Mass overheads in IMLEO introduced by the commonality scheme compared to a customized system design were 15 % or less. Although IMLEO is often used as a proximate metric for launch cost, the real metric for evaluating the commonality impact should be based on the implications for the launch strategy itself. For the system designs presented here, no impact due to commonality overheads is expected for the Mars launch strategy and the lunar surface habitat launch. For the lunar crew transportation system, taking away the commonality overhead might lead to a launch strategy with 2 launches: one single stick (30 mt to LEO) and one Saturn V class SDHLLV launch (135mt to LEO). Although this would facilitate crewed lunar mission operations, it would also impose the additional development cost for the Saturn V class SDHLLV on the short lunar mission development phase.
- Man-rating of the HLLV potentially enables a simplification of the launch strategy and / or a reduction in the number of launches required.
- For the commonality of lunar surface, Mars surface, and in-space long-duration habitats, technologies using gravity-independent effects are required as enabling factors.
- Before developing highly integrated designs of combined crew compartments and propulsion stages (as for example was the case with the Apollo LM ascent stage), options for other use cases for the propulsion stage as well as for the crew compartment should be carefully investigated. Highly integrated designs reduce options for commonality.
- Choosing the same propellant combination and propellant-feed type (e.g., pump-fed, pressure-fed) for all ascent and in-space maneuvers as well as the same propellant and feed type for all descent maneuvers might enable significant engine commonality, and potentially propulsion stage commonality (modular propulsion stages can accommodate different propellant volumes[22]). Commonality between the two classes (ascent; in-space and descent) should also be investigated.

The conclusions and recommendations provided here are based on work that is still in progress: future work will be focused on validation of the feasibility of high-level commonality in more detailed system and subsystem models, as well as on the derivation of commonality options for other lunar and Mars architecture pairs. The impact of additional technology options such as ISRU / ISPP on the lunar and Mars surface and advanced propulsion will also be evaluated.

Appendix

Table 1. Reference delta-v values for Moon and Mars mission maneuvers. *Delta-v values represent ideal delta-V including gravity and (in the case of Mars ascent) drag losses; ideal delta-v is used for propulsion stage sizing. Delta-v values based on [15,21] plus margin.*

Maneuver	Delta-v [m/s]
Lunar ascent & trans-Earth injection (TEI) from LLO	2933
Lunar descent from LLO	2126
Lunar orbit insertion (LOI)	900
Trans-lunar injection (TLI)	3180
Trans-Earth injection from LMO	2805
Ascent to low Mars orbit (LMO) & rendezvous	4080
Mars descent	1224
Trans-Mars injection (TMI) from LEO	4080

Table 2. Reference masses and durations for lunar and Mars exploration system habitable elements, surface payloads and sample masses. *Habitable elements & associated equipment serve as additional payloads for propulsion stages payloads; additional lunar surface payload mass (up to 30 mt) could be provided with unmanned cargo missions to the lunar surface.*

Habitat / crew compartment, payload	Mass [kg], duration
Lunar CEV mass (excluding crew) including capsule + IPU	7757
Lunar CEV extended power unit mass (left on lunar surface)	1622
Mars CEV mass (excluding crew)	8027
Lunar surface habitat mass	26763
Lunar surface habitat duration	5 crew for 180 days
Mars transfer & surface habitat (including inflatable surface tent) mass	42777
Mars TSH duration	5 crew for 810 days
Earth return vehicle habitat mass (including contingency consumables)	25375
Additional ERV contingency consumables, jettisoned before TEI	3600
ERV duration, including 250 days of contingency duration	5 crew for 450 days
Sample mass brought back from lunar surface	100
Sample mass brought back from Mars surface	100
Payload mass to lunar surface brought with crew	500
Mars surface payload mass on MAV	7000
Mars surface payload mass on TSH	5000

Table 3. Structural mass fractions of common LCH4 / LOX propulsion stage design variants. *Structural mass fraction is defined here as the propulsion stage dry mass (including engines, landing gear) divided by the propellant mass.*

Propulsion stage	Structural mass fraction [-]
Lunar ascent & TEI stage (common core stage+2 common ascent engines)	0.2
Lunar descent stage (common core stage + common strap-on tanks + lunar landing gear & exoskeleton + 2 common descent engines)	0.23
Mars ascent stage (common core stage + strap-on tanks+ 4 common ascent engines)	0.15
Mars descent stage (common core stage + common strap-on tanks + Mars landing gear & exoskeleton + 4 common descent engines)	0.38
ERV TEI stage (common core stage + XL strap-on tanks + 4 ascent engines)	0.13
Common Earth departure stage	0.11

Table 4. Specific impulse values for LCH4 / LOX and LH2 / LOX propulsion used in common system design. *LCH4 / LOX propulsion chosen for Mars because of low boil-off, comparatively high specific impulse, Mars ISPP suitability. Pressure-fed system chosen for safety reasons*[20].

Propellant combination	Specific impulse [s]
Pressure-fed LCH4 / LOX	362 s
Pump-fed LH2 / LOX for Earth departure propulsion	462 s

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