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Solar System Longboats: A Holistic and Robust Mars Exploration Architecture Design Study

Donald C. Barker[1]
Mars Advanced Exploration and Development, Inc., Houston, TX, 77258, USA

A conceptual mission with a two vehicle architecture for the exploration and settlement of Mars and planetary surfaces is presented based on an integrated mission, environment and systems design approach. The first vehicle, the Surface-to-Surface Return Vehicle (SSRV), is designed to transfer three crewmembers between the surface of Mars and the surface of the Earth. The second vehicle, called the Planetary Habitat Vehicle (PHV), is a three deck, six-person habitat designed to support permanent Mars surface habitation and operations. The mission design initially includes four launches at each conjunction class alignment opportunity. Two SSRV and two PHV vehicles are launched into Earth orbit, and then dock as pairs during the outbound flight to Mars. During each piloted flight, a six person crew will transit, between 180 and 220 days, and land on Mars within a PHV. An autonomous SSRV pair follows and lands sequentially. Previous, autonomously landed PHV and SSRV pairs arrive one conjunction opportunity prior to first crew launch in order to provide redundancy and establish a ready return capability. Following adequate infrastructure emplacement, later, round trip transfers to an established base could be completed using the dual SSRV configuration alone. The SSRV and PHV have an estimated Mars landing mass of roughly 31 tons. Creative sequencing, transfer and jettisoning of vehicle mass prior to orbit insertions effectively reduce the shielding mass required during atmospheric entry. Here, the jettisoning of in-flight water based radiation shielding prior to Mars arrival reduces structural mass and shielding requirements. Complimentary design features that also minimize vehicle structural mass during the entry phase include both aero-braking and impulsive orbit entry assist; both of which help to minimize absolute entry velocities and reduce the overall shield mass. This design requires the use of generally larger upfront launch masses (e.g., propellant and water) to provide a tradeoff for enhancing mission safety and reducing shielding mass. The overall architecture draws from human behavior and operations driven requirements. Two environmental drivers, planetary dust mitigation and radiation protection also predominate design rational. A HZETRN 2005 transport code radiation analysis was used to ascertain the radiation protection characteristics of water bearing structures. The shielding design includes water shell structures that encapsulate crewmembers during various mission phases and modeling demonstrated the intent of reducing expected radiation dose rates by one third through these structures alone. Dust mitigation is addressed by a novel integrated airlock and suit compartment that maximizes operations efficiencies, while minimizing dust impingement and consumable waste. Other major operational and consumable usage design drivers include redundancy, standardization, and in-situ water reclamation. Two apriori capabilities of heavy-lift launch and nuclear electric power production directly engender a robust, adaptable and long-lived system design. Ultimately, each module is designed to adapt, function and survive the combined environments in which they operate; a design as much about philosophy as it is about vision.

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[1] President, P.O. Box 58915 Houston, TX. 77258, AIAA Member.

1. Introduction

A viable, safe, self-sufficient, permanent and growth oriented Mars architecture design study is presented. The associated architecture outlines the framework under which specific designs have been conceptualized to address and encompass the unique operations and environments associated with each mission phase. Designing vehicles without proper environmental and operational consideration and integration would be like building an Inuit igloo in the Sahara desert; therefore, this work emphasizes design requirements based on the integration of several key environmental and operational aspects. These include radiation environments, surface dust, the transition from space flight to surface installation and the redistribution of non-structural consumable masses based on the acquisition and use of indigenous resources. These and other drivers directly involve multiple systems that affect architectural designs. The goal of this design is to provide appropriate vehicle sizing and mass estimates for an accurate model of spacecraft structures.

The overall mission of this architecture is to land large structures safely and establish a growing colony on the surface of Mars. Once humans occupy the surface, the primary operational goal is then to locate, secure and initiate near surface water reclamation, processing and storage. Water or its components serve the widest range of basic needs (e.g., propellant, atmosphere, radiation protection, and cellular hydration) that humans require and in this author's opinion, is the single most important goal if permanent habitation of the planet is to occur. Basic scientific research is the second goal (i.e., establishment of a ground-truth geochronology and geo-evolution). Should humans themselves not venture to the planet, it is likely that fewer short-term advances in basic understanding will occur. A tertiary goal is to establish a growing enclave of humanity on the closest terrestrially related planetary surface (i.e., redistributing some of the eggs from the terrestrial basket).

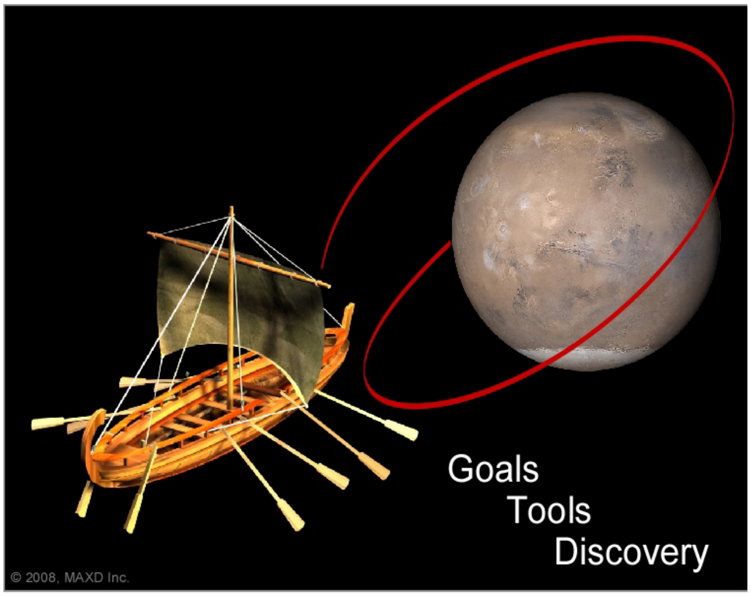

Figure 1. Needed: A long boat for Mars.

In order to carry out this goal in a sustained, efficient and safe manner, human space exploration needs the equivalent of a 21st century Conestoga wagon or Longboat (Figure 1). Derivation of an appropriate architecture and its ultimate fruition should incorporate all relevant historical lessons. For example, inspiration and guidance for the concepts within this paper were drawn from Fridjof Nansen's scientifically successful three-year ice-locked voyage across the Arctic North Pole in 1893; and how his ship the Fram, and her intrepid crew endured the severest trials that nature had to impose.[1]

In general, this work primarily presents high-level systems and operations perspective, the exception being a few cases where novel enabling systems and designs are outlined in sufficient detail in order to enhance the connection between design and function. Several key design philosophies have also been incorporated including what this author calls the 'multiple-use methodology' whereby all objects or pieces of hardware are designed to serve multiple integrated purposes if at all possible. This concept is slightly different from redundancy where either copies or different hardware components support the same purpose or function. The concept of multiple-use methodology and redundancy is increasingly coordinated by another design point that requires the standardization of systems and components that have similar functions.

This design was conceived to address the revived space exploration initiative that was spawned at the highest levels of national government in early 2004. It is the belief of this author that a disproportionate amount of short sightedness related to the subsequent rendering and espousing of exploration architectures has and continues to occur with respect to this belated focusing of the nation's space exploration goals. Basic psychology teaches that a specific long-term goal cannot be sustained or efficiently met without a concise sequence of enabling and short-term attainable goals, all of which directly feed forward towards the longer-term goal. Without such a structure, any venture is apt to fail. The NASA architecture-of-the-moment seems simply to be Apollo revisited given 40-odd years of technological upgrades and enhancements. As an answer to the proposed long-term exploration initiatives, it provides little or even diminishing correlation to vehicles and operations that would actually be needed to go to Mars. Earth is a more suitable and less expensive analogue for the majority of comparative environments, research and operations. The saving grace of this critique is that the current program has touted the desire to, as much as possible, feed forward design and production investments as the architecture evolves in order to buy down Mars mission risks; a tenuous promise given another 20 plus years of politics and administration changes. In a similar light, we applaud the programs championing of a shuttle derived heavy-lift capability as a means to control costs by building on systems already produced and understood; an asset directly needed to initiate, implement and invigorate all future space exploration endeavors. Heavy lift is a launch asset required to support the primary mission design proposed in this document.

Finally, the motto of this exploration design is "Build for Mars, fly at the Moon". Therein, the vehicles presented herein would perform well in 1/6th lunar gravity with little or no modifications since they have already been directly designed to work and perform in the extremes range of micro gravity to the 0.38 g's at the surface of Mars. Understood benefits to testing include the relative proximity of the moon, where logistical support and quick return options insure a safe and successful systems and architecture shakedown.

2. Mission Design and Overview

The efforts required to initiate the first human missions to Mars are expansive, and yet in all respects completely manageable in the near term. This venture, as with the integration of any transportation system throughout history or any previous exploration endeavor exhibits risks that must be, to the best of our ability, identified, accepted and mitigated. This drive to push the boundaries of experience and knowledge is exemplified by those such as Christopher Columbus, who pursued a dream to find a better trading route to the Far East or the audacious circumnavigation of the globe by Ferdinand Magellan. In keeping aligned with the design philosophies forwarded by this author, and others, no completely novel or unconceived human space flight technologies (e.g., ion or nuclear propulsion) are required or will be needed in order to successfully implement this endeavor within a decade from the committed allotment of expenditures; Columbus alas did not await the development of the steamship.

Critical life support and propulsion systems need only be as efficient as current standards dictate given this architectures acquisition and use of insitu consumables, system redundancy, reparability and adequate short-term reserves that are accounted for and transported in hand. Life-support systems, given adequate reserves, can tolerate a certain amount of leakiness and waste and missions should not be delayed until fully regenerative closed-loop systems are developed. These advanced systems can be included in follow on vehicles as such systems come on line. Replacements can even be transported to refurbish older habitats given a common interface standard. What is required, though, is a committed, near-term, research and development program that focuses on enhancing the efficiency, robustness and safety of existing technologies and capabilities. This architecture controls the leaky systems concept by always providing a fully redundant system nearby and by rapidly establishing surface resource reclamation and storage in order to provide immediate local reserves to supply such systems. The concept is more akin to general submarine design and is supported from current space flight operations given that technologies have surpassed original leakage design requirements (i.e., the ISS has performed better than the required with a per module leakage rate of 83 kg per year).[2]

Another way of looking at this methodology and design is that long duration supplies are distributed across multiple nearby vehicles resulting in an increase in useful mission mass per vehicle that would be otherwise removed should a single vehicle be required to provide supplies for the entire duration of the mission. Therefore, the design becomes oversupplied in the context of the shorter, individual segments of the overall mission. And finally, the gap in the transport of long-term consumables is compensated for by their insitu reclamation on the surface of Mars. Again, one of the most important facets is that current technologies should be enhanced to be as robust, reliable and self sufficient as possible, including establishing a system interface standard and forward looking methodologies whereby system upgrades can be incorporated and delivered to Mars as they are developed (i.e., roughly every two years). Finally, all enhancements need to parallel ongoing environmental characterization in direct support of long duration trans-terrestrial habitation.

Overall, the mission architecture presented here roughly parallels the findings of the Exploration Systems Architecture Study;[3] yet, in a broader scope it draws from other designs including the Mars Direct and its derivative plans,[4,5] the NASA Design Reference Mission[6] and NASA Reference Mission V3.0.[7] The concept forms a optomized blend of Mars exploration architectural elements and mission spanning trades in order to enable a long duration space exploration design.[8,9,10] This design then quickly departs from these previous concepts most notably in that a set of vehicles called a "Launch Group" are sent to a single location on the planet at each orbital alignment opportunity. The overall premise of this mission design is based on establishing a permanent, self sufficient and rapidly growing base using two specific, long life and partially reusable vehicles: the Surface-to-Surface Return Vehicle (SSRV) and the Planetary Habitat Vehicle (PHV). Figure 2 provides a schematic outline for the proposed flight schedule. Table 1, below, further provides a breakout of the associated vehicle masses and propellant requirements for the first few missions as discussed throughout the body of this document. The table displays this

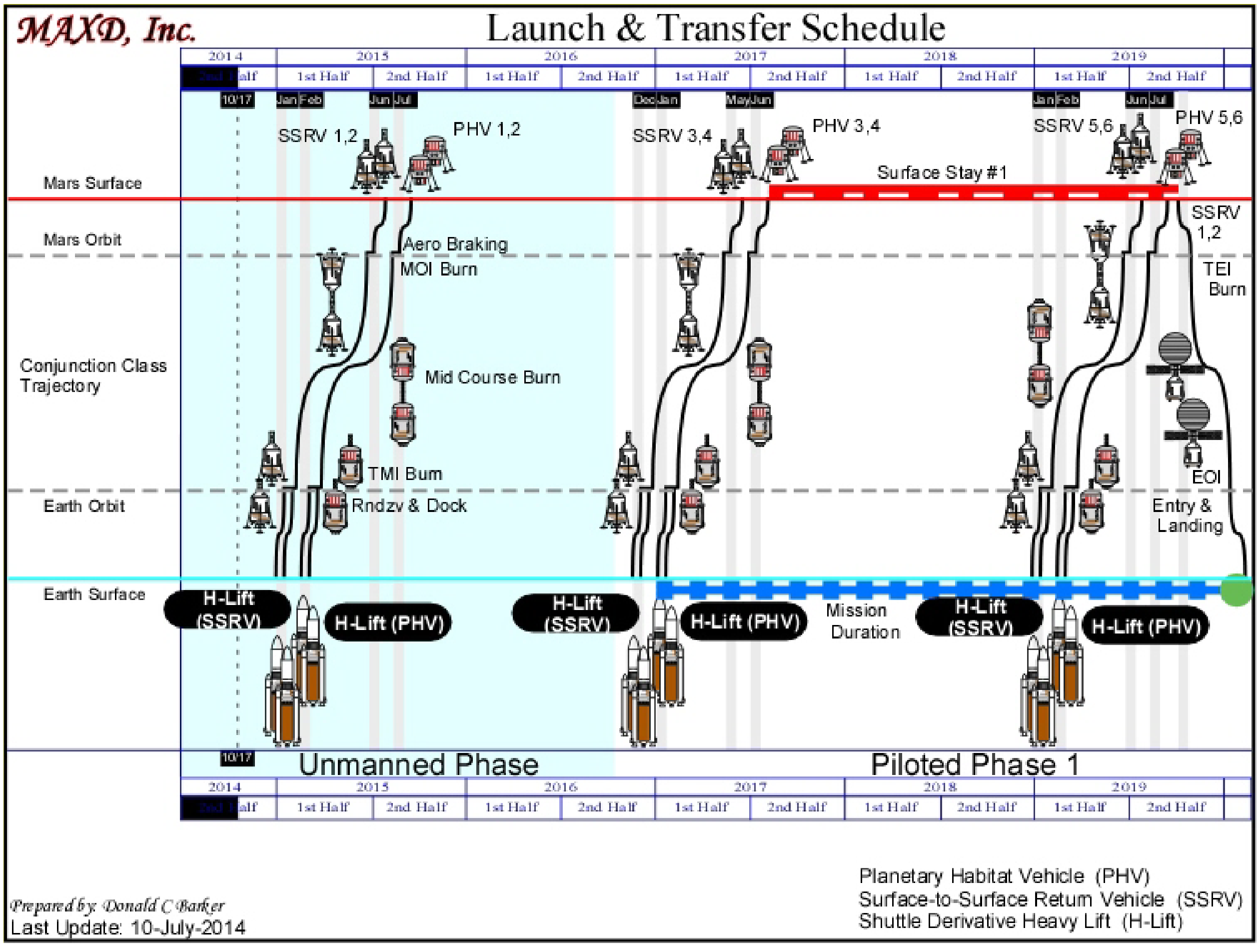

Figure 2. Three year mission sequence opportunity overview and timeline (representative example).

architecture's calculated working masses for each vehicle at specific phases during the mission using average published delta-V values.[11,12,4,5] The top sections detail the breakout and masses for the following phases: SSRV Mars entry and landing, SSRV at Mars launch and both the piloted and unpiloted PHV at Mars landing. The lower block provides a breakout of the mass just prior to the Trans-Mars Insertion (TMI) burn as required to leave Earth orbit (i.e., equivalent to the mass a heavy-lift vehicle would need to place in Earth orbit otherwise termed the initial mass in low Earth orbit (IMLEO)). Therein, the remainder of this work only addresses further details beyond initial mass estimates of the life support, power, thermal, data and communications systems where they directly influence the novelty of this design, and therefore assume that current functional systems will fulfill requirements given the masses already provided.

The Launch Group, as mentioned, is comprised of a pair of SSRVs or a pair of PHVs, instead of the single habitat and single crew return vehicle as outlined in other plans. This design again departs from all previous space flight programs and paradigms in that during Earth return, the crew of six is broken into two three person crews aboard each SSRV. Therefore, it is believed that this design affords greater overall redundancy while addressing a more productively reprioritized inclusion of transient vehicle mass resulting in greater mission flexibility and safety. The flight schedule as outlined would continue until at least a substantial or predefined base infrastructure had been emplaced. Note that three launch opportunities initially defined in this architecture would emplace six habitat modules capable of supporting a crew size of up to 36. This allows program designers the unique downstream flexibility of flying modified SSRVs, instead of PHVs, as the primary crew-transfer vehicle or the ability to begin establishing completely new bases at different locations.

Increasing the number of vehicles produced and employed during any given opportunity positively affects a number of mission architectural and development factors including: enhanced safety through overall system

Table 1 (Part A and Part B). Vehicle mass line-item breakouts.

