HUMAN MARS MISSION: TRANSPORTATION ASSESSMENT

Larry Kos*
NASA / George C. Marshall Space Flight Center
Preliminary Design Office / Mission Analysis Team
Mail Code PD31
Huntsville, AL 35812

ABSTRACT

The first Human Mars Mission is being projected to take place during the 2011 and 2013 / 2014 Mars opportunities. Two cargo flights will leave for Mars during the first opportunity, one to Mars orbit and the second to the surface, to prepare the descent transportation, landing site reconnaissance, and ascent vehicle and propellant for the crew arriving during the following opportunity. Each trans-Mars injection (TMI) stack will consist of a payload portion (currently coming in at 60 to 84 metric tonnes) and a TMI propulsion stage (currently coming in at 68-156 mt loaded with propellant) for performing the departure ΔVs for the appropriate Mars trajectories. Three different options are being considered for the system(s) that will comprise the TMI stage for each stack and will perform ΔVs that range from 848 meters/second (m/sec) to 4232 m/sec as required by the trajectory (with gravity losses and various performance margins included). Liquid oxygen (LOx) / liquid hydrogen (LH_2) and LOx / liquid methane (LCH_4) are the propellant options for other transportation elements utilized for descent, ascent, and trans-Earth injection (TEI) in these Mars architectures. This paper will discuss the current applications of the necessary transportation stages to a human Mars mission and project the implications these various options have on an exploration visit to Mars.

NOMENCLATURE

A/B	Aerobrake
AR&C	Automatic rendezvous and capture
C_3	Spacecraft energy relative to Earth
c. g.	Center of gravity
CFM	Cryogenic fluid management
DRM	Design reference mission
DRP	Design reference point
ECRV	Earth crew return vehicle
HEO	High Earth orbit
HMM	Human Mars mission
I_{sp}	Specific Impulse
ISPP	In-situ propellant production
ISRU	In-situ resource utilization
kWe	Kilowatts electric
LCH_4	Liquid methane
L/D	Lift to drag ratio
LEO	Low Earth orbit
LH_2	Liquid hydrogen
LMO	Low Mars orbit
LOx	Liquid oxygen
MLI	Multi-layer insulation
MLV	Magnum launch vehicle
Mpa	Megapascals
MR	Mixture ratio
NSP	Nuclear surface power
NTP	Nuclear thermal propulsion
O:F	Oxidizer to fuel ratio
SEP	Solar electric propulsion
Sol	Martian day, approximately 24.6 hours
TMI	Trans-Mars Injection
TEI	Trans-Earth Injection
V_e	planetary entry speed at 125 km altitude
ΔV	Delta velocity

INTRODUCTION

The Design Reference Mission (DRM) is the term used to describe the current Mars baseline architecture by the Exploration Transportation Office at MSFC, the

* Aerospace Engineer, Senior Member AIAA

Exploration Office at JSC, and the exploration community at large to compare and evaluate approaches to mission, system, and transportation concepts that could be used for human missions to Mars. It does not represent a final or recommended approach to human Mars missions. The DRM captures what is considered to be the best approach to human Mars missions, based on the current understanding of technologies and cost, and is continually in a process of improvement and refinement. An example of this, is the comparison of the description of the current mission proposal to that of previous major studies.[1, 2] Note that this updated "semi-direct" DRM has been built upon earlier versions of this architecture.[3, 4]

The Design Reference Point is the term used to describe various exploration destinations. Currently there are three DRPs. The first one is Mars, the second is the Moon, and the third DRP targets nearer asteroids of possible interest. Quite often the terms DRM and DRP are used interchangeably, since the current efforts have been characterizing only the Mars DRP with the current set of objectives, groundrules and assumptions. The eventual goal is to enable all three DRPs with the highest commonality possible in the fleet of exploration transportation vehicles.

The 2011 opportunity is envisioned to be the earliest a low(er) cost exploration mission could be attempted for the cargo flights, with the corresponding earliest crewed flight to follow in early 2014. If projected milestones for technology developments, lessons learned from the robotic program, hardware developments and demonstrations can occur as scheduled[5], then human exploration of Mars will occur early in the second decade of the next millennium. This schedule assumes typical funding levels of an agency-wide undertaking with both parallel development paths when possible, and serial development paths when necessary.