SSRV (Unpiloted Mars Landing)		SSRV (Piloted Mars Launch)		PHV (Unpiloted Mars Landing)		PHV (Piloted Mars Landing)	
Structure/System/Consumable	Mass (kg)	Structure/System/Consumable	Mass (kg)	Structure/System/Consumable	Mass (kg)	Structure/System/Consumable	Mass (kg)
CCEC Structure Mass w/out Earth EDL Heatshield	2300	CCEC Structure Mass w/out Earth EDL Heatshield	2300	PHV Structure Mass w/out Mars EDL Heat Shield	4500	PHV Structure Mass w/out Mars EDL Heat Shield	4500
TVAL Structure	1000	TVAL Structure	1000	RCS System	500	RCS System	500
Inflatable for TVAL - 6 meter diameter	800	Inflatable for TVAL - 6 meter diameter	800	Thermal System	412.5	Thermal System	412.5
ECLSS: includes hardware dry mass	3030	ECLSS: includes hardware dry mass and 220 days O2 at 0.85 kg/CM-d and	3591	ECLSS: includes hardware dry mass	3500	ECLSS: includes hardware dry mass and 220 days O2 at 0.85 kg/CM-d	4618
SSRV Solar Arrays	1000	SSRV Solar Arrays	1000	Mars Parachute Entry System	900	Mars Parachute Entry System	900
Thermal System	357	Thermal System	357	Comm and Control	200	Comm and Control	200
Furnishings, Exercise, Galley	375	Furnishings, Exercise, Galley	375	Furnishings, Exercise, Galley	1000	Furnishings, Exercise, Galley	1000
RCS System	450	RCS System	450	Food: 500 days at 1.36 kg/CM-d	4080	Food: 500 days at 1.36 kg/CM-d	4080
Earth Parachute Entry System	635	Earth Parachute Entry System	635	Clothing: 200 days at 1.7 kg/CM-d	2040	Clothing: 200 days at 1.7 kg/CM-d	2040
Comm and Control	100	Comm and Control	100	Other (Crew Accessories, Consumables, Health Care)	1000	Other (Crew Accessories, Consumables, Health Care)	1000
Food: Reserves at 1.36 kg/CM-d	224.4	Mars Surface Suits - 3	300	Field Science Equipment	700	Mars Surface Suits - 6	600
Mars Parachute Entry System	700	Food: 220 days at 1.36 kg/CM-d	897.6	Lab Equipment	600	Crewmembers - 6	660
200 kWe Micro-Size Nuclear Reactor or Solar Array	3500	Clothing: 220 days at 1.7 kg/CM-d	1122	EVA Spares and Other	1000	EVA Spares and Other	200
ISPP Plant	453	Other (Crew Accessories, Consumables, Health Care)	500	Spares and Growth	3269	Spares and Growth	3313
H2 Feedstock	5715	Crewmembers - 3	330				
Reactor Transporter or Rover/Truck	500	Return Science	500				
Spares and Growth	2391	Spares and Growth	2391				
CCEC Earth EDL Heat Shield	781.91	CCEC Earth EDL Heat Shield	781.91				
SSRV Total Dry Mass w/out Ascent/Earth Return or Descent/Landing Propulsion Structures (GRDAC)	24312	SSRV Total Dry Mass w/out Ascent/Earth Return Propulsion Structure (GRDAC)	17431	PHV Total Dry Mass w/out Descent and Landing Propulsion Structures (GRDC)	23702	PHV Total Dry Mass w/out Descent/Landing Propulsion Structures (GRDC)	24024
Descent/Landing Propulsion Structure (GRDAC) - dry	4862.5			Descent/Landing Propulsion Structure (GRDC) - dry	3555.2	Descent/Landing Propulsion Structure (GRDC) - dry	3603.5
Ascent/Earth Return Propulsion Structure (GRDAC) - dry	2188.1	Ascent/Earth Return Propulsion Structure (GRDAC) - dry	2188.1				
Return Propulsion and Descent/Landing Propulsion Structures (GRDAC)	31363	SSRV Total Dry Mass with Ascent/Earth Return Propulsion Structure (GRDAC)	19619	PHV Total Dry Mass with Descent/Landing Propulsion Structures (GRDC)	27257	PHV Total Dry Mass with Descent/Landing Propulsion Structures (GRDC)	27627
		SSRV Total Water at Mars Launch (0.25 meter IRaSS wall)	3580.8	PHV Total Water at Earth Launch	800	PHV Total Water at Earth Launch	7011.6
				PHV Total Landed Water (post entry jetessoning)	800	PHV Total Landed Water (post entry jetessoning)	1752.9
				PHV Total Mass with Descent /Landing Propulsion Structures (GRDC) and Landed Water	28057	PHV Total Mass with Descent /Landing Propulsion Structures (GRDC) and Landed Water	29380
Total Mars Descent Propellant Mass	7735	Total Mars Ascent/Earth Return Propellant	115146	Total Mars Descent Propellant Mass	6919.6	Total Mars Descent Propellant Mass	7246
Mars EDL Heat Shield/System (PROTEA)	5864.7			Mars EDL Heat Shield/System (PROTEA)	5246.5	Mars EDL Heat Shield/System (PROTEA)	5493.9
SSRV Total Mars Deorbit Mass	44963	SSRV Total Mars Launch Mass	138345	PHV Total Mars Deorbit Mass	40223	PHV Total Mars Deorbit Mass	42120
SSRV Total Mars Landing Mass	31363			PHV Total Mars Landing Mass	28057	PHV Total Mars Landing Mass	29380
		CCEC Capsule at Earth Entry with EDL Shield (post TVAL, SPS & water jetessoning)	5994.7				
Part A (above)		**Part B (below)**				*Note: "Dry" means NO Propellant*	
		SSRV 1: SPS BI-Prop Mass at TMI Burn	78670	PHV 1: PHV BI-Prop Mass at TMI Burn	70377		
		SSRV 1: Total (w/BI-Prop) Earth Orbit Mass at TMI Burn	123633	PHV 1: Total (w/BI-Prop) Earth Orbit Mass at TMI Burn	110600		
		SSRV 2: SPS LOX-LH Prop Mass at TMI Burn (Pusher)	195264	PHV 2: SPS LOX-LH Prop Mass at TMI Burn (Pusher)	182967		
		SSRV 2: Total (w/LOX-LH-Prop) Earth Orbit Mass at TMI Burn (pre docking)	240226	PHV 2: Total (w/LOX-LH-Prop) Earth Orbit Mass at TMI Burn (pre docking)	230346		
		TOTAL Dual Docked SSRVs Earth Orbit Mass at TMI Burn	363859	TOTAL Dual Docked PHVs Earth Orbit Mass at TMI Burn	340946		

Note: components introduced later in text: CCEC - Central Control and Entry Capsule, TVAL - Transfer Vestibule Airlock, GRDAC - Generic Removable Descent/Ascent Cradle, PROTEA - Protective Entry Aerodecelerator, GRDC - Generic Removable Descent Cradle, SPS – Space Propulsion System.

redundancy during all mission phases; assembly line vehicle production resulting in a cost reductions; increased system validation from development thru installation and use; multiple vehicle expanded living space allowing for smaller overall vehicle designs which reduce individual vehicle mass at a decrease cost; and a built in ability to efficiently evolve vehicle structures and systems as lessons are learned between launch opportunities. Ultimately, with a plethora of implications and future mission profile choices, the surface infrastructure would be able to be expanded at twice the rate of previous Mars exploration paradigms.

During the assessment of this architecture, several major considerations were addressed which directly influenced the methodology across the design. The first being that launches to Mars would occur during every Conjunction class (or "fast-track") transfer alignment opportunity[4,5,6] in support of launch mass efficiency and moderate interplanetary and micro-gravity space flight durations (see Figure 3). The second is the desire to go to locations in the northern lowlands between 30 and 65 degrees latitude. This constraint resulted from the desire to collocate sufficient available and extractable near surface resources of which ice/water has been deemed the most important[13] and any scientific points of interest. Initial data supporting near surface ice theories has already been revealed by ongoing robotic exploration and surface characterization initiatives (Figure 4). In addition, targeting the northern lowlands enhances ballistic entry and landing profiles by taking advantage of the larger atmospheric column to assist descent deceleration and landing. Further investigations and precursor robotic studies may also directly impact specific mission concepts and designs.[14,15,16] A third consideration is that the primary launch platform for both to these Mars vehicles would be either a Shuttle-Derived Launch System (Figure 5) or another, yet to be developed and flight qualified, heavy lift alternative, with a capability of delivering roughly 240,000 kg into low Earth orbit (LEO). Though earlier in the production timeline of such a large vehicle, an existing medium lift expendable platform could be utilized for SSRV development in support of ISS operations or lunar testing and exploration. The rest of this document assumes access to these launch assets and does not further address any aspects of their development, testing or production.

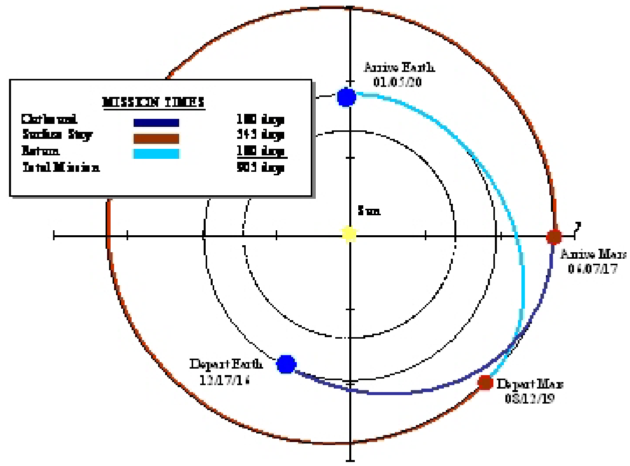

Figure 3. Conjunction class schematic.

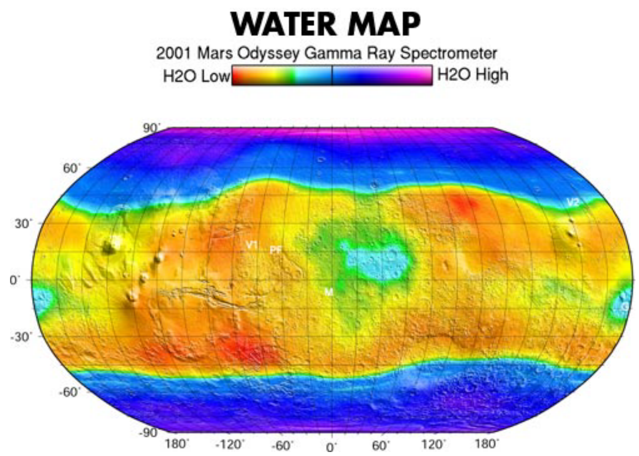

Figure 4. Gamma Ray Spectrometer water map, NASA/JPL.

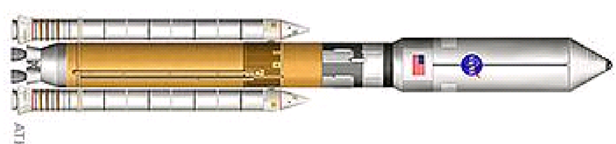

Figure 5. Shuttle Derived Launch System (historical ATK, Ref. 17).

The proposed architecture initially includes SSRVs configured for surface In-situ Propellant Production (ISPP) in support of system redundancy, mass constraints and surface fuel production. ISPP uses hydrogen feedstock and a Sabatier fuel-processing reactor ($4H_2 + CO_2 \rightarrow CH_4 + 2H_2O$) to produce methane and water. Additional plant processes would supplement and continue to process products via Reverse Water Gas Shift (RWGS) and water hydrolysis into a variety of consumables. Best estimates show that for every kilogram of hydrogen feedstock provided, roughly 18 kg of propellant can be produced.[4] In this design, initial human operational priorities will focus on building the infrastructure for in-situ water/ice reclamation, storage and processing. Yet, until sufficient water reserves are stockpiled, at least the first two Launch Group's ISPP systems will be designed to use hydrogen feedstock transported from Earth to Mars.[4] The transport of such Hydrogen supplies exemplifies one of the technologies that must be enhanced in order to increase cryogenic storage efficiencies (both in space and on the surface). As water supplies and fuel reserves are developed on the surface of Mars, future SSRVs will not need to carry either whole ISPP systems or the associated hydrogen feedstock from Earth, which ultimately further reduces subsequent Mars bound vehicle launch masses (e.g., decreased launch vehicle performance requirements) or increases the SSRVs useful mass transfer capability.

This design presumes the availability of nuclear electric power production and therefore similarly will not address the development of this topic further in this text, though alternative energy sources (e.g., solar, wind, ect.) are considered for the sake of redundancy. The initial Launch Group will be responsible for transferring the initial

Radioisotope Thermoelectric Generators (RTGs) or other power sources (e.g., the Toshiba/CRIEPI 4S reactor under development[18]) to the surface to provide ISPP and base power.[5]

Ultimately, and in short order, substantial habitable volumes, facilities and infrastructure could be emplaced with this architecture (i.e., landing six PHVs in six years provides the capability to accommodate roughly 36 crew members at one time), thus enhancing mission safety by maximizing systems and resource redundancies. Following sufficient base development an evolution to this plan could occur that would allow for the use of a quick-transfer version of the SSRV for moving crews to and from Mars. At that point, additional PHVs would only be needed to expand the current base, establish new bases or replace aging habitats. This methodology highlights a key design point in that it supports rapid evolution and adaptation while requiring infrastructure longevity.

The remainder of this chapter outlines the overall mission architecture and flight sequence as already presented in Figure 2 above. Note that any precursor test flights of such hardware in Earth orbit or on the lunar surface have not been identified pictorially.

A. Phase 1: The First Launch Group

This section describes the initial emplacement of human rated structures on the surface of Mars. As presented in the Figure 2 launch sequence, the initial three launch opportunities are completely parallel in composition. The mission profile begins at the first Mars launch opportunity before humans are sent and is comprised of the first "Launch Group"– two SSRVs.

At the opening of this launch window, two SSRVs are launched within a few days of each other (see Figure 6). Though this phase is unmanned, each SSRV is outfitted with a tower launch escape system that is designed to save hardware or crews during certain Earth launch abort windows or scenarios. Additionally and more uniquely, the towers are designed to contain a docking interface mechanism that provides physical connectivity between two SSRVs. Once an SSRV achieves Earth orbit, the rocket escape section of the tower is jettisoned leaving a roughly three-meter section of tower connected to the SSRV nose. After the second SSRV successfully enters orbit, it maneuvers to rendezvous and dock with the previous SSRV using the same tower docking interface mechanism (see Figure 7 & 8).

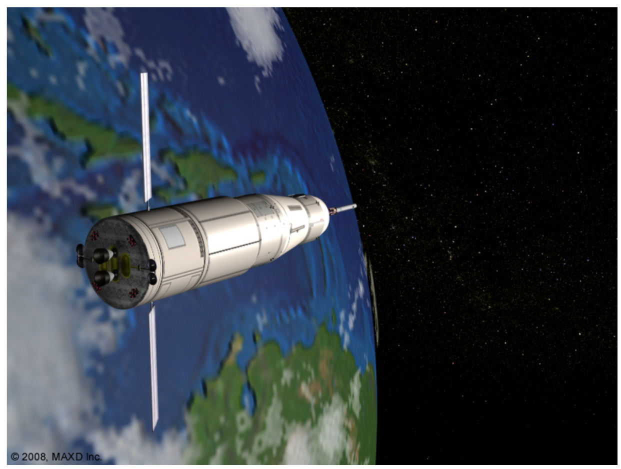

Figure 6. SSRV in Earth orbit following launch.

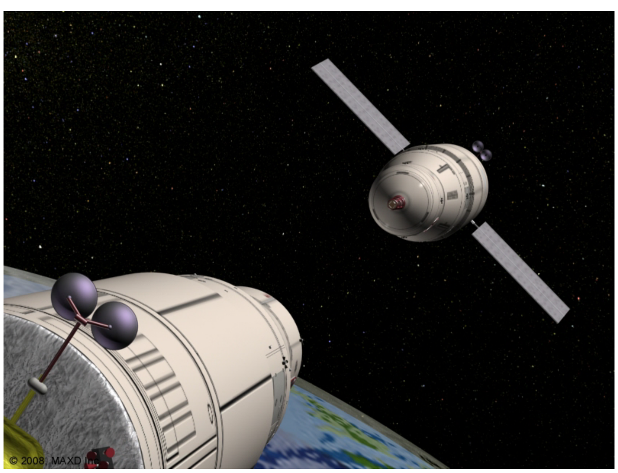

Figure 7. SSRVs rendezvous in Earth orbit.

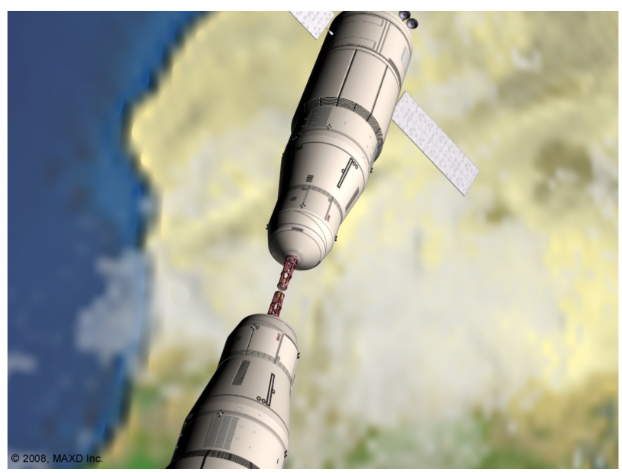

Figure 8. SSRVs docking via escape tower.

Within the same Mars launch window/opportunity, the second Launch Group consisting of two PHVs will similarly be placed into Earth orbit. Like the SSRV, each PHV is also equipped with a launch escape tower that also provides for safe distancing and returns to the Earth's surface for aborts from specified critical launch altitudes. Again, like the SSRV, the PHV attains Earth orbit and jettisons the rocket escape portion of the tower leaving clear the tower's docking interface mechanism. Finally, the two PHVs perform an Earth orbit rendezvous and dock.

These PHVs are launched with a maximum supply of long life consumables, equipment and roughly 1,000 thousand kilograms of water so as to augment the logistics of the upcoming piloted missions.

This "Launch Group" approach has several advantages for piloted interplanetary flight. First, the docking of two vehicles establishes complete vehicle redundancy during transit should any catastrophic failure of the primary vehicle or its systems occur (an Apollo 13 lesson). The concept of complete vehicle redundancy has not been earnestly considered in any designs to date and herein is not only a standard architectural driver but directly supports rapid base development and mission safety. Secondly, though not specifically a requirement, while in the docked configuration there exists the added potential or benefit that the two vehicles could be rotated during transit in order to provide artificial gravity (~0.4 g or the equivalent of a medium-radius centrifuge). Yet, even without rotational artificial gravity, the bio-physiological impacts (e.g., musculoskeletal and cardiovascular deconditioning and osteoporosis) incurred during micro-gravity transits between six and eight months are herein considered safely manageable given the 0.38 g immersion while on the surface of Mars and the use of currently understood exercise and pharmacological countermeasure regimes.[19] Though some authors challenge this assertion,[20] the initiation of the human exploration of Mars should not be dissuaded given the current state of knowledge concerning human physiological responses to extended duration space flight (e.g., spanning six and twelve months) and reduced gravity environments. Ultimately, these questions may only prove answerable by actually conducting such expeditions. In the mean time continued research and developments with regards to human physiological adaptation through space flight countermeasures and experience will only enhance our current capabilities and further support future Mars exploration.[21,22,23,24,25]

When launched, each individual vehicle (i.e., one SSRV or PHV) carries a fully fueled Space Propulsion Systems (SPS) for departing Earth orbit (see the aft section of the SSRV depicted in Figures 6 through 8 above). This section of the stack has a function that is similar to a combination of the Apollo Saturn V third stage (S-IVB) and Service Module stage[26] and is the primary interplanetary propulsion stage used for the Trans-Mars Insertion (TMI) and course correction burns. Yet, this design again quickly diverges by keeping the first SSRVs SPS stage attached, fueled and available for use throughout the planetary cruise phase. The second SSRV SPS provides sufficient propellant to propel a mated stack (e.g., two SSRVs or two PHVs) to Mars and alone each SPS is fully capable of propelling its associated vehicle (e.g., a single SSRV or PHV) to Mars. Again, the SPS on the launch of the first vehicle in any pair is fueled with an efficient space storable propellant (e.g., ideally a bipropellant with an Isp approaching 340 s) in order to support extended space operations (i.e., either while in Earth orbit awaiting the launch of the second vehicle or nominally as the orbit entry propellant at Mars). The second vehicle launched, and all piloted launches, use an SPS that is fueled with a higher Isp propellant such as oxygen/hydrogen (~ 460 s). After vehicles have docked, the SPS containing the non-space storable high Isp propellant is used to more efficiently conduct the TMI burn for the complete Launch Group mated stack. The remaining, fully fueled SPS serves as a backup emergency propulsion system during transit and is used to decelerate the stack during Mars orbit insertion, thus reducing orbit entry velocities and reducing entry shielding requirements.

Within the Launch Group concept additional SPS benefits can be demonstrated. The SPS provides primary electrical power and deep space telecommunications to the associated vehicles during the in route portion of the planetary transit. An inherent benefit of the mass distribution in the Launch Group configuration given the potential desire to create artificial gravity as mentioned above, is that the center of rotation would be shifted towards the unmanned side of the stack due to the unburned mass within the SPS stage of the vehicle launched first; thus increasing the rotation axis and allowing slower rates for a given artificial g-level. Finally, given such a configuration also adds to the vehicles long axis radiation protection due to increased mass.

The intent of the SPS is to provide a complete trans-Mars propulsive redundancy in case of errors or failures during TMI or other course adjustment burns (e.g., redundancy provides a direct-return emergency abort capability limited to a certain maximum transfer distance from Earth), to increase orbit entry precision by allowing for unplanned, real time orbit corrections (e.g., due to real-time Martian atmospheric changes); and to minimize the ballistic profile and stresses incurred during orbit entry resulting in simpler, low-mass heat shielding. The ultimate result is to bypass the use of direct entry, minimize the need for high velocity aerocapture techniques, enhance and supplement the use aerobraking and thus alleviate the Mars unique, supersonic to subsonic transition problem.

Using the SPS for propulsive orbit entry and refinement places the vehicle stack into a 500 km circular orbit with an entry velocity approaching 3.31 km/s, which enables the use of either conventional heat shield technologies or lighter weight inflatable or hybrid hypersonic balute type decelerators (e.g., the Inflatable Reentry and Descent Technology (IRDT) currently being tested by the Lavochkin Design Bureau, DaimlerChrysler Aerospace and others) used during atmosphere Entry Descent and Landing (EDL).[27,28] As with all components in this architecture the dual use of a single heat-shield structure (see Figure 9) in support of orbital capture, insertion and entry is an example of this designs holistic nature. This multiple-use shield design coordinated with the inclusion of sufficient propulsive systems increases mission flexibility and safety and lends to a redistribution of mission mass to more useful areas. Finally, by adhering to the Launch Group configuration methodology, this design uniquely safeguards long duration explorers by having two stages available for the trans-Mars flight.

With the destination in sight, the used SPS stages along with the remaining portion of the SSRV's and PHV's launch escape towers are jettisoned following vehicle stack separation and successful Mars Orbit Insertion (MOI). The SPS module though important, is not detailed separate from this section because it is considered as an intermediary part of the primary heavy-lift launch system architecture. Finally, each vehicle descends to the predefined landing area, equipped with transponder systems that collectively enhance the landing precision for each vehicle following in turn. Ideally, given sufficient surface arrays, the landing precision could be similar to aviation Instrument Landing Systems (ILS), Local Area Augmentation Systems (LAAS) or Receiver Autonomous Integrity Monitoring (RAIM), especially given the possibility of having a limited Mars Global Positioning System (MGPS).

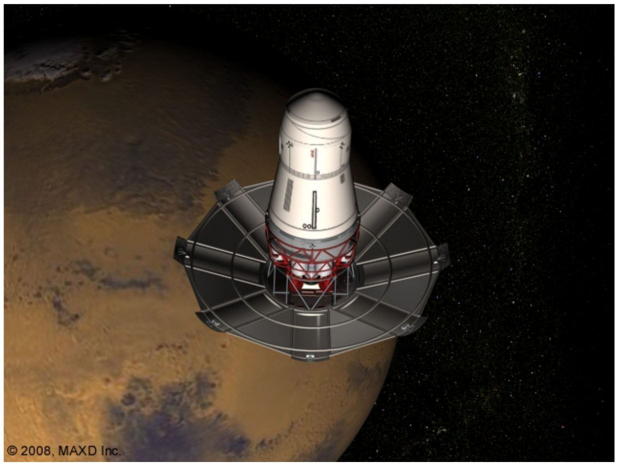

Figure 9. SSRV preparing to deorbit and land on Mars.

This unpiloted phase ends with two SSRVs on the surface, ISPP equipment generating methane and oxygen supplies for return and ground operations, and two PHVs positioned to provide expanded accommodations, provisions and spare parts. The SSRVs and PHVs hibernate on the surface until the next conjunction or roughly 500 days. Again, until propellant stockpiles are emplaced at Mars, each SSRV carries ISPP equipment and stockpiles in order to produce fuel for surface operations and Earth return. Fully rigged with sensors, the SSRVs and PHVs will also provide invaluable feedback on material lifetimes and environmental interactions for one entire launch cycle in support of the next phase.

B. Phase 2: Second Launch Group, the First Humans

The primary goal of the second phase is to place humans on the surface for long durations and return them safely to Earth. The first shakedown crew will have the responsibility of proving the safety and reliability of the vehicles and operations. In addition, the first crew will establish initial water reclamation and storage facilities, methane production and storage facilities, base infrastructure and lastly, organize and ascertain ground truth scientific data via reconnaissance and exploration.

As before, this phase begins with the launch, rendezvous and docking of a pair of SSRVs, and pair of PHVs. The major difference this time is that a six-person crew will be on the second of the two PHVs. A significant capability afforded these vehicles in the docked configuration, which enhances redundancy and safety, is the ability to transfer water, power and data via the docking tower. In the event that the piloted PHV should suffer a major, unrecoverable malfunction (i.e., damaged or uninhabitable), the crew could then transfer to the second vehicle via a space walk while retaining access to radiation shielding and consumables; thus saving both the crew and the mission.

Each PHV in this Launch Group carries with it its full structural capacity of water (~7,000 kg) in support of crew radiation protection and water consumption needs. Though the incorporation of such a mass of water may be considered a brute force method of mitigating radiation risks, it is otherwise overly compensated for by the nature of the integrated vehicle design, its cross use of consumables, and the flexible and rapid emplacement of a minimally sized redundant surface infrastructure.

Use of this dual launch design could continue at each opportunity favoring rapid surface expansion or increasing the duration of surface operations and habitation. Minor changes in the configuration of the SSRV, including expanding crew size and reusing various specialized components (e.g., inflatable structures), would allow it to eventually become the primary piloted transfer vehicle of choice once sufficient PHV habitats are emplaced. Only through a design that duplicates complete systems, in this case the docking of completely redundant vehicles for the Earth to Mars transit, is crew safety and mission success outcomes maximized. It is believed that this is the only modern architecture to include completely redundant vehicles for interplanetary flight in its initial scope (a lesson culled from Apollo 13). In addition, the operations and safety philosophy adhered to within only allow complex space operations like rendezvous and docking to occur in Earth orbit (i.e., with a clear and immediate communications and abort path to ground resources).

Mars orbit entry and landing nominally includes a propulsive capture using the Launch Group's unused SPS propellant as outlined in the previous section. The piloted and unpiloted vehicles (see Figures 10 and 11) descend in sequence to the landing site of the first Launch Group under manual (piloted case only) or automatic guidance and control. In the event that the piloted vehicle lands at a considerable distance from the primary site, the second PHV and SSRVs would then be redirected to this location in order to safeguard the crew.

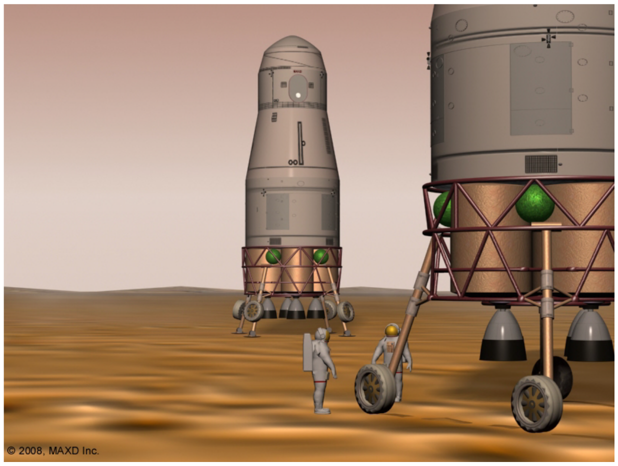

Figure 10. SSRVs on Martian surface.

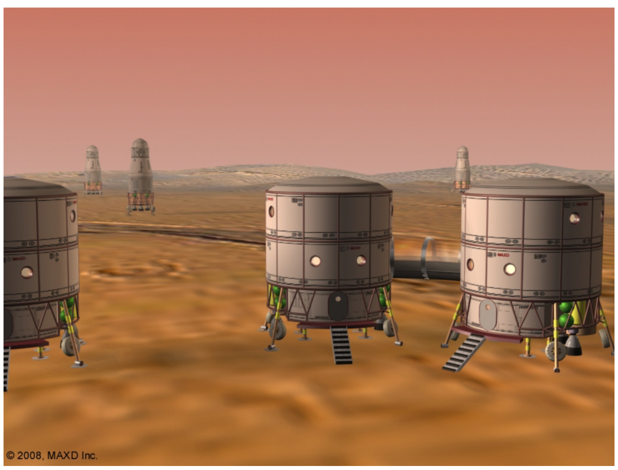

Figure 11. Building a base on the Martian surface.

At this point, there are four fully operational ISPP facilities available for ongoing propellant production. Once near-surface water reclamation operations have been successfully established and sufficient quantities are being processed and stored, and hydrolysis extraction of hydrogen enable continued methane production, would operations profiles be ready to change eliminating the need to ship additional hydrogen feedstock from Earth. This reduction in transfer mass would enhance all future SSRV transfer capabilities by roughly 6 tones. Again, it is important to note that in the unpiloted or piloted configuration, each vehicle alone is fully capable of traveling to and landing on Mars; though, in doing so it would most likely require either a heightened entry velocity (e.g., greater than Mars Orbit velocity of 3.31 km/s) or a longer duration orbital aerobraking phase (considered only for unpiloted Launch Group case).

For the last part of this phase, the first crew would prepare for the second crew and support their arrival by providing an appropriate handover before returning to Earth in their respective and completely inspected SSRVs. The phase ends with the SSRVs entering Earth orbit (see Figure 12), possibly rendezvousing with the ISS or other orbital platform for a period of decontamination and later return or simply a direct entry and landing of the crew module.

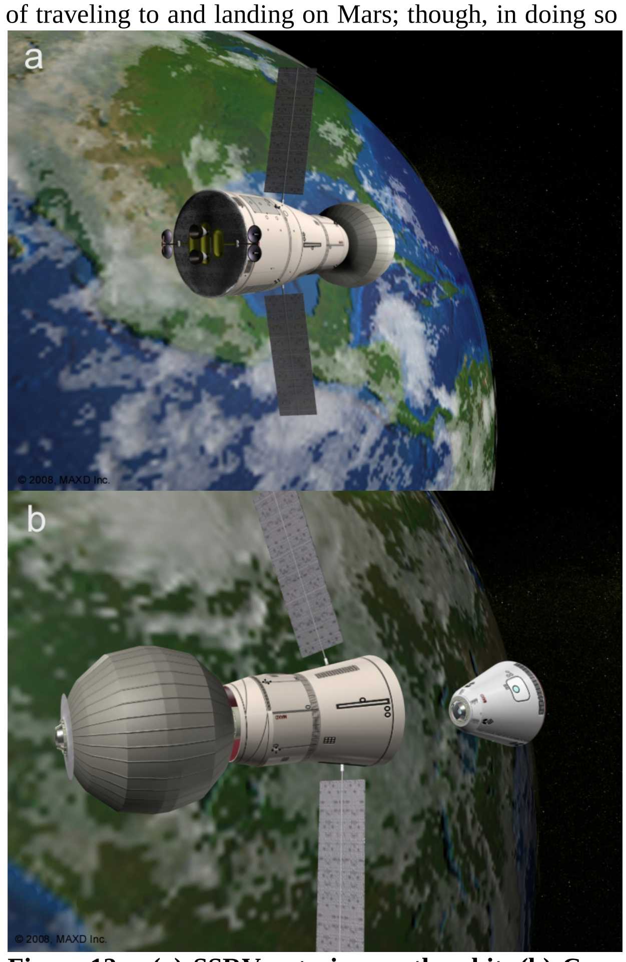

Figure 12. (a) SSRV entering earth orbit. (b) Crew return module separating from SSRV prior to Earth entry and landing.

3. Vehicle Description and Element Overview

This section provides a sequential review of the design philosophy and structural elements for the 34 ton Surface-to-Surface Return Vehicle (SSRV) and 32 ton Planetary Habitat Vehicle (PHV) as outlined in the previous sections (i.e., total Mars landed dry mass). As mentioned, two main drivers for overall vehicle design are radiation protection and surface dust mitigation. Another central theme to this design is the separation of functions throughout different compartments or sections depending on use during different phases of the mission. And lastly, all designs hold to the multiple-use-methodology and system component standardization.

A single PHV is designed to support a six-person crew for a minimum of 780 days (i.e., roughly 220 days interplanetary transfer and 560 days of surface operations) given this architectures enhanced and distributed vehicle emplacement approach (i.e., doubling of each vehicle type at every opportunity). Ideally in order to efficiently support ongoing crew rotations and long duration habitation, all PHVs and associated structures and systems elements should be designed to survive and operate on the planets surface for no less than 10 years without major refurbishment or replacement. The SSRV on the other hand, serving as a planetary transfer vehicle is directly designed to support two six month interplanetary flights, two planetary entries using separate entry and landing structures and systems, and a minimum 600-day hibernation period on the planet surface.

Note that even though the SSRV is designed to support a returning three person crew for between 180 and 220 days of interplanetary flight, in an emergency condition, this vehicle should be capable of supporting up to the entire six person crew for safe Earth return. This emergency scenario would prove cramped due to the addition of supplies and operated under special conditions such as sleep shifting in order to support nominal radiation protection regiments; and such a change in mission operations would be at the expense of creature comforts and any returned science. Ultimately, it is this author’s belief that given sufficiently reliable and redundant systems, adequate warmth, food, communications and hope, most any foreseeable condition can be dealt with and survived. History again aptly demonstrates the human capacity to survive harsh, chaotic and confining conditions as exemplified by the staunch resolve shown during the spectacular-failure of the Shackleton Antarctic expedition of 1915 (Figure 13).

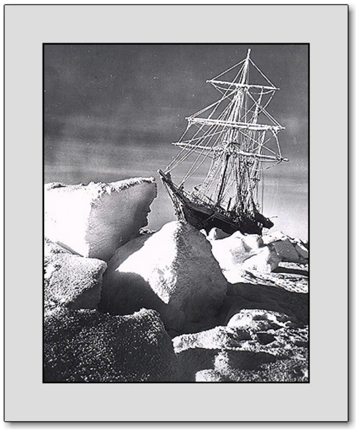

Figure 13. Frank Hurley photo of HMS Endurance, circa 1915, Ref. 29.

A. The Surface-to-Surface Return Vehicle (SSRV)

The SSRV (see Figures 14 and 15) consists of three primary components or segments. The first is the Transfer Vestibule Airlock (TVAL). The TVAL contains an airlock, water closet, stowage space and provides passage and connectivity to the Expendable Extended-Duration Inflatable Gondola (EEDIG); which provides an expanded habitable volume for use during interplanetary transfers. The second component is an Earth return crew module, which provides a centralized location for vehicle control and is called the Central Control and Entry Capsule (CCEC). The third is the Generic Removable Descent/Ascent Cradle (GRDAC), which houses propulsion, storage and the landing systems and structures of the SSRV.

Designed to nominally ferry three persons from the surface of Mars back to the surface of Earth combined with

Figure 14. SSRV (TVAL and CCEC) transparent with breakout.

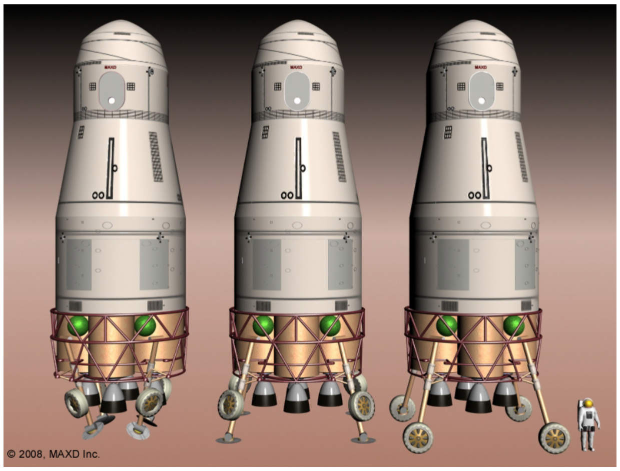

Figure 15. SSRV with GRDAC: launch, landing and surface transit configurations.

the ability to survive long durations and rapidly evolve make the SSRV a true exploration vehicle. As an assemblage of three components it is designed to support sequential segments of interplanetary and surface operations for missions lasting in excess of three years. The primary exploration-enabling components in this vehicle are the TVAL and the EEDIG inflatable structure (either spherical or barrel in shape); and together they form a large lightweight habitable volume that provides protected vehicle access and a division between unneeded and needed Earth return hardware, systems and components.