TRANSPORTATION OBJECTIVES AND ASSUMPTIONS

The objectives of the transportation system, as follows, are to:

- Safely deliver the HMM crew to the Mars surface and back to Earth.
- Reliably deliver all cargo to Mars orbit and to Mars surface.
- Maximize risk mitigation with architecture and hardware designs.
- Minimize development and recurring costs associated with transportation.
- Minimize the mass and power required of the transportation elements to the extent reasonable.
- Limit the time that the crew is continuously exposed to the interplanetary space environment.
- Provide an extendible type of transportation for growth capability to enable later missions.
- Envelope all opportunities of the 15-year cycle with single TMI and TEI stage designs.
- Maximize commonality of systems and subsystems of all hardware element designs.

The assumptions pertaining to the transportation systems are the following:

- Solar electric propulsion specific impulse (I_{sp}): 2100 seconds (sec)
- LH_2 fuel chemical propulsion I_{sp}: 466 sec, MR: 6, Thrust: 24,750 pounds force (lb_f)
- LCH_4 fuel chemical propulsion I_{sp}: 377 sec, MR: 3.5, Thrust: 22,000 lb_f
- Nuclear thermal propulsion I_{sp}: 940-955 sec (-3% for cool-down losses), Thrust: 15,000 lb_f
- Low Earth orbit (LEO): approximately 400 kilometers (km) circular altitude
- High Earth orbit (HEO): 800 km perigee altitude by 70,761 km apogee altitude
- Mars parking orbit: 250 km periapse altitude by 33,793 km apoapse alt., 40° maximum inclination
- Aerobrake scaling equation[6]: $M_{A/B} = \sqrt{(M_{PYLD})}*(a + b*V_e) + M_{STR}$; (where $M_{A/B}$, M_{PYLD}, M_{STR}: mt, V_e: km/sec, coefficients: a = -0.55, b = 0.19)
- Cryogenic fluid management boil-off rates: 0.0 - 1.9% of the initial propellant load per month

MISSION OVERVIEW

The mission design for the crew utilizes relatively fast transits (170 to 200 days) to and from Mars with long surface stays (18 to 22 months; 560 days nominal). Cargo missions are sent to Mars on near optimum low energy, long-transit-time trajectories. Figure 1 illustrates a top level mission sequence for the first opportunity. All cargo is delivered to low Earth orbit (LEO) on an 80-84 metric tonne (mt) class Magnum-type launch vehicle (MLV). The DRM requires one automatic rendezvous and capture (AR&C)

operation in LEO for each Mars cargo element launched on the MLV prior to the injection of each outbound trans-Mars Injection (TMI) cargo stack to Mars. The piloted flight in the high thrust DRM requires a second rendezvous in LEO for crew boarding, while the low thrust DRM requires a rendezvous in high Earth orbit (HEO) prior to injection of each outbound piloted stack to Mars. Two cargo TMI stacks and one piloted TMI stack are the minimum required for each human mission. Each mission, though, has an additional two backup cargo payloads which are launched during the same opportunity as the crew and provide full cargo vehicle-level redundancy, while at the same time functioning as the primary hardware for the subsequent piloted mission.