The SSRV measures a little over 7 meters in length without the launch escape tower, and is 5 meters in diameter at its widest dimension (i.e., the maximum diameter of the frustum-protective shroud which shields the CCEC). The maximum total dry (i.e., no water or propellant) mass of the SSRV including the GRDAC is approximately 31,363 kg. Crews safely access the TVAL airlock in either Mars or lunar gravity via a ladder that includes a rigging system, guy-wires, adjustable decking and handrails. This vehicle, along with the PHV detailed in following sections, normally operate at a pressure of 10.10 psi or roughly equivalent to the elevation of the highest incorporated city in the United States, Leadville Colorado (10,152 feet or 3,094 m – population ~2,800); Yet, the overall vehicle hull would be designed to withstand a maximum pressure of 14.7 psi (i.e., Earth sea level). Upon return to the Earth (see Figure 12b), the CCEC undocks and separates form the expendable TVAL (including EEDIG) prior to atmosphere entry and landing; in effect, shedding all unneeded mass. As a combined structure, each of the SSRVs components uniquely define an integrated lightweight advanced vehicle, which provides a large internal volume enabling long duration space and planetary exploration.

B. The Transfer Vestibule Airlock (TVAL)

The TVAL is a relatively small (i.e., at 13 m^3 it is a little over twice the internal volume of an ISS Pressurized Mating Adaptor (PMA)), lightweight aluminum and composite structure that provides three way access between the surface airlock-compartment, the EEDIG and the CCEC (using common docking/hatch mechanisms). Table 2 provides a detailed list of the major structural and operational functions associated with this unique, long duration

Table 2. Transfer Vestibule Airlock Enhanced Breakout.

System	Characteristics
Vestibule 1 - Airlock	A lateral, two-door, compartmentalized compartment primarily used for module access and entry on a planetary surface.
Airlock Hatchways & Frames	Electro-statically charged door frames repel or redirect suit born dust.
Primary TVAL Cylindrical Volume	Electro-static/magnetic floor grating and containment area, vacuums, gas pressure shower, and post-pressurization water cleaning systems for dust and contamination mitigation from surface suited crews.
Vestibule 2 - Stowage	Provides stowage and containment for used surface suits and other supplies.
Vestibule 3 - Lavatory	Provides a water closet alcove (latrine) and hygiene area.
Vestibule 4 – Systems-Lock	Provides system hardware, stowage and plumbing control access.
Conical Protective Shroud - Internal	Encases structural mounts used during interplanetary flight for ECLSS pressurized vessels (oxygen and nitrogen), plumbing and tanks for hypergolic or cold-gas propellants, solar arrays stowage (~ 4 watts/sq.ft.) and power distribution, and thermal heat rejection plating/arrays (max ~ 12,000 Btus where the Apollo equivalent for three persons was 7,220 Btus.[26]
Conical Protective Shroud - External	RCS propulsion system thruster mounts, external ladder assembly enabling transit between the airlock, the GRDAC platform and the surface, omni direction low-rate communications antennae (e.g., ISS s-band), protects vehicle from dust accumulation on Martian surface, and from micro-meteorites during interplanetary flight.
EEDIG Stowage Ring and Launch Cap	An enclosed, dust-protected space for storing a folded and compressed EEDIG inflatable (~2.5 cubic meters). Launch cap provides attach points for launch-escape tower and redundancy plumbing, and structural connections for the Mars entry parachutes.
TVAL Standard Docking Mechanism with Hatch	Provides structural mounting and seals for two standard docking rings and hatches for pass through between the EEDIG and CCEC.

enabling spacecraft component. The habitable portion of the TVAL (see right side transparency and breakout in Figure 14 above) is approximately three meters long and two meters in diameter at its widest point. The total mass of the TVAL and associated components is roughly 1000 kg. Additionally, by incorporating minor structural modifications, a TVAL vestibule could be adapted to support any number of Earth reentry vehicle shapes (e.g., bullet or semispherical).

The TVAL is specifically designed to address the need to mitigate surface dust contamination, which is considered one of the primary architectural drivers. Currently it is estimated that the Martian atmosphere entrains electrostatically charged dust particles that could range as large as 100 microns.[30,31] Even given our intended landing constraints of 30 to 65 north latitude, dust adhesion and contamination is an expected factor given that several hundred local or regional dust transferring events (e.g., dust devils or equatorial-crossing storms) can occur each year.[32,33,34] This also includes the occasional global dust storm. Once entrained in the atmosphere, dust is expected to inundate surfaces and mechanical joints through either Van der Waals forces or electrostatic adhesion.[35] Therefore, long-term protection from dust requires an examination and coordination of both the crew surface-suits and the vehicle design as a whole. Both the habitat and the suited crewmember are considered herein as integrated systems that periodically separate and then reconnect in a manner that maintains environmental integrity and isolation. Suit designs must therefore seamlessly integrate with the habitats airlock structures and functions. Additionally, the design should minimize atmosphere leakage while providing an isolated and contamination controlled workspace for suit stowage and repair. Lunar extravehicular activity (EVA) suit history teaches that wrapping or covering critical access points proved marginally useful and that continued use without repair or replacement would hinder ongoing EVA capabilities. Further, crews reported the distinct odor of lunar dust since they suited inside the only habitable portion of Lunar Excursion Module (LEM) following each EVA.[26] Frequent long-duration exposure on Mars will require additional simple and creative mitigation schemes in order provide protection and insure operability and longevity (e.g., clean plug-n-play connecting of the suit to base power in order to apply an AC signal to surface layers of a suit prior to ingress).[36] As for the vehicle, several methods will be designed into the airlock structure in a systematic and efficient manner to enable rapid, frequent and uninhibited excursions between the surface and the habitat. This architecture espouses a partitioned airlock design approach that incorporates a combination of mechanical, liquid washing, gas blowing/vacuuming, electromechanical and electrostatic cleaning procedures directly into the vehicles initial design. Many of these techniques are well proven and routinely used in terrestrial environments.[37,38]

Reviewing at an overview level, the vehicle provides the following primary functions: 1) an airlock which incorporates several dust and contamination mitigation devices and structures as mentioned (e.g., ambient atmosphere vacuum cleaner, CO_2 pressure wash, electrostatic and electromagnetic threshold plating and brushing devices, sonic or radio bathing, and post-pressurization wet wipe-down); 2) the structure for supporting the habitable pass-through volume, including stowage vestibules and lavatory, 3) the structure for stowing and deploying a six meter diameter inflatable structure (the EEDIG), 4) the structural connectivity, plumbing interfaces and pass-troughs between the EEDIG and the CCEC (i.e., power, data, ventilation (e.g., like ISS inter-module vent valves), 5) the structures for systems and consumable storage required during the planetary transit phase (e.g., compressed gases, photovoltaic arrays, thermal regulation either body mounted or arrayed and waste management systems).

C. The Expendable Extended-Duration Inflatable Gondola (EEDIG)

The EEDIG is an inflatable structure that has been incorporated into this architecture to maximizing habitable volume and packaging efficiency while reducing overall spacecraft mass and cost. At launch, the EEDIG is folded and stowed in an environmentally controlled and monitored peripheral containment ring atop the TVAL structure (see Figure 16 and right side see-through in Figure 14 above). Each end of the EEDIG has a standardized docking hatch interface mechanism that provides a mechanical connection to the cylindrical transfer vestibule of the TVAL when in either the stowed or deployed configuration. During inflation, the outermost docking mechanism releases a series of internal clamps allowing the inflatable to deploy symmetrically using internal tensioning rings

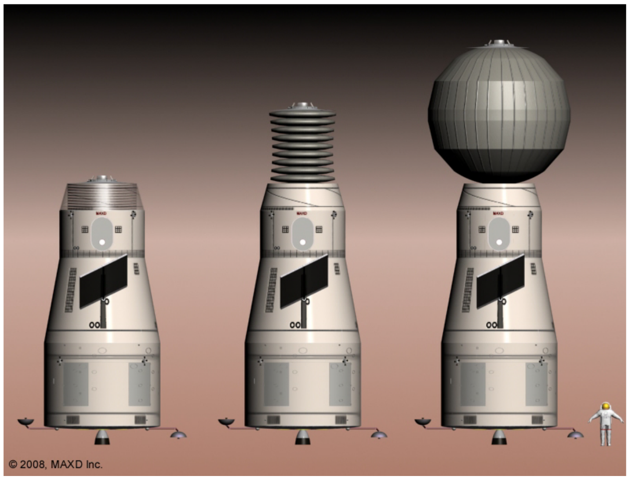

Figure 16. SSRV: EEDIG deployment sequence.

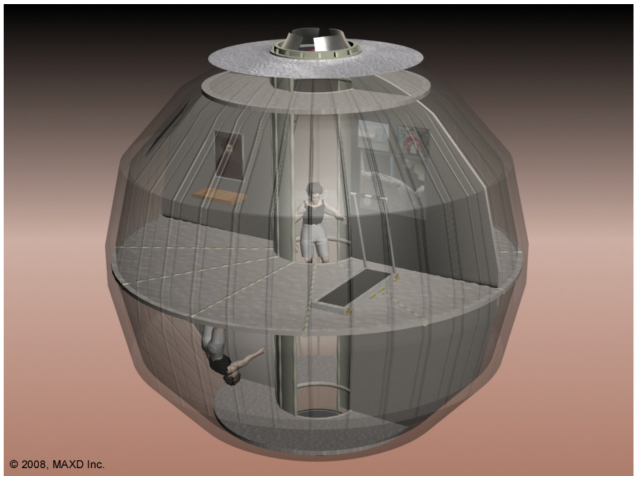

Figure 17. Transparent EEDIG in inflated configuration.

and guide wires. Once fully inflated (see Figure 17), the EEDIG inner diameter spans roughly 5.8 meters with an outer diameter 6 of meters. This provides a non-furnished volume of roughly 100 m^3 and a volume-to-surface ratio of 0.97. The total mass for the primary structures and inflatable is estimated to be 800 kg based on current technologies.[39] The skin is composed of a series of layers that will provide adequate thermal and space environment protection (e.g., debris impact protection). At a minimum, the skin is layered using of an optimized mix of the following materials and high strength polymers: Multi Layer Insulation (MLI), Kevlar, Vectran, Spectra, Kapton, Nextel and Sonex.[39,40]

This amount of volume provides returning crews more than adequate accommodation space for a six-month voyage. The primary uses of this expanded volume include the following: A) crew exercise, B) crew recreation, C) preliminary sample analysis and research, D) hygiene and waste storage and E) depending on future analysis an Earth return orbit entry assist aerobraking surface. The internal structure of the EEDIG consists of an internal tensioning ring which provides a framework for installing inflatable, plastic or self tensioning fabric dividers and decking; quick installation plumbing tracts for systems; and accommodation space to connect portable systems and hardware (e.g., thermal conditioning, power distribution, data handling, ect.). Once inflated the crew will transfer and install the internal framework and any hardware and equipment needed to support the six-month return journey (e.g., fans, light-weight portable treadmill, cycle-ergometer, ect.). As mentioned in potential future use above, when the EEDIG is oriented in the direction of flight, it could serve as a throw-away aerobraking surface that could assist the SSRV during an elliptical Earth orbit entry enabling the possibility of rendezvous with the International Space Station (ISS) or other orbital platform in support of planetary quarantine operations. Another look ahead poses the possibility of a second generation SSRV that would serve as the primary interplanetary crew-transit vehicle. It is further suggested that the EEDIG structure could be inflated and re-stowed prior to Mars entry following an Earth to Mars transit; the same inflatable being used again during the return flight between Mars and Earth.[40]

D. The Central Control and Entry Capsule (CCEC)

This structure is a vehicle unto itself (Figure 18) and consists of a metallic capsule that is designed to reenter planetary atmospheres (Apollo derivative) using an ablative heat shield and reentry reaction control system (RCS).[26] The CCEC design also includes the following functions and capabilities: 1) a standard interface-docking mechanism with hatch which provides connection to the TVAL, 2) primary Environmental Control and Life Support (ECLSS) systems, 3) data handling and commanding systems, 4) radio communications systems, 5) consumable stowage (e.g., food), 6) TVAL/EEDIG portable outfitting systems and components, 7) side hatch, 8) windows, 9) return sample stowage, 10) reentry RCS propellant, 11) reentry parachutes and 12) landing systems. The internal empty, un-outfitted volume is 30.6 m^3 with a maximum diameter at the base (i.e., the heat shield) of 4.2 meters and a cone height of three meters. The usable volume of the CCEC is taken as 1/3rd the pressure volume or roughly 10.2 m^3 (Larson and Pranke,

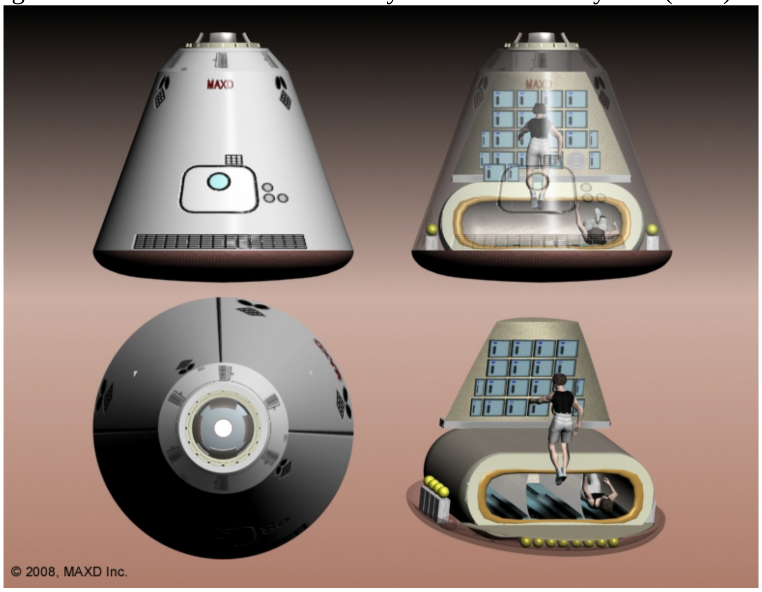

Figure 18. Central Control and Entry Capsule (CCEC).

2000). The total empty and dry mass of the CCEC structure is approximately 2,300 kg or 5,994 kg fully loaded for Earth reentry, including the Earth-entry shield and landing systems. Table 3, below, provides a comparison of similar historical, existing and proposed Earth return spacecraft.

Table 3. Crew-return vehicle comparisons.

	Apollo Program	Russian Space Agency	ESAS Architecture	MAXD Architecture
Return Capsule	Command Module	Спускаемый аппарат - "Descent Module"	Crew Exploration Vehicle - CEV	Central Control and Entry Capsule - CCEC
Diameter, max (m)	3.9	2.2	5 (+)	4.2
Mass, max (kg)	5806	2980	9237	At Earth reentry w/heat shield, crew and cargo ~ 5307
Cargo Capacity, max (kg)	Minimal < 200	Minimal ~ 50	Minimal < 200	> 500
Crew compliment (off nominal)	3 (5)	3	4 to 6	3 (6)
Radiation Protection	Vehicle structure ~ 4.5 g/cm^3	Vehicle structure ~ 2-3 g/^{cm3}	Vehicle structure ~ 2-3 g/cm^3	IRaSS ~ 27 g/cm^3 (water), not including vehicle structural shielding
Space Lifetime, max	~ 2 weeks	200 days	6 months	3 years
Mission Versatility	- Earth orbit - Lunar orbit - Skylab	- Earth orbit - MIR - ISS	- Earth orbit - Lunar orbit - ISS	- Earth orbit - Lunar orbit/surface - Mars orbit/surface - ISS

Unlike other proposed and past earth return vehicles, the CCEC has been stripped of all non-entry and landing essential equipment (e.g., long range power and thermal, hygiene and waste systems). The exception to the essential Earth return structures list is the primary long duration radiation protection containment system. Radiation protection systems are paramount and are considered a leading threat to long duration human space exploration. Therefore, structures designed to provide such protection were defined herein as the second primary driver used in developing this architecture.

Radiation environment characterization and resulting exposure limits remain uncertain and undefined for human travel above low Earth Orbit (LEO) and remains one of the communities' greatest limiting factors to human space exploration.[41,42,43,44,45] Accurate environmental prediction and downstream design optimization are currently obtained from radiation transport modeling and available insitu measurements (i.e., the Mars Odyssey MARIE experiment[43,46]). Given the amount of uncertainties within the field of radiobiology, vehicle designs must therefore be stringent enough to accommodate worst-case conditions (i.e., based on historical solar events such as that of August 4, 1972) in order to support near term long duration free-space travel and permanent surface habitation. Radiation dose requirements provided herein use the 95% Confidence Interval (CI) limits, which are roughly taken as 3.5 times related the traditional point estimate calculations.[47] Tables 4 and 5 provide a list of the National Council on Radiation Protection and Measurement (NCRP) recommendations for radiation dose limitations. The units of dose equivalent are the Sievert (Sv) where, 1 Sv = 100 cSv = 100 Rem.[48,49]

Table 4. Organ dose equivalent limitations, adapted from NCRP-132, Ref, 48.

Exposure Interval	Blood Forming Organs (BFO) Dose Equivalent (cSv)	Ocular Lens Dose Equivalent (cSv)	Skin Dose Equivalent (cSv)
30-day	25	100	150
Annual	50	200	300
Career	See Table 5	400	600

Table 5. Low earth orbit career whole body dose equivalents (cSv), adapted from NCRP-132, Ref, 48.

Exposure Interval	Blood Forming Organs (BFO) Dose Equivalent (cSv)	Ocular Lens Dose Equivalent (cSv)	Skin Dose Equivalent (cSv)
Age	35	45	55
Male	100	150	290
Female	60	90	160

Spacecraft structures provide direct and/or indirect shielding by either scattering or absorbing incident radiation. Shield protection, and therefore effectiveness, is driven primarily by material chemical composition, mass density and ultimately by its response to the transport of energetic particles. Materials with high hydrogen content, like water or hydrogen gas, are considered the most effective per unit-mass with respect to high-energy charged particles (i.e., the most difficult spectral component to shield from). Vehicles designed to date have relied on basic spacecraft structures and interior components to mitigate space radiation. Aluminum (2.7 g/cm^3 per cm of thickness) has been the most common material and has been demonstrated to be unsound protection against galactic cosmic radiation (partially due to the production of secondary radiation[42]). Additional constraints on vehicle designs have been and will be driven by mass to orbit costs, requiring trade offs and optimizations, which inherently impinge and diminish adequate shielding designs.

For interplanetary flight to Mars, two sources including solar particle events (SPE) and the galactic cosmic radiation (GCR), including its high charge and energy spectrum (HZE) portion, are of utmost relevance. GCR is primarily composed of extra-solar protons and an abundance of various element; the six most common being H, He, C, O, Si and Fe.[42,44] GCR when compared to the more rarely occurring SPE events has been considered the primary career limiting exposure source during long duration missions. The most constraining exposure limit in general is the 50 cSv annual BFO limit presented in Table 4. The minimum free-space, zero shielded GCR dose equivalents (95% CI) have been estimated to be over 200 cSv/year (0.595 cSv/day) during Solar Minimum and roughly half again as much during Solar Maximum.