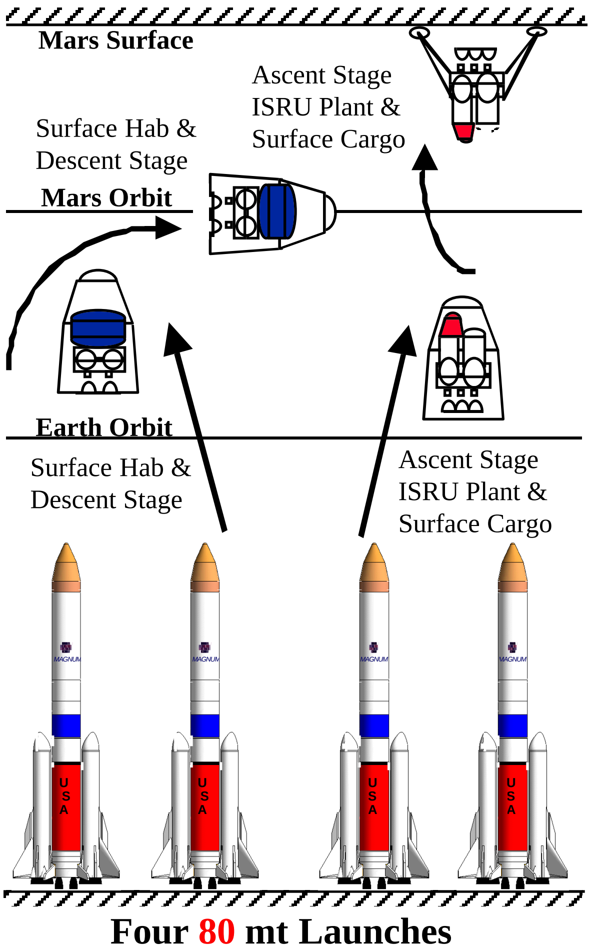

Figure 1 - *2011 Design Reference Mission Illustration*

The first cargo launch of the first opportunity in 2011 includes a fully fueled piloted descent vehicle and surface habitat, is delivered to Mars orbit, and aerocaptures into the 1 Sol orbit (250 km x 33,793 km) having the 24.6 hour period. The crew will rendezvous with this vehicle in Mars orbit, and use it to descend to the surface to within tens of meters of the cargo stack that landed previously during the 2011 opportunity.

The second cargo delivery in 2011, which lands on the surface immediately after its arrival at Mars, includes, but is not limited to the following main elements. The primary element is the ascent vehicle, which includes the empty ascent stage's tanks that will be filled on the Martian surface during the time prior to the crew's departure in the next opportunity. Also included is the in-situ resource utilization (ISRU) plant with a nuclear surface power (NSP) unit to provide its power for the in-situ propellant production (ISPP) process, and the seed liquid hydrogen to be used as a reactant to produce the ascent propellant. This plant provides other resource utilization for crew consumables of oxygen and water as well. After this stack lands, the NSP unit will be autonomously deployed one kilometer or more from the lander, and the ISPP facility will begin producing the liquid oxygen (LOx) and liquid methane (LCH_4). This fuel and oxidizer will be required to launch the crew back into the 1 Sol Mars orbit to the trans-Earth injection (TEI) stage for their return trip to Earth. Propellant production on Mars' surface will be completed and verified prior to the scheduled departure (TMI) of the first crew from Earth.

In the second opportunity, during January of 2014, the crew of six departs for Mars in a transit habitat with the return (TEI) stage and backup capsule also attached. The outbound trip time is 178 days, with a limitation on the entry speed of 7.4 km/sec at Mars arrival. After aerocapturing into the highly elliptical 1 Sol Mars orbit, the crew will dock with and board the descent stage / surface habitat stack, descend through the atmosphere using aeroentry maneuvers and parachutes, and land on the surface with the throttlable, liquid methane chemical propulsion system. The rendezvous with the previously delivered cargo elements already on the surface occurs by means of a controlled precision landing using the high lift/drag (L/D) ratio of the aerobrake, the cross-range capability during the powered descent portion, and a tracking beacon for the trajectory guidance. Also sent during the second opportunity is the cargo for the second piloted flight, which also serves as the backup hardware for the first crew. The mission architecture illustration for the second opportunity is shown in figure 2.