Conversely, crewmembers on planetary surfaces have additional protection from radiation exposure through half a sphere by the regolith (e.g., the surface blocks 50% of the GCR). Mars, as opposed to the Moon, also provides between 16 and 22 g/cm^3 protection through half a sphere in the atmospheric direction depending on altitude.[44] Lastly, by choosing landing sites in the northern lowlands, crews are afforded the greatest extent of atmospheric protection while on the surface. The surface aspects of radiation protection are primarily important with regards to the 560-day surface stay in the PHV as described in detail below.

The design of the SSRV's primary radiation protection system originates within the state-of-being and human behavior driven design philosophy as demonstrated by the integrated, structural and operational capabilities of this architecture. We observe that humans inhabit only two primary states of being; that of being awake and of being asleep. Each is distinct and one is usually highly predictable with respect to physical localization. While asleep we are usually immobile and when awake we move about. This design combines this state information and incorporates another needed material and consumable, water (1.0 g/cm^3), into a structure for storage and radiation protection.

The Interplanetary Radiation Sleep Shelter (IRaSS) is a three-person water-wall shell within the CCEC and is this architectures primary example of multiple-use-methodology (see Figures 19 and 20). The design takes advantage of the localized human sleep state by surrounding crewmembers in a layered aluminum/composite water tank. The IRaSS is internally compartmentalized into tanks that can contain potable and waste water (e.g., condensate, ect.). This sleep-shelter tank is subsequently wrapped by a few light-weight layers of radiation and structurally protective materials such as Kevlar or Demron.[50]

Upon Earth return, all remaining water is jettisoned just prior to reentry in support of reducing entry mass.

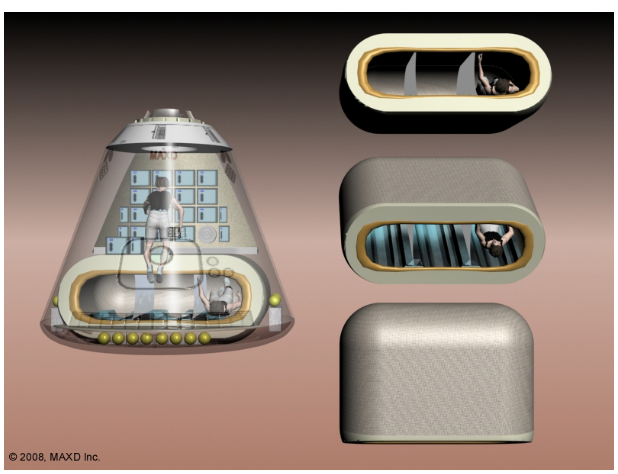

Figure 19. CCEC with IRaSS breakout.

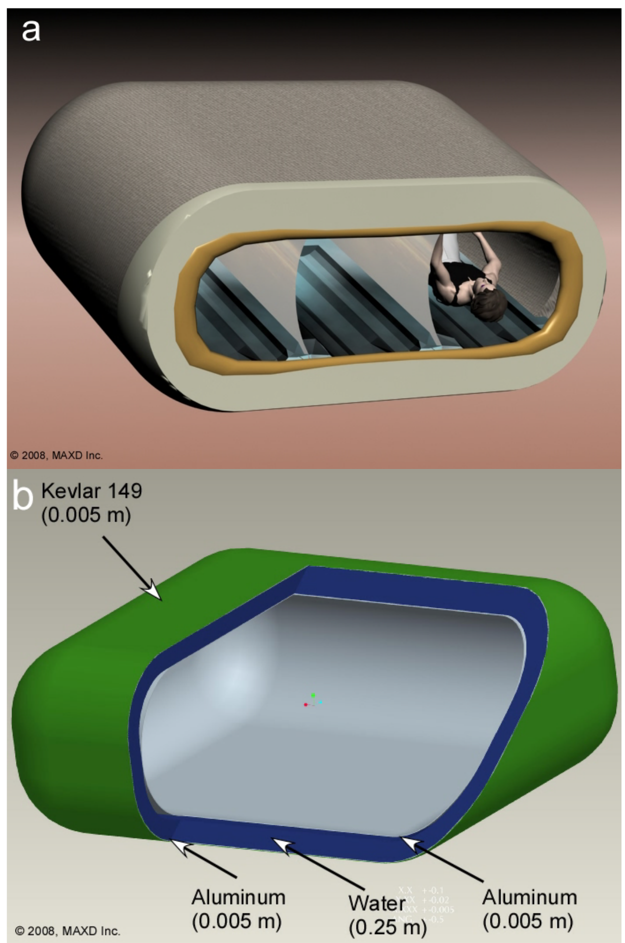

Figure 20. (a) Interplanetary Radiation Sleep Shelter (IRaSS), (b) IRaSS cutout.

Limited waste water processing and recycling capabilities for the interplanetary transit would be provided in order to maintain sufficient consumable supplies and shelter shielding. Other unprocessed waste (e.g., urine and wash water) is collected in an expandable collection tank located in the bottom of the EEDIG peripheral containment ring mentioned above.

In all, the IRaSS provides sufficient radiation protection for both nominal and emergency scenarios. Ideally, estimates of radiation protection for trans Mars flight could be reduced by a factor of 1/3rd given overall vehicle structural composition and an eight-hour sleep period within the IRaSS (such a reduction would allow crews to be exposed to longer duration Extra Vehicular Activity (EVA) surface operations which are considered herein as the most important aspect of Mars exploration).

Figure 21 schematically represents the radiation environment within IRaSS. Analysis was performed using the HZETRN 2005 transport code, Badhwar-O'Neill modulated free-space spectra with CRÈME 96 abundances and the SRAG Barrier Thickness Evaluator and post–processor (v4.0 and v4.2 respectively). The calculations indicated a very low, free space dose rate of 0.035 cSv/day and 0.065 cSv for the SPEs within the IRaSS. A conservative result of using the IRaSS in the manner described above is to decrease the nominal one-way interplanetary trip exposure by nearly 36 cSv (36 Rem) to 74 cSv (74 Rem) assuming a total 110 cSv (110 Rem) exposure based solely on a 220 day transit and 10 g/cm^2 average structure equivalent aluminum shielding at Solar Minimum. Though this value is still above the annual 50 cSv NCRP limit, it is suggested that crew selection for initial base development focus on males over 50 years of age. It is also the belief of this author that any person intent on engaging in such an adventure will be completely willing to take such risks and that future program administrators need to take this risk-value into consideration if any such missions are to occur within the life time of the current population of space professionals. It is important to note that the remaining SSRV mass and structures have been minimally considered in this rough dose estimate, and therefore the true value should be somewhat lower. This design inherently reduces overall exposures by up to a third, though, more modeling and work remains to be done in order to quantify the exact dose reduction.

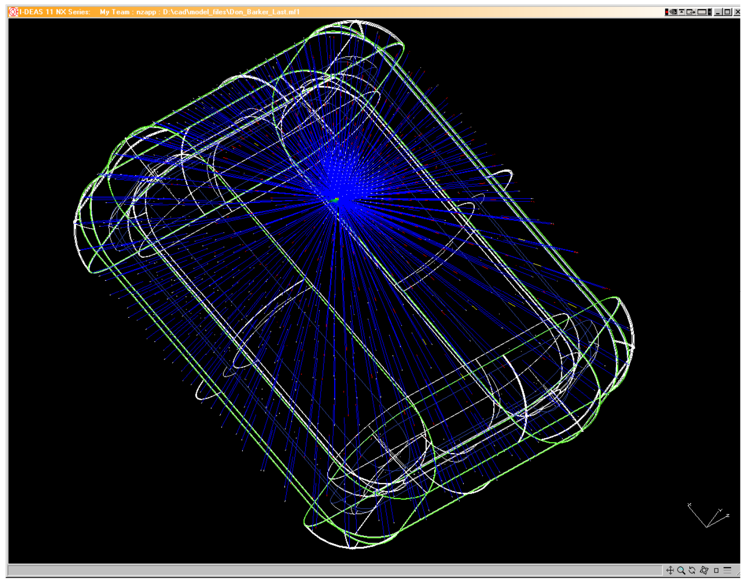

Figure 21. Interplanetary Radiation Sleep Shelter (IRaSS) radiation track analysis (blue tracts depicts low dose).

The IRaSS is currently envisioned as a Kevlar-polyethylene wrapped (possibly bound with a polyetherimide resin) aluminum shell with a sectioned internal bladder/structure that accommodates 0.25 meters of encompassing water insulation. This results in approximately 25 g/cm^2 of hydrogen rich shielding alone with a water volume of roughly 4.21 m^3 and mass of 4,209 kg. This amount of water is roughly twice the allotment needed to support a three-person crew for the 220-day (i.e., a crew of three is estimated to require 2,145 kg water) return portion of the mission given basic human requirements. The exception being the 6 person emergency scenario where this volume of water provides nearly the exact amount needed to safely return all crewmembers. Water need estimates used a value of 3.25 kg per person per day depending on environmental temperature and exertion levels. The total mass of a fully wet CCEC is then raised to roughly 10,2034 kg. Since this water is carried only during the interplanetary transfer portion of the flight, the excess storage capability of the IRaSS above the three person consumption requirement is considered to be a transient-replenishable radiation shielding (i.e., when empty it is only a light weight reusable shell); and, in this authors opinion such an impingement to launch mass is a necessary burden, or driver to enhance launch capabilities, in order to provide long duration crew radiation protection. Again, upon Earth arrival all excess water is dumped overboard just prior to atmosphere entry and landing, thus further reducing impacts on EDL system performance, overall vehicle mass and design.

Another major function of the CCEC is to provide the primary interfaces for the control of systems and operations of the SSRV throughout all flight phases including Earth entry and landing (see Figure 22 below). The IRaSS supports the functional multi-use methodology aspect of the CCEC in that its second main design is to provide a location for vehicle operational control. That is, the IRaSS berths function as fully recumbent couches used during launch and landing. Each crew berth is fully plumbed with air conditioning, power and data, and is outfitted with mobile workstations and communications terminals. The fully recumbent position directly supports physical adaptation to heightened launch and landing g-loads and reduces the likelihood of incidences of orthostatic intolerance. The concept of multiple-use-methodology is thus served in at least three ways: a sleeping berth, a radiation shelter and a launch/landing couch. Additionally, as previously mentioned for the six person emergency return scenario, the CCEC could hold three additional recumbent, folding launch/reentry-chairs mounted atop the IRaSS in a fashion similar to the Space Shuttle middeck recumbent seating used during Shuttle-Mir crew return flights. Lastly, as with many of the new, Apollo-derivative concepts, the CCEC is ultimately recovered and refurbished for future flights.

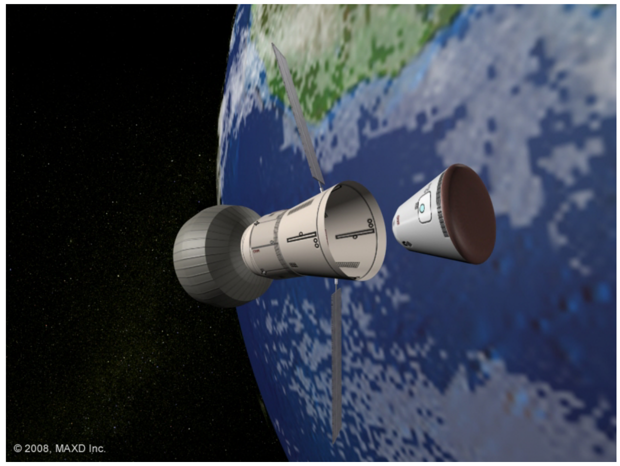

Figure 22. CCEC separating from SSRV prior to Earth entry and landing.

E. The Generic Removable Descent/Ascent Cradle (GRDAC)

The GRDAC is a metal and composite truss framework that provides at a minimum, the following: A) a descent navigation and propulsion system; B) an un-fueled ascent and Earth return propulsion system (to be presented below); C) Mars ISPP hardware systems and hydrogen feedstock storage; D) landing leg structures, pads, wheel axels and electric or methane/oxygen surface drive mechanisms; E) alignment and grapple attachments to the SSRV and Mars EDL shield (also outlined below); F) command and control interfaces to the CCEC/SSRV; E) deployable ladder and porch assemblies for accessing the TVAL airlock from the surface.

The terminal descent portion of the propulsion system, at least for the first few flights, uses approximately 7,206 kg of a space-storable hypergolic bipropellant, which is selected in order to assure ignition during descent. The maximum total SSRV mass at Mars entry is calculated to be roughly 44,963 kg including fuel and EDL shielding.[12] The last function of the GRDAC is to serve as the launch platform and gantry for the SSRV at the time of crew departure. The ascent and Earth return stage of the SSRV uses approximately 132 tons of methane-oxygen propellants derived from roughly 6 tones of cryogenically stored hydrogen feedstock.[4,5] As discussed previously, this propellant is processed by the ISPP systems Sabatier reactor using the feedstock hydrogen and the ambient Martian atmosphere. Additionally, roughly 16,000 kg of water are produced by each SSRV ISPP process. This water is transported and stored within the empty hulls of the PHV for eventual use (e.g., hydrolysis) or simply as additional surface radiation shielding.

The GRDAC is somewhat unique in that the organization and mounting of the propulsion system elements, including tanks, plumbing and engines, it is designed to be easily removed and replaced as necessary by suited personnel while on a planets surface. This easy access allows crews to inspect and repair in the short term while in the future, allowing for complete refurbishment and reuse of hardware intentionally left on the surface. Such a capability should drive engine and propulsion system designers to adopt designs that are capable of being easily reused, refurbished and reconfigured in support of building insitu hardware and component reserves and stockpiles.

F. The SSRV Hybrid Entry, Descent and Landing Shield

As briefly outlined in earlier sections, this portion of the architecture is the most theoretical in that it incorporates current concepts of the large inflatable aerodynamic decelerators (i.e., ballutes).[27,51] Such systems can provide a low ballistic coefficient, which allows high altitude decelerations with reduced heating. This work proposes a novel hybrid that combines both a hard-shell portion and an integrated inflatable portion as extensions to the standard centerline heat shield. Though more analysis is needed on this kind of hybrid concept, one of the primary purposes for this design is that it takes advantage of the shield’s hard portions while in its collapsed configuration to act as a portion of the SSRVs protective launch shroud and therefore eliminates the need for such additional protective structures. Ideally, the advantage being the reduction of launch mass by combining functions. As shown in Figure 23, this hybrid ballute is called the Protective Entry Aerodecelerator (PROTEA) system. The primary centerline heat shield forebody forms between a 60 and 70-degree sphere-cone in this rendition (i.e., attaining the highest L/D ratio possible). The five-meter diameter central heat shield is a common ablative, designed to withstand the brunt of the entry heat loads. The vehicle connects to the aft heat shield via a truss network and the SSRV upper body surfaces are hardened to withstand some amount of backflow impingement that occurs during entry. In the pre-deployed configuration, the hard petals of the outer portion of this hybrid shield cylindrically encase the SSRV and act as the vehicles protective shroud through all phases of flight preceding Mars atmosphere entry (i.e., including aerobraking or entry and landing).

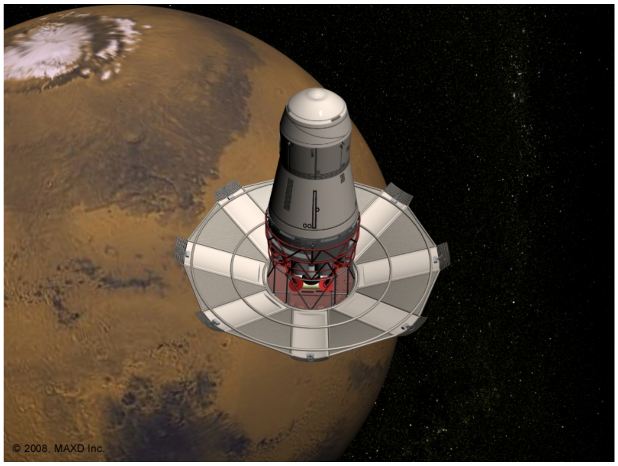

Figure 23. Protective Entry Aerodecelerator (PROTEA).

Following the SSRV propulsive orbital capture and orbit maintenance maneuvers, the shield is extended and then inflated in support of any further needed aerobraking and final atmosphere entry. As the hard shell panels of PROTEA unfold, petal interconnect structures inflate to fill the gaps between the hard portions of the shield to provide a structurally ridged framework. The full PROTEA aeroshell can have slightly variable maximum sizes depending on length of associated launch shroud, with an average being roughly 16 meters in diameter.

As this architecture is primarily designed to target northern Martian latitudes below the minus two kilometer datum, the descent takes advantage of the maximum aerial column available during entry and landing (though conceivably landings could be accomplished in the Hellas basin or given extremely high accuracy, a landing on the floor of Vallis Marinaris). As the vehicle decelerates through the upper atmosphere, parachutes (i.e., large high Mach and subsonic canopies) are incorporated to further decelerate the vehicle and lend 'hang-time' during PROTEA shield separation (i.e., the parachutes assist in allowing vehicle to separate from the aero-shell). It has also been considered that during both nominal or emergency shield separation the possibility exists for the directed gambling of the descent engines towards the inflatable panels and using them as blow-though sections in support of shield separation at supersonic velocities. Finally, after separating from the entry heat shield and parachute systems the propulsion system actively controls the remainder of descent to touchdown.

G. The SSRV Earth Return Propulsion Stage

Connected between the upper portion of the GRDAC and lower section of the SSRV, this portion of the architecture is a functional mix of the Apollo lunar architecture (i.e., the Lunar Module ascent stage and Service Module propulsion stage) in that it is designed as the primary propulsive stage for Mars ascent as well as the Trans-Earth Insertion (TEI) and course correction burns. Mid course correction burns will be accomplished using cold-gas systems. The structure for this stage is also similar to the SPS outlined earlier, whereby it provides some supporting system functionality for the SSRV vehicle stack (e.g., high rate data communications) until it is jettisoned around the time of Earth orbit entry (see Figure 24).

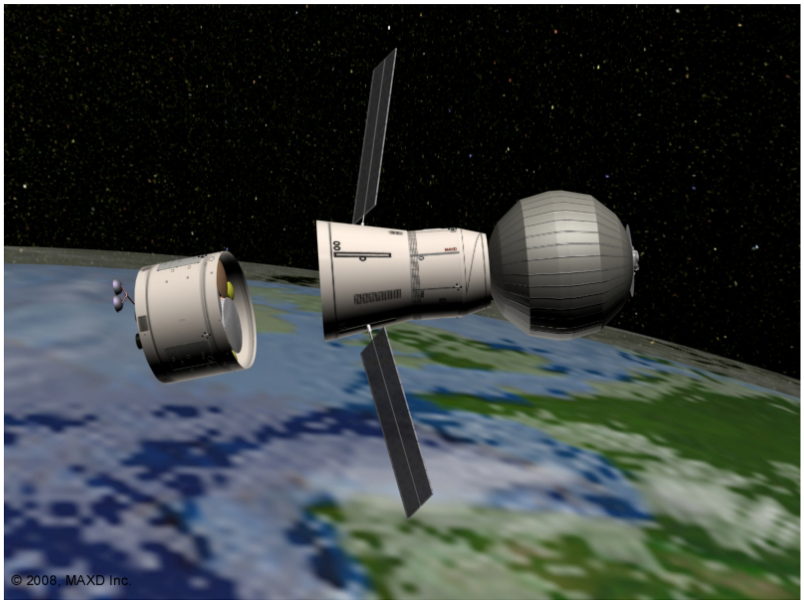

Figure 24. SSRV separating from Earth return stage and entering Earth orbit.

The methane and oxygen propellant for the ascent portion of the SSRV is produced insitu, on the Martian surface using the ISPP reactors. As all the fuel is consumed during the TEI burn, there is no need for cryogenically active storage capabilities. ISPP uses the ambient Martian atmosphere to produce (e.g., the Sabatier process) methane (fuel), which when combined with Mars mined and produced (e.g., through hydrolysis) oxygen (oxidizer) provides propellant for all return space flights and ground operations. Given that initial missions are outfitted with 6 tones of hydrogen feedstock, the total amount of methane/oxygen propellant produced amounts to roughly 100 tones per ISPP processor[4,5]. Additionally, nearly 18 tones of water are produced per ISPP, which will be stored in the empty PHV water-wall structures. As mentioned previously, hydrogen feedstock transported from Earth for the initial missions is later supplanted by supplies mined from near surface Martian reservoirs.

H. The Planetary Habitat Vehicle (PHV)

The PHV, shown in Figure 25, is the primary habitat for the surface of Mars and other planetary bodies. It is designed to support long duration habitation and continuous EVA operations based on the following primary philosophical drivers: radiation protection and dust mitigation. The interrelation of the design is based on material functional integration, safety and redundancy through compartmentalization and vertical contamination isolation.

The PHV stands approximately 11 meters in height and is a maximum of 8 meters in diameter. The habitat is divided into six airtight and isolatable compartments that include Deck 1a, 1b, 2a, 2b, the Interdeck Transfer Tube (ITT) (see Figures 26 and 27), and the airlock. The airlock, discussed in a following section, is further compartmentalized into three operationally distinct chambers designed around dust mitigation and containment (i.e., the External & Internal Surface Operations Rooms (ESOR and ISOR) and the External Operations Antechamber (EOA)), which support frequent, rapid and efficient EVA ingress and egress operations.

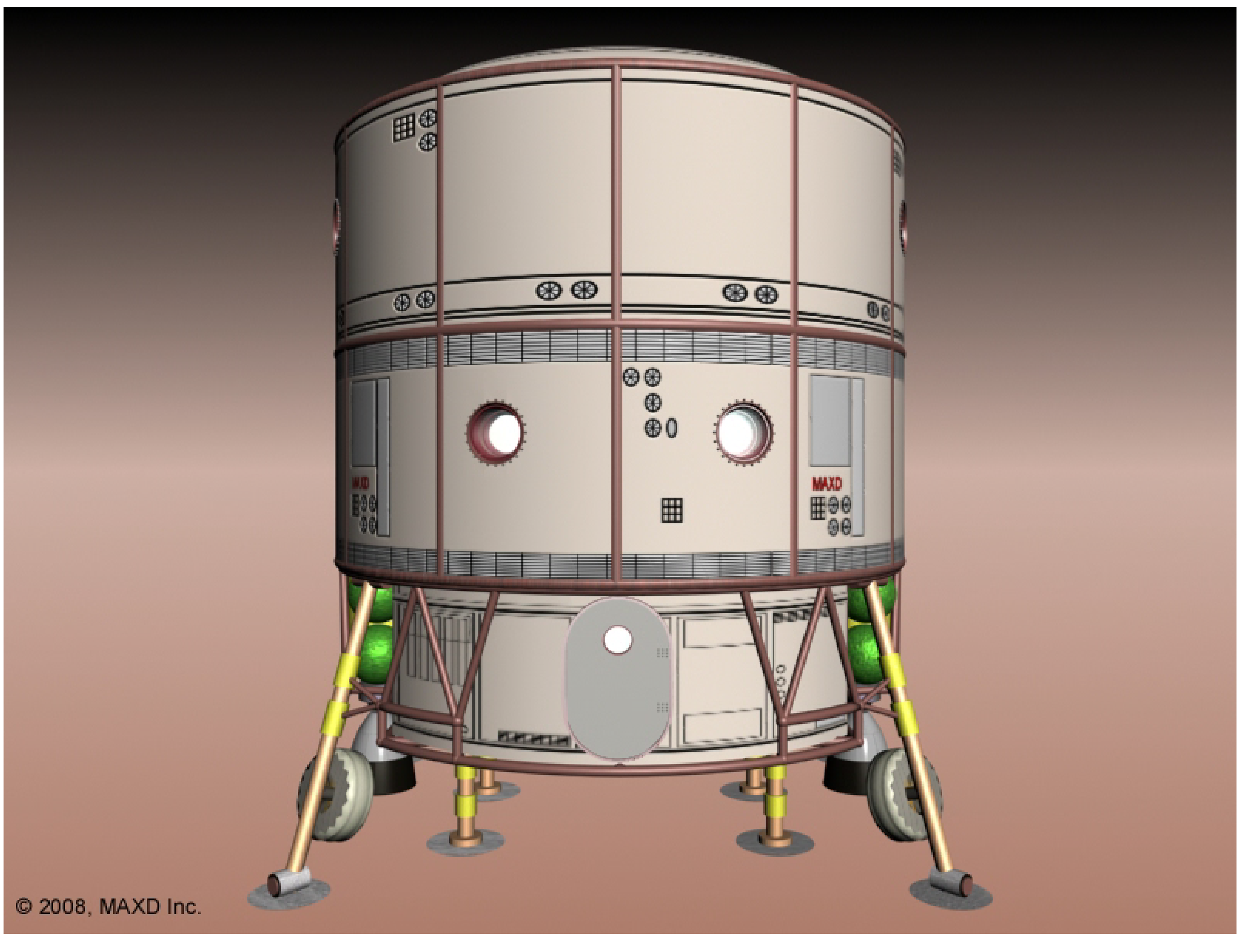

Figure 25. PHV external front view.

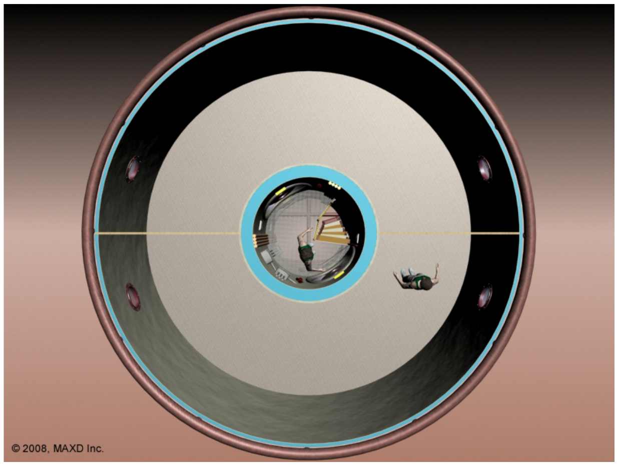

Figure 26. PHV compartment overhead view.

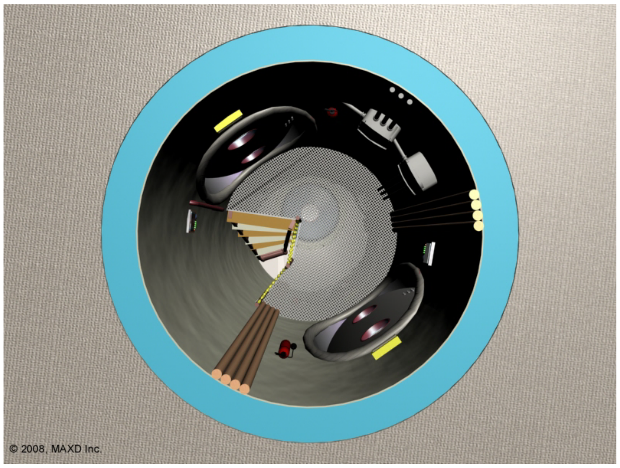

Figure 27. PHV Interdeck Transfer Tube (ITT) overhead close-up.

In general, this architecture directly supports structural isolation, mitigation and containment during conditions of emergency depressurization, fire or habitat contamination. As with the SSRV above, this vehicle's nominal operating pressure is 10.10 psi, which supports frequent EVA operations given a suit pressure that is somewhat greater than those currently used for ISS or the Space Shuttle. The airtight portion of the hull is made from two, approximately 2 mm aluminum waffle-panels separated by an internal honeycombed bladder structure that stores and funnels water. Five primary hatches branch from the ITT to all deck compartments, and three hatches in the airlock provide vehicle compartment isolation. One additional hatchway is located on the lower deck outer hull (Deck 2b) which supports a second story inflatable Interhab-Causeway (IC), which facilitates mobility between modules as base facilities are expand, integrated and collocated. The outermost hull is constructed using various layers including aluminum panels, Beta cloth, Mylar, Nextel, Kevlar, Aluminum Mesh and polyethylene. These layers would be optimized for the following basic functions given this models available 0.24 m thickness: structural integrity, pressure hull containment, micrometeorite protection, thermal insulation and radiation mitigation.

I. The Bi-Deck Toroidal Hab

This two deck structure is the primary habitation area and is designed around a water containment shell that acts as both a storage vessel for water and more importantly as the primary radiation shield for the vehicle during interplanetary transit and long duration surface habitation (see Figures 26 and 27 above & 28). Dual, compartmentalized, overhead conjoining water cylinders form the PHV's inner and outer hull. The outer hull is comprised of a multi-shock micrometeorite shield and ensconced within the PHV primary truss structure.[52] There are a minimum of four isolatable water sections in each hull (internal and external) that are outfitted with interconnected plumbing, filtration and pumps. As with the IRaSS in the CCEC above, the design exploits human behavior in that the most protected area, the ITT core, is used as the microgravity sleep chamber (i.e., reducing total radiation exposure by 1/3rd) and radiation bunker for both the interplanetary portion of the flight and during surface habitation. It is only within the ITT that the crew would have the maximum radiation shielding and therefore protection. Interplanetary transit radiation doses for crews using the PHV are similar to those estimated for the CCEC as presented previously. Figure 29 shows the HZETRN analysis for the estimated PHV radiation environment within the ITT core hull, and the analysis again predicts a very low, free space dose rate of 0.042 cSv/day and 0.13 cSv for SPEs (blue tracts in figure). Surface doses using standard shielding at the 95% CI are

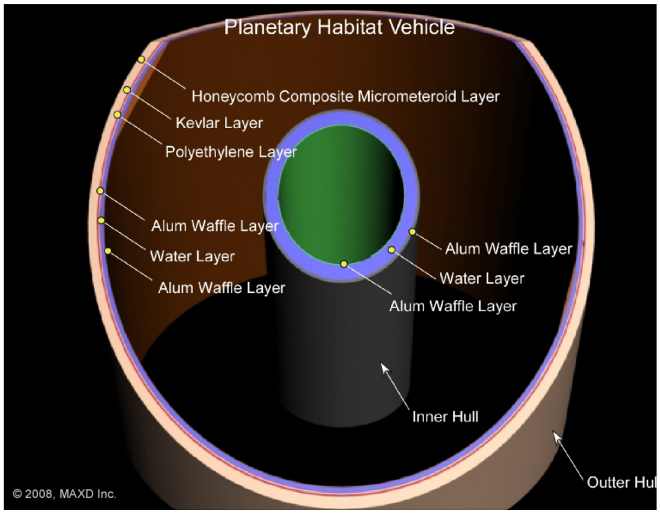

Figure 28. PHV wall cutout.

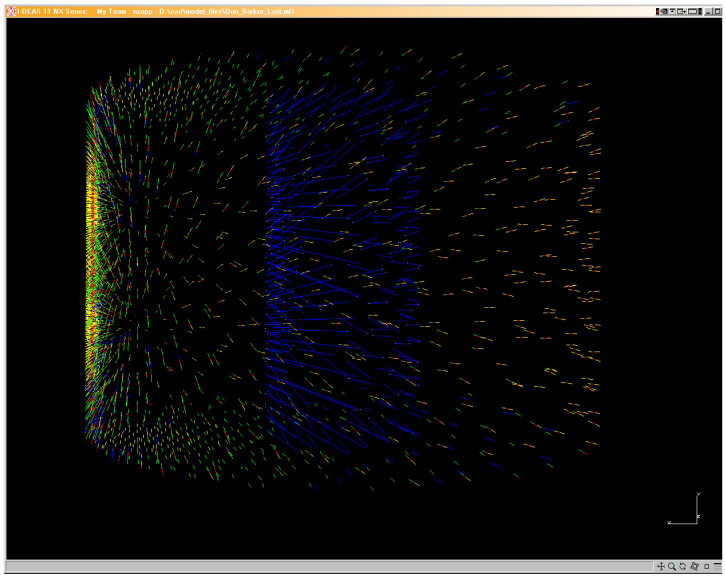

Figure 29. PHV hull radiation track analysis.

estimated to be roughly 0.21 cSv/day. A conservative result of using this water wall approach is to decrease the nominal predicted surface mission exposure by nearly 35 cSv (35 Rem) to 83 cSv (83 Rem) assuming a total 118 cSv (118 Rem) exposure based on a 560 day stay time and 10 g/cm^2 average aluminum structural shielding at Solar Minimum. This result remains below the annual 50 cSv NCRP limit. Again, it is important to remember that overall PHV mass and structural components have been only marginally considered in this rough dose estimate, and therefore the true value should prove to be somewhat lower.

The radiation protection extent within the core or the PHV or the ITT is 0.25 meters of water. The containment of this water is divided between the inner and outer hulls, where the bladder thicknesses are 0.05 and 0.2 meters respectively. The water tanks with nearly 7 m^3 of volume can be isolated into either potable, technical or waste (e.g., processed urine). As opposed to the water usage mentioned above for the SSRV, the PHV will maximize all water processing and recycling capabilities.

The total mass of water at launch is approximately 7,011 kg, which supports both crew consumable needs and radiation shielding throughout the transit flight. Approximately 5,500 kg of the remaining water (waste and potable) will be vented overboard just prior to Mars entry and landing. The remaining water is transferred to the lower inner hull, with the effect of focusing and lowering the center of gravity during entry and landing. The piloted PHV descends with roughly 1,752 kg of water or 15% of that needed by a 6-person crew for the estimated 560-day surface stay. All unpiloted PHVs would land on the surface with a certain minimal amount of water that is offset with an increased amount of dry consumable mass. Once on the surface, crews will transfer and consolidate water from operating ISPP facilities (e.g., roughly 30,000 kg from SSRV 1 and 2) into the empty PHV tanks. Again, the excess water supplied during the piloted Launch Group phase above the six person consumption requirement is considered to be a transient-replenishable radiation shielding; and such an impingement to launch mass is deemed a necessary burden in order to provide long duration crew radiation protection.

The PHV has two decks that are 2.7 meters in height, and divided in half to provide four isolatable chambers. Each half deck, or chamber, has roughly 21 m^2 of floor space giving a total empty volume of 57 m^3. The two deck empty volume of the PHV is roughly 230 m^3. The ITT tube provides an additional 3 m^2 of room per deck (~ 8 m^3 total) and is primarily used as a passageway, the deck transfer ladder and microgravity sleep area (see Figures 30 and 31). It is assumed that roughly 1/3 of the total interior volume or 82 m^3 is available for use by the crew on a

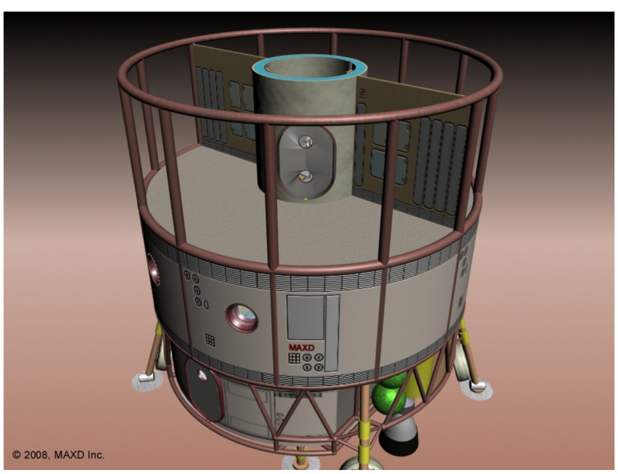

Figure 30. PHV oblique view without upper-deck outer-hull.

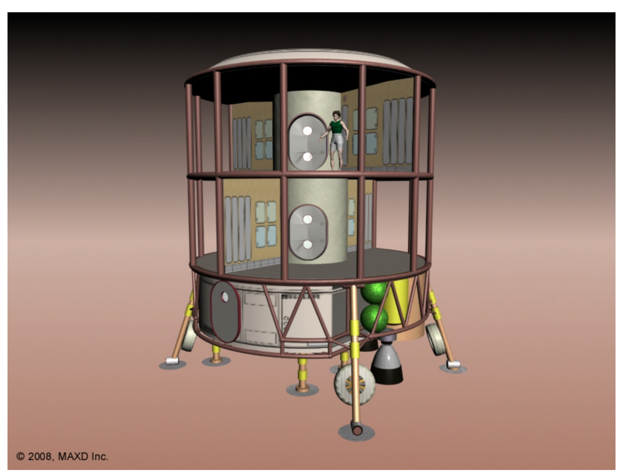

Figure 31. PHV without outer-hull - both decks.

daily basis.[11] Human performance driven habitable volumes sizes have been shown to peak around 11.3 m^3 per crewmember for mission durations longer than 6 months and studies have broadly emphasized using a minimum of 20 m^3 per crewmember for similar and longer durations;[11,41] Yet, this design presumes that most any confined-space issues afforded by habitat design will be psychology elevated by several issues including the availability of multiple PHVs and frequent surface excursions once on Mars. When compared to historical maritime accounts of cramped conditions and given the psychological readiness of Mars crews, this issue seems to be manageable.[25] The total empty volume of a single PHV, including a portion of the airlock presented below, is 249 m^3 or roughly 83 m^3 of usable space, which provides a little over 13 m^3 per crewmember per PHV. Establishing the first base, given a nominal mission as outlined, by moving and connecting the propositioned PHVs with the first piloted PHV group, presents the crew over 1,000 m^3 of available living and working space.

Interior space allotment and outfitting is divided based on related functionality and ascribed cleanliness levels. All chambers are configured with command and communication modules and sensor arrays that help monitor and control the health of the habitat. Four windows are provided on each deck and are together oriented to provide 360 degree viewing and are positioned to direct light to enter through the ITT hatches in case of lighting system failure. The top deck includes the crew quarters, storage, wardroom, galley, office and recreational space. The lower deck contains the medical-exercise facility, hygiene-shower-laundry facility, laboratories and the Interhab-Causeway hatch (used to connect a second floor inflatable tunnel to an adjacent PHV). This paper does not further address interior design as there are innumerous configurations for space utilization; yet, the use must ultimately adhere to the space available and the overall design requirement of controlling the movement of dust and contaminants through the interior.

J. The Tri-Con Planetary Airlock

The Tri-Con Planetary Airlock is a unique airlock designed explicitly for planetary surface operations (see Figures 32 and 33) by specifically mitigating surface dust and particulate contamination and reducing atmospheric consumable usage and waste. This airlock is divided into three primary compartments whose operation is based on clean-room pressure gradients, atmospheric resource conservation, safety and efficiency of operations. This airlock presumes that the final design of the surface suit has a working pressure of somewhere between 5 and 7 psi as well as having a rear entry hatch similar to the Russian Orlan-M EVA suit or related deep-sea diving suits.[53]

A total of four hatchways, three unique chambers and one gravity-gradient ascent (i.e., a ladder ascent to upper decks inhibiting dust propagation) are used sequentially as the primary means of dust and contamination control within the PHV. As this structure is considered the work horse of planetary surface operations it is specifically designed mitigate dust impingement during regular and long duration use given the previously mentioned characteristics of the Martian dust. For the remainder of this section, each room will be described in order of being accessed as if upon returning form surface EVA operations.

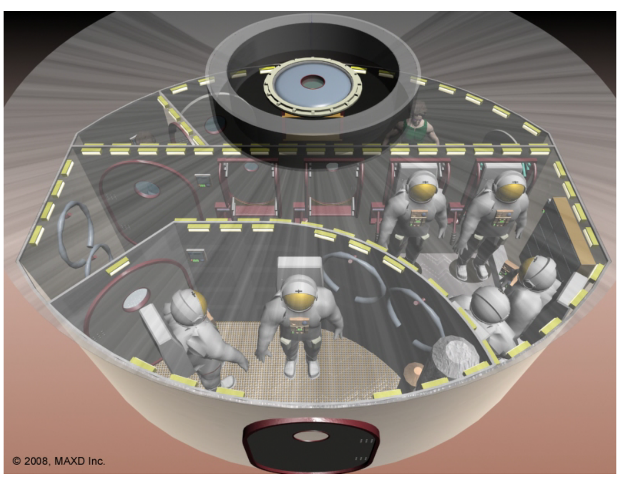

Figure 32. Airlock oblique overhead view with ghosted PHV decking.

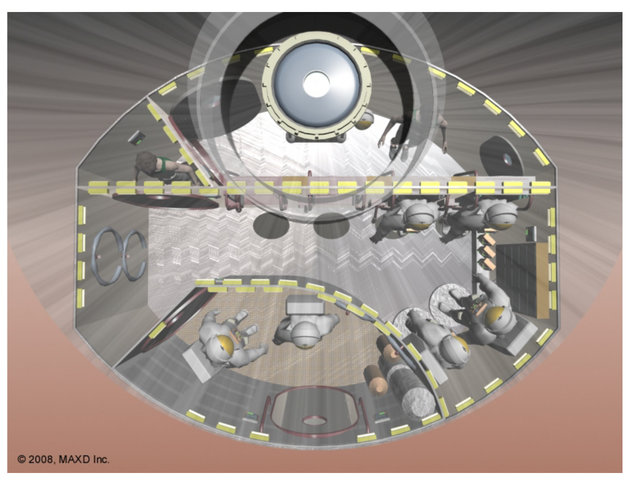

Figure 33. Airlock overhead view.

Figure 34. External Surface Operations Room (ESOR).

The first room, the External Surface Operations Room (ESOR), is accessed through the PHV's external airlock hatch (see Figure 34). Crews enter into a small ovoid chamber (~ 11 m^2) capable of holding 4 standing adults or two standing and one recumbent crewmember during an emergency mobilization. After the crew enters this chamber, they re-pressurize to approximately 9.1 psi and begin decontamination. This re-pressurization occurs in a 100% CO_2 atmosphere attained from filtered and compressed native Martian air. The purposes for using a CO_2 atmosphere is to both keep the suits and their components largely in the environment that they are intended to serve, and to ultimately save valuable air resources during pressure change cycles. Yet, if needed, as in an emergency, the room could be pressurized using the standard PHV oxygen atmosphere.

According to terrestrial research, nearly 70% of the dust inside buildings is transported from outdoors[54] and given a need to perform daily surface excursions, living on Mars is expected to be no different. In order to combat the pervasive Martian dust, several dust removal techniques are employed sequentially to protect the PHV and clean the surface suits during ingress. These include hatchway electrostatic barriers, floor electrostatic and electromagnetic gratings, high pressure CO_2 "gas-wash" and vacuum cleaners. These systems are both manual and automatic (e.g., activated as the crewmember ingresses the associated area). A nominal buddy system cleaning procedure is estimated to take approximately 20 minutes for four crewmembers. During the ingress and suit cleaning period, plug-in support panels provide connections for communication, power, air and water between the suit and PHV. Before egressing this chamber inwards, crews increase the pressure to 9.8 psi to match internal levels.

Figure 35. External Operations Antechamber (EOA).

The airlock chamber's second hatch leads into the External Operations Antechamber (EOA) (see Figures 35 thru 37). This chamber nominally operates with the same 100% CO_2 atmosphere as used during recompression in the ESOR as mentioned above; yet, this room maintains a constant 9.8 psi pressure. This elongated room covers 10 m^2, contains 4 rear-entry suit hatch docking stations[55,56] and two sealed-suit storage

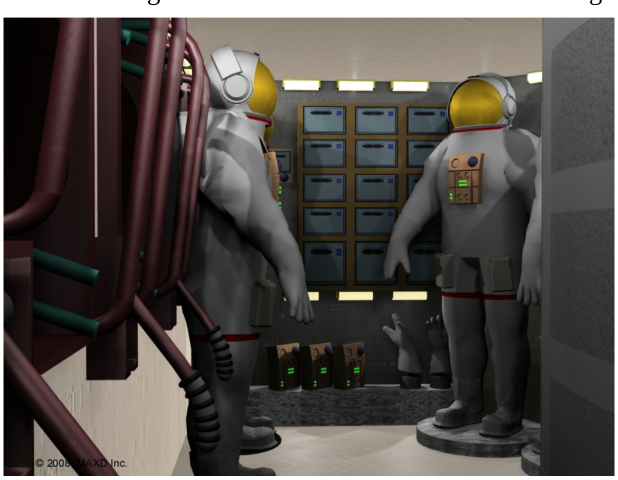

Figure 36. EOA views 2.

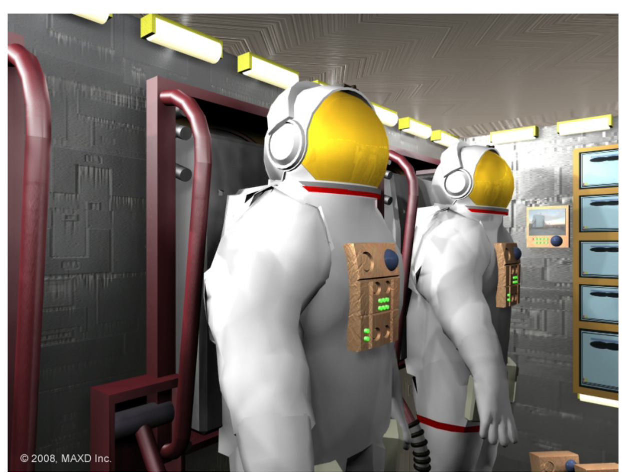

Figure 37. EOA view 3.

racks. Therefore, suits and other surface hardware are maintained in a constant Martian atmospheric environment albeit a higher pressure. The suit docking stations have electrostatic dust guards and are designed so that a crewmember can align, dock and seal the suit to the suit-dock without assistance. Once docked, the proximal casing surrounding the suit-pack and hatch is washed with a high pressure CO_2 stream. Finally the suit-dock is vented of the remaining CO_2, pressurized with ambient PHV cabin air and the crewmember egresses the suit into the next compartment by manually opening the suit-dock and surface-suit hatches. The purpose of this final process is two fold; first it provides a final cleaning of the suit-pack and suit hatch cover and second it removes remaining dust and CO_2 form the suit-backpack enclosure prior to venting to the 10.10 psi cabin air environment. Expected time for complete egress of the suit is estimated to be less than 10 minutes per crewmember. The time estimate for crews exiting the PHV will be somewhat longer, but only due to pre-breath and pre-EVA suit check operations as needed. Finally, the EOA chamber contains additional cleaning systems (e.g., vacuums), storage lockers, containers for suit spare parts and accessories, and space for sample storage and processing (i.e., for transport to the laboratory facilities). This chamber may also be pressurized using normal PHV oxygen supplies on an as needed basis.

The final portion of the airlock, shown in Figure 38, is the short-sleeve Internal Surface Operations Rooms (ISOR) (~11 m^2), which connects to the ITT via a ladder through PHV hatchway number five. This chamber normally operates at the PHV nominal pressure of 10.10 psi and is considered the final interface between the Martian and PHV's habitable environments. A final level of dust protection is passive and consists of the gravity gradient that crews must ascend in order to access the ITT and the rest of the PHV. Again, the ISOR is entered following an EVA via the rear entry suit-pack hatch that interfaces with the four aft wall docking stations of the EOA. Giving crews the capability to individually ingress or egress surface suits through this suit-hatch interface system directly enhances efficiency and safety.

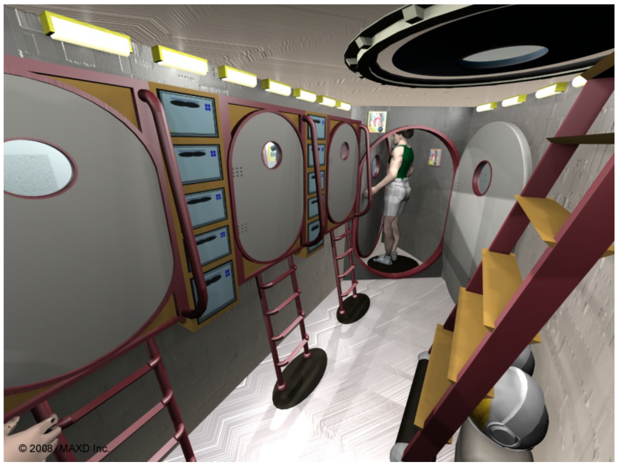

Figure 38. Internal Surface Operations Rooms (ISOR).

The ISOR as opposed to the rest of the PHV will be uniquely tested to support limited hyperbaric operations in support of decompression sickness (DCS) emergencies. In addition, the room may also be sealed thus allowing crewmembers to perform EVA pre-breath operations between 26.5% and 28.5% oxygen, as needed depending on final surface suit design.

The ISOR and the EOA have one more unique vestibule that is used by crewmembers performing housekeeping, suit maintenance or sample transfer tasks and is mentioned here because this volume usually remains at PHV nominal pressure until used. A single person, double-door vestibule separates and provides an interface between the ISOR and EAO. This vestibule permits movement between compartments or allows contingency pass-through of injured crewmembers. A unique operation afforded by this vestibule is that crews can access the 9.8 psi EOR compartment rapidly and without virtually any impact to atmospheric gas provisions. This is accomplished by having crews work inside the 100% CO_2 environment for predefined periods while breathing PHV oxygen through protective face masks or ventilators. The hazard guide and material data sheets provided by the National Institute of Occupational Safety and Health[57] and others purport that no adverse effects come from exposure to the skin of gaseous CO_2 for extended durations. Crewmembers will perform activities by donning a static-guard dust garments (see Figure 39) and use quick-connect/disconnect air ports and face covering positive pressure regulators similar to those used for hypobaric chambers, terrestrial clean room facilities or bio-chemical protection. The use of these suits


Figure 39. EOR maintenance ingress suits (examples of DuPont personal-protection gear, Ref. 58).

allows crews to work and perform repairs in a near standard pressure environment (i.e., affording nearly normal dexterity) while providing protection from Martian surface dust and contaminants.

Finally, in addition to its transfer interfaces, the ISOR acts as a general machine shop and work area which provides some storage for EVA hardware, tools and material supplies (e.g., nuts, bolts and copper tubing). It is the unique and overall holistic nature of this design that supports and enables frequent, rapid, safe and efficient EVA ingress and egress operations. This is of utmost importance since surface operations are deemed to be the most important aspect of human exploration on the surfaces of Mars.

K. The Generic Removable Descent Cradle (GRDC)

Unlike the SSRVs GRDAC, the GRDC of the PHV only acts as a cradle that provides structural support for the PHVs descent propellant system (see Figures 30 and 31 above) and structures. The PHV's struts (including mobilization motors and systems) and support columns encage the GRDC and provide support, anchoring and thermal isolation for the PHV as it rests on the surface. The GRDC partially surrounds the airlock and attaches and interfaces at the bottom of the PHV's lower deck. Four descent engines and tankage providing roughly 6,400 kg of hypergolic bipropellant are used to safely land the PHV on the Martian surface following parachute deceleration and release. The propellant system, engines and tanks, are further designed to be easily removed from the cradle in sections by crews as needed or prior to placing the PHV's support struts in the final resting configuration. Further study is needed to assess whether these systems could be refurbished on the surface for use as spares and replacements for future launch vehicles. Otherwise, unusable hardware such as this should be addressed as part of the base wide waste management plan.

L. The PHV Hybrid Entry, Descent and Landing Shield

The PHV, herein, incorporates the same basic EDL structure used by the SSRV (the PROTEA) though it is larger in radius due to the larger size and mass of the PHV structure and associated launch shroud (see Figure 23 above). This structure still requires additional modeling and testing to be considered as a fully acceptable EDL system. It is important to reiterate, though, that the ballistic entry profile for this architecture is reduced from other aeroentry concepts as a result of the redundancy built into the Launch Group design; i.e., the combined PHV stack having a fully fueled SPS stage used to perform a retrograde orbit entry burn during Mars arrival in support of entry and landing.

4. Testing and Operations at the Moon or Space Station

The Lunar surface could serve as a working test bed and checkout environment for a majority of the systems and structures for both the PHV and SSRV designs (see Figure 40). The idea of building spacecraft for Mars and back-testing them at the moon was an explicit part of the Mars Direct plan[4,5] and still holds true if earnest human Mars exploration is to occur within current career time-spans (i.e., rather than building vehicles designed strictly for the moon and testing Mars analogue systems within them). The vehicles presented in this work would require only minor alterations in order to serve as permanent lunar structures and accommodate the 1/6 g surface environment. For example, the SSRV during pre-PHV missions, could serve, not only as the crew transfer vehicle but also as the primary short term (~ 3 to 6 months) habitat if the inflatable EEDIG were to be used throughout the mission or during the surface portion of the excursion. The

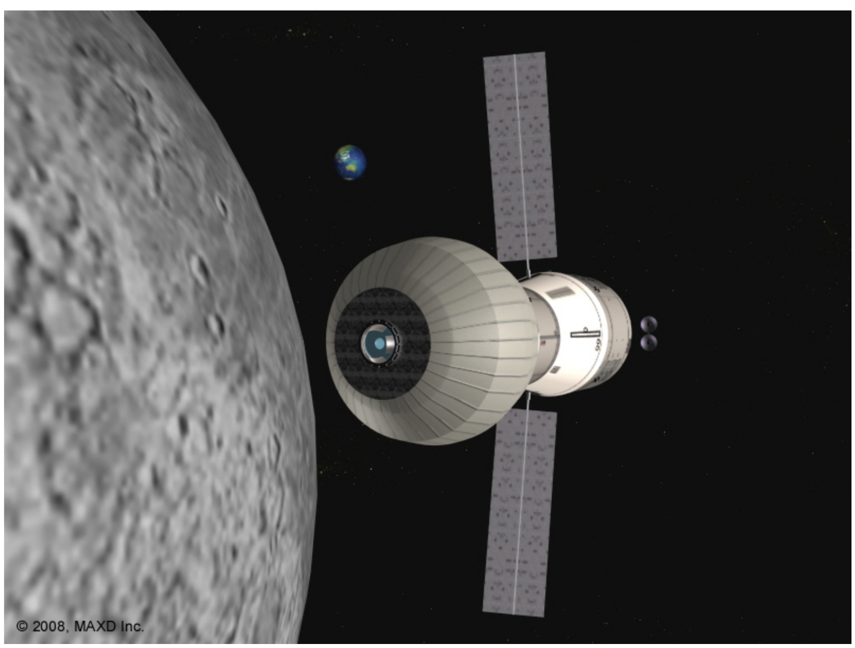

Figure 40. SSRV testing in lunar orbit.

SSRV could also support a larger crew (i.e., 4 to 6 persons) for the short duration lunar checkout and testing flights using the same sleep-shifting paradigm as the emergency Earth return scenario.

Another scenario is to use the CCEC in a stand-alone configuration to serve as a transfer vehicle to an Earth orbiting space station for either a 3 or 6 person crew; giving ample opportunity for lengthy space environment and operations shakedowns in a manner similar to that proposed for lunar testing. The main difference is that the TVAL and landing systems would not be required. In addition, the CCEC could be arranged to act as an assured crew return vehicle, providing transportation to a tertiary, definitive medical care facility for Earth orbit medical emergencies. A single patient connected to life support devices would be stabilized and transported recumbently either within or in the space located above the IRaSS water chamber.

The testing of overall vehicle designs either in conjunction with a space station platform or the lunar surface directly affords a rapid turnaround of lessons learned, thus enabling timely vehicle upgrades and enhancements. The advantage of the proximity of these locations is in the quick round trip capability that provides for easy and frequent re-supply, as well as rapid emergency response options. It is this author's opinion that such testing is the only valid and sustainable use of the lunar surface in preparation for Mars surface operations. Again, the most important concept presented herein is that these vehicles have been initially designed to go to Mars and are being tested on the Moon – build for Mars, fly at the Moon.

5. Discussion

Design is the first method of controlling and mitigating risk, insuring efficient and effective operations, and ultimately has been a driving factor in the development of this architecture. Integrating operations and program goals further this objective and ensure efficiency and mission safety. The exploration goal highlighted herein is the desire to attain self-sufficiency and permanent habitation on the surface of Mars by initiating the mining of water from various planetary reservoirs. A philosophical treatise was devised which embodies a ground up design that enables an evolving efficient and relatively cost effective strategy for mission implementation. Novel and derivative mission components take advantage of multiple resource utilization, enhanced system redundancy and multiple module production in order to reduce costs and increase mission redundancy and safety. Table 6 provides a list

summarizing the most prevalent contributions that this architecture has endeavored to offer as a means of satisfying the need for a consolidated ability and goal for the exploration of Mars by humans. Complete system

Table 6. The top ten characteristics defining the MAXD architecture.

Maximization of Earth to Mars transit and Mars surface radiation protection due to 0.25 meter water wall shielding structures for crew sleep and emergency sheltering.
Maximize the mitigation of planetary dust contamination to insure hardware longevity and habitat environmental safety.
Design performance enhancement, mission longevity and reduced mission risks due to the localization, acquisition and use of indigenous resources.
Enhanced EVA operations due to a robust airlock design that supports efficient daily surface operations with reduced atmospheric consumable loss.
Enhanced mission safety due to complete vehicle redundancy during piloted Earth to Mars transit in the Launch Group configuration.
Reduction in mass required for Mars aeorcapture/aeroentry shielding and enhanced realtime orbital flexibility as a result of a Launch Group configuration that provides for propulsive assist during Mars Orbit Insertion using space storable propellants.
Embedded limited duration/distance direct Earth return abort options due to the complete vehicle redundancy in propulsion as provided in Launch Group configuration.
Enhanced mission safety and reparability options during entire surface stay given that no architectural components are placed in Mars orbit and out of reach for hands on inspection prior to Earth return.
Rapid base infrastructure construction and evolution due to dual vehicle Launch Group flight rate.
No requirement for a previously emplace lunar outpost, though vehicles could be flight tested in the lunar environment.

redundancy has been provided through the Launch Group concept and thereby greatly enhanced mission safety. Given that each individual launch comprises of a complete vehicle system, any critical failure will not impact the mission chain by stranding or significantly delaying the full implementation of the mission. In order to enable the human exploration of Mars in the most efficient and rapid manner possible the testing of vehicles in various locations in parallel including the surface of the Moon has also been considered.

The design presented adheres to the primary premise of radiation protection and dust mitigation. Radiation shielding results, for this design, indicate that the total round trip exposure would be lower than the calculated 231 cSv as determined by the sum of the three legs presented above (74 cSv + 83 cSv + 74 cSv = 231 cSv). Over a 1000-day mission this would be roughly 94 cSv above the NCRP recommended dose over the same period of time and exceeds all but the oldest category male BFO career limits. The value, though, is roughly 3 times less than previous architectures which only consist of standard space vehicle structures for radiation protection. Again, it is important to remember that this initial value is due to the novel water-bearing structures only and therefore higher than the actual dose for this architecture given that the intervening vehicle structures, hardware and mass were not considered in the gross shielding calculation. Given a conservative approach the combination of our current biophysiological understanding of radiation hazards associated with a Mars mission along with the most cost effective means of providing protection and countermeasures, it is suggested that the early phase missions be carried out by crewmembers who are 50 years of age or older, given present guidelines, until reduced transit times can be achieved. Dust is observed to be ubiquitous and highly mobile and will therefore pose an inconvenience and a hazard, physiologically, mechanically and electrically, to long duration surface habitation. This design has attempted to mitigate any such hazard by designing a unique three-compartment airlock that uses native resources and sequential staging and cleaning operations. Efficient and rapid crew access to surface EVA operations is also afforded by an integrated, single person rear-entry suit cabin design.

For the first time in nearly 35 years, the US human space flight program has been able to address and focus resources on the exploration of a destination other than low Earth orbit. Yet, major concerns and questions remain in that the fundamental drivers for this endeavor have not been adequately focused in either form or function. Why are we going? What are we doing? Where are we going? And how long will we be there? Any exploration architecture and venture must sufficiently address these questions and others if it is to stand the test of time. This will, in this author’s opinion, most likely be the cause of deferring the exploration of Mars to the generation our children’s children or further.

The intent herein was not to reinvent the wheel, as it now exists, but to consider the wheel and ultimately improve upon its form and function given the possibility of a new road. In roughly 55,000 years of modern human technical evolution and expansion, we have arrived at a new crossroad with the ability to expand our species into a multi-world civilization; thus helping to insure our long-term survival. The planet Mars accommodated us with two very close orbital alignments in nearly as many years. Normally separated by roughly 225 million km, Mars passed within 69.3 million km from Earth in the fall of 2005. Our next opportunity lies on the horizon, and our technology is standing by, ready to be tested at the moon. The only question that remains is whether or not our motivation and hubris will inhibit our establishing a permanent presence (the next best opportunity being 2018) on this majestic neighboring world. This author believes that no other human initiated event will have such a positive influence on the fabric of global society and that the directed exploration of Mars will be one of humanities final great enticing and cross-culturally motivating adventures. Sending humans to Mars to live will be our boldest adventure, and certainly the most widely experienced event in human history.

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7. References

[1]Shaler, N. S., “Nansen’s Heroic Journey,” *The Atlantic Monthly*, Vol. 79, Issue 475, Boston-Atlantic Monthly Co., pp 610-617, May 1897.

[2]Hanford, A. J., “Advanced Life Support Baseline Values and Assumptions Document,” JSC 47804, CTSD-ADV-484A, NASA JSC, Crew and Thermal Systems Division, Rev. A, 2004.

[3]Exploration Systems Architecture Study (ESAS), NASA-TM-2005-214062, www.sti.nasa.gov, November 2005.

[4]Zubrin, R. M. And Weaver, D. B., “Practical Methods for Near-Term Piloted Mars Mission,” *American Institute of Aeronautics and Astronautics*, AIAA-93-20898, 1993.

[5]Zubrin, R. M., Baker, D. A. and Gwynne, O., “Mars Direct: A Simple, Robust, and Cost Effective Architecture for the Space Exploration Initiative,” *American Institute of Aeronautics and Astronautics*, AIAA-91-0328, 1991.

[6]Weaver, D. B. and Duke, M. B., “Mars Exploration Strategies: A Reference Program and Comparison of Alternative Architectures,” *American Institute of Aeronautics and Astronautics*, AIAA-93-4212, 1993.

[7]Drake, B. G., editor., “Reference Mission Version 3.0 Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team,” NASA/SP-06107-ADD, Lyndon B. Johnson Space Center. June 1998.

[8]Claybaugh, W. Garriott, O. K.,Garvey, J., Griffin, M., Jones, T. D., Kohlhase, C., McCandless II, B., O’Neil, W. and Penzo, P. A., “Extending Human Presence into the Solar System,” The Planetary Society Independent Study,

http://planetary.org, July 2004.
[9]Hoffman. S. J., editor., "The Mars Surface Reference Mission: A Description of Human and Robotic Surface Activities," NASA/TP—2001–209371, Science Applications International Corporation, December 2001.
[10]Cooke, D. R., Weary, D. P. and Kaplan, D. I. "Analysis of the Synthesis Group's Mars Exploration Architecture," Lunar and Mars Exploration Program Office, Code XE, Johnson Space Center, October 25, 1991.
[11]Larson, W. J. and Pranke, L. K., "Human Spaceflight: Mission Analysis and Design'" McGraw-Hill, Inc., New York, Space Technology Series, 2000.
[12]Humble, R. W., Henry, G. N. and Larson, W. J., "Space Propulsion Analysis and Design," McGraw-Hill, Inc., New York, Space Technology Series, 1995.
[13]Feldman, W. C., Perttyman, T. H., Maurice, S., Plaut, J. J., Bish, D. L., Vaniman, D. T., Mellon, M. T., Metzger, A. E., Squyres, S. W., Karunatillake, S., Boynton, W. V., Elphic, R. C., Funsten, H. O., Lawrence, D. J. and Tokar, R. L., "Global distribution of near-surface hydrogen on Mars," *Journal of Geophysical Research*, Vol. 109, 2004. doi: 10.1029/2003JE002160.
[14]Mars Exploration Program Analysis Group (MEPAG)., "Mars Science Goals, Objectives, Investigations and Priorities: 2006," NASA, February 10, 2006.
[15]Chamitoff, G., James, G. and Barker, D., "Martian Resource Locations: Identification and Optimization," *Acta Astronautica*, 53rd International Astronautical Congress, IAC-02-IAA.13.3.07, 2002.
[16]James, G. Chamitoff, G. and Barker, D., "Resource Utilization and Site Selection for a Self-sufficient Martian Outpost," NASA/TM-98-206538, Lyndon B. Johnson Space Center, April 1998
[17]Alliant Techsystems (ATK), Shuttle Derived Heavy Lift, http://www.atk.com/, 2006.
[18]Central Research Institute of Electric Power Industry (CRIEPI), "The Sodium Cooled Small Sealed Fast Reactor (4S) with Non-refueling," http://criepi.denken.or.jp/en/e_publication/2004.html, 2004.
[19]Buckey, J. C. Jr., "Space Physiology," *Oxford University Press, Inc.*, Oxford, 2006.
[20]Whedon, G. D. and Rambaut, P. C., "Effects of long-duration space flight on calcium metabolism: Review of human studies from Skylab to the present," *Acta Astronautica*, 58, pp. 59-81, 2006, doi: 10.1016/j.actaastro.2006.03.074.
[21]Bioastronautics Roadmap: A Risk Reduction Strategy for Human Space Exploration. NASA/SP-2005-6113, http://bioastroroadmap.nasa.gov/index.jsp, February 2005.
[22]Fong, K., "The Limits of Medicine,"*British Medical Journal*, Vol. 329, p. 1441-1444, December 2004.
[23]Sawin, C. F., Hayes, J. Francisco, D. R. and House, N. , "Considerations for development of countermeasures for physiological decrements associated with long-duration space missions," *Acta Astronautica*, 2006, doi: 10.1016/j.actaastro.2006.09.029.
[24]Horneck, G., Facius, R., Reichert, M., Rettberg, P., Sebolt, W., Manzey, D., Comet, B., Maillet, A., Preiss, H., Schauer, L., Dussap, C. G., Poughon, L., Belyavin, A., Reitz, G., Baumstark-Khan, C. and Gerzer, R., "HUMEX, a study on the survivability and adaptation of humans to long-duration exploratory missions, part II: Missions to Mars. Advances in Space Research," *COSPAR*, Vol. 38, 2006. doi: 10.1016/j.asr.2005.06.072.
[25]Wickman, L. A., "Human performance considerations for a Mars mission," *IEEEAC*, paper #1080, Ver. 5, 2005.
[26]Baker, D., "The History of Manned Space Flight," *Crown Publishers*, Inc., New York, 1981.
[27]Wilde, D., Walther, S. and Gmbh, A., "Inflatable reentry and descent technology (IRDT) – Further developments," *2nd International Symposium of Atmospheric Reentry Vehicles and Systems*, Arcachon, France, 26-29 March 2001.
[28]Whitehead, J., "Propulsion engineering study for small-scale Mars missions," Lawrence Livermore National Laboratory & NASA Ames, UCRL-CR-122442, September 12, 1995.
[29]Endurance Imagery and History, Frank Hurley, http://www.kodak.com/US/en/corp/features/endurance/; developed by http://www.secondstory.com/.
[30]Edgett, K. S. and Christensen, P. R., "The Particle Size of Martian Aeolian Dunes," *Journal of Geophysical Research*, 0148-0227, Volume 96, December 1991.
[31]Calle, C. I., "Measuring Electrostatic Phenomena on Mars and the Moon," *Proceedings to The Institute of Electrostatics*, Japan Annual Meeting, Tokyo, Japan, September 2001.
[32]Farrell, W. M., Smith, P. H., Delory, G. T., Hillare, G. B., Marshall, J. R. Catling, D., Hecht, M., Tratt, D. M., Renno, N., Desch, M. D., Cummer, S. A., Houser, J. G. and Johnson, B., "Electric and Magnetic Signatures of Dust Devils form the 2000-2001 MATADOR Desert Tests," *Journal of Geophysical Research*, 03004, Vol. 109, 2004.
[33]Cantor, B. A., James, P. B., Caplinger, M. and Wolff, J. J., "Martian Dust Storms: 1999 Mars Orbiter Camera Observations," *Journal of Geophysical Research*, 23653, Vol. 106, 2001.
[34]Landis, G. A. and Jenkins, P. P., "Dust on Mars: Materials Adherence Experiment Results from Mars Pathfinder," *IEEE*, 26th PVSC, 0-7803-3767-0, Volume 97, Sept. 1997
[35]Landis, G. A., "Mars Dust Removal Techniques," *AIAA Journal of Propulsion and Power*, Volume 14, No.1, 126-128, Janurary 1998.
[36]Mazumder, M. K., Calle, C. I., Pruessner, K., Sharma, R., Srirama, P. K., Immer, M. A. C., Trigwell, S., Clements, J. S., Biris , A. S., Mantovani, J. G. and Chen, A., "Research Needs in Electrostatics for Lunar and Mars Space Missions," *IEEE*, 0-7803-9208-6, Volume 06, 2005.
[37]Biris, A. S., Saini, D., Srirama, P. K., Mazumder, M. K., Sims, R. A., Calle, C. I., and Buhler, C. R., "Electrodynamic Removal of Contaminant Particles and its Application," *IEEE*, 0-7803-8486-5, Vol. 04, 2004.
[38]Calle, C. I., Buhler, C. R., Mantovani, J. G., Clements, J. S., Chen, A., Mazumder, M. K., Biris , A. S. and Nowicki, A. W., "Electrodyanmic Shield to Remove Dust from Solar Panals on Mars," *Proceedings of the 41st Space Congress*, Cape Canaveral, Florida, April 27-30, 2004.

[39]Volger, A., "Modular Inflatable Space Habitats," *First European Workshop on Inflatable Space Structures*, ESA/ESTEC, Noordwijk, The Netherlands, May 2002.

[40]Marchetti, M., Cioeta, M., Vasta, E., Speranza, C., Calabrese M. and Tuninetti, C., "New Concept of Inflatable Modules for the International Space Station," 2nd European Workshop on Inflatable Space Structures, ESA/ESTEC, Tivoli, Italy, June 2004.

[41]Wilson, J. W., Nealy, J. E., Schimmerling, W., Cucinotta, F. A. and Wood, J. S., "Effects of Radiobiological Uncertainty in Vehicle and Habitat Shield Design for Missions to the Moon and Mars," NASA Technical Paper 3312, Langly Research Center, August 1993.

[42]Wilson, J. W., Cucinotta, H. T., Simonsen, L. C., Shinn, S. A., Thibeault, S. A. and Kim, M. Y., "Galactic and Solar Cosmic Ray Shielding in Deep Space," NASA Technical Paper 3682, Langley Research Center, December 1997.

[43]Kim, M. Y. Wilson, J. W., Thibeault, S. A. Nealy, J. E., Badavi, F. F. and Keifer, R. L., "Performance Study of Galactic Cosmic Ray Shield Materials," NASA Technical Paper 3473, Langley Research Center, November 1994.

[44]Saganti, P. B., Cucinotta, F. A., Wilson, J. W., Simonsen, L. C. and Zeitlin, C., "Radiation Climate Map for Analyzing Risks to Astronauts on the Mars Surface from Galactic Cosmic Rays," *Space Science Reviews*, Volume 110, pp 143-156, 2004.

[45]Rapp, D., "Radiation effects and shielding requirements in human missions to the moon and Mars," Mars 2, 46-71, 2006; http://www.marsjournal.org/contents/2006/0004/ doi: 10.1555/mars.2006.0004

[46]Cucinotta, H. T., Premkumar, B. S., Xiaodong, H., Kim, M. Y., Cleghorn, T. F., Wilson, J. W., Tripathi, R. K. and Zeitlin, C. J., "Physics of the Isotopic Dependence of Galactic Cosmic Ray Fluence Behind Shielding," NASA Technical Paper 2003-210792, Johnson Space Center, February 2003.

[47]Cucinotta, H. T., Kim, M. Y. and Ren L., "Managing Lunar, Radiation Risks, Part I: Cancer, Shielding Effects," NASA Technical Paper 2005-213164, Johnson Space Center, 2005.

[48]National Counsel on Radiation Protection and Measurement (NCRP), "Radiation Protection Guidance for Activities in Low-Earth Orbit'" Report No. 132, December 31, 2000.

[49]Turner, J. E., "Atoms, Radiation, and Radiation Protection," McGraw-Hill, Inc., New York, 1992.

[50]Marcy, j. L., Shalanski, A. C., Yarmuch, M. A. R. and Patchett, B. M., "Material choices for Mars," J*ournal of Materials Engineering and Persormance* (JMEPEG). Vol. 13(2), April 2004. doi: 10.1361/10599490418479.

[51]WPP-200, Proceedings of the First European Workshop on Inflatable Space Structures. European Agency (ESA), 21-22 May 2002.

[52]Christiansen, E., "Meteoroid/Debris Shielding," TP.2003-210788, NASA Johnson Space Center, Houston, Texas, August 2003.

[53]Ruzakov, M. ed., "Orlan-M Spacesuit for Extravehicular Activity Workbook," Yu. A. Gargarin Cosmonaut Training Center, Star City, 2000.

[54]Textile Rental Services Association of America (TRSA), 2002. http://www.trsa.org/.

[55]Cohen, M. M., "Suitport Extra-Vehicular Access Facility," United States Patent 4,842,224, June 27, 1989.

[56]Culbertson, P., "Docking Fixture and Mechanism for a Protective Suit," Ames Research Center, http://www.nasatech.com/Briefs/Mar03/ARC14102.html. United States Patent 5,697,108, 2003.

[57]National Institute of Occupational Safety and Health (NIOSH), Dust Indoors, 2005. http://www.cdc.gov/niosh/npg/npgd0103.html.

[58]DuPont Personal Protection, http://www2.dupont.com/Personal_Protection/en_US/, 2006.