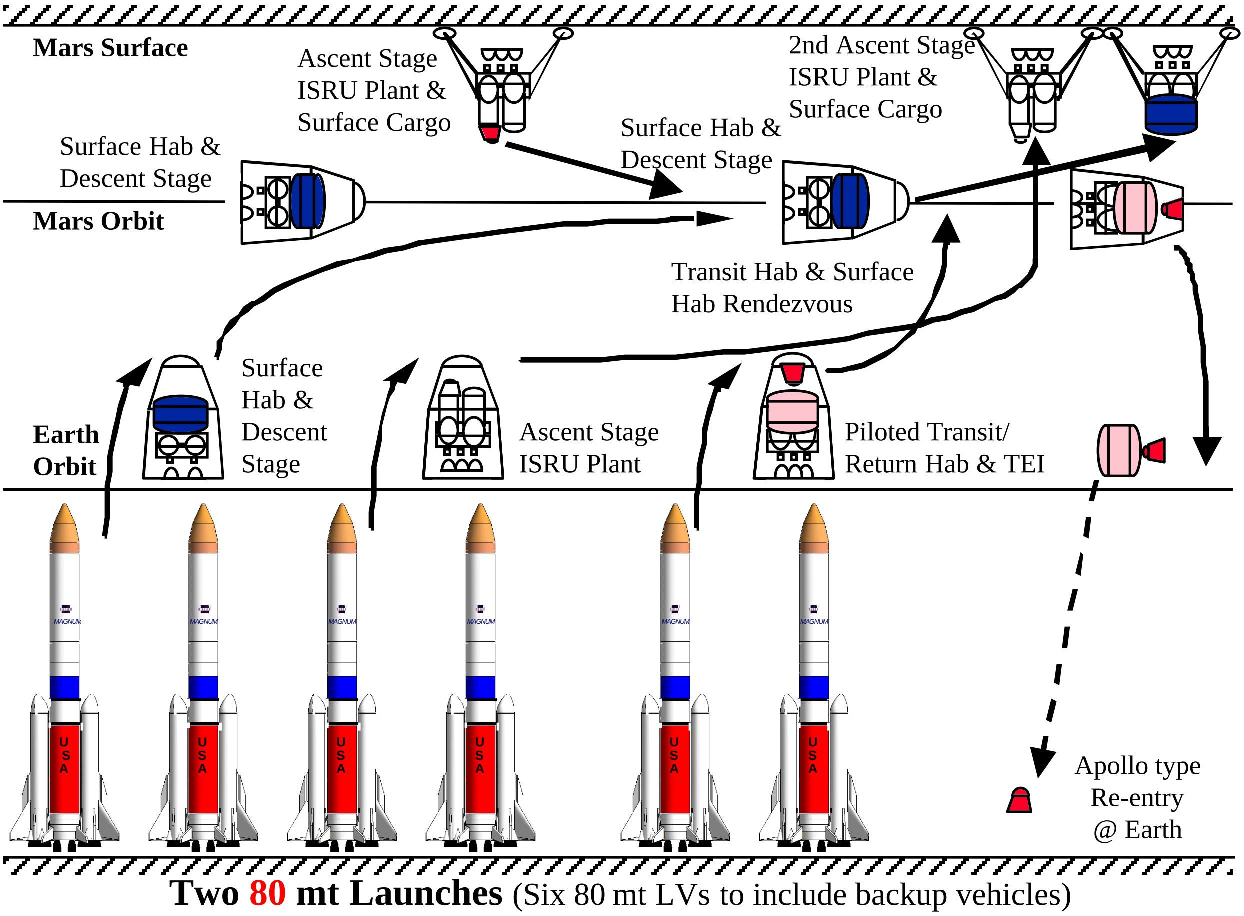

Figure 2 – *2013/14 Design Reference Mission Illustration*

The return leg for the first piloted visit begins with the TEI maneuver during January of 2016. The crew ascends from Mars in the Earth crew return vehicle (ECRV), which also is referred to as a capsule, and does an AR&C-type rendezvous with the TEI stage. The return trip time is 156 days with a maximum entry speed of 13 km/sec at Earth return. The atmospheric entry is done with the ECRV, and is the only piece of transportation hardware returned to Earth. Both portions of the piloted trajectory and the stay time on the surface are shown in figure 3.

More details of both the TMI propulsion systems and the Mars vicinity transportation can be found in very recently published papers [7, 8]. Two recent Earth/ Mars trajectory studies have also contributed significantly to the characterization and selection of outbound trajectories aerocapturing through Mars atmosphere into Mars orbit and return trajectories with aeroentry into Earth's atmosphere. Their influence on mission design and vehicle sizing has been such that trip duration has been traded against aerobrake mass and propellant mass to reduce architecture sizing while still accommodating sufficiently short piloted trip times [9, 10].

All transportation was sized for the current, October '98, version of the DRM cargo [11, 12, 13]. The cargo elements are the flight-one surface habitat and additional surface cargo at 31.5 mt, the flight-two surface cargo including the primary ECRV at 35-40 mt, and the piloted flight with the transit habitat, on-orbit cargo, and the abort ECRV at 37.5 mt. This ECRV is a second return capsule for the case when the crew cannot land. The definition that has been received on the cargo has improved the transportation sizing and packaging assessment significantly during the last few design iterations of the DRM development. This can be seen in figure 4, which shows the scaled drawings of the three TMI stacks for one of the high thrust architectures of this DRP. The following section will discuss each transportation element in more detail.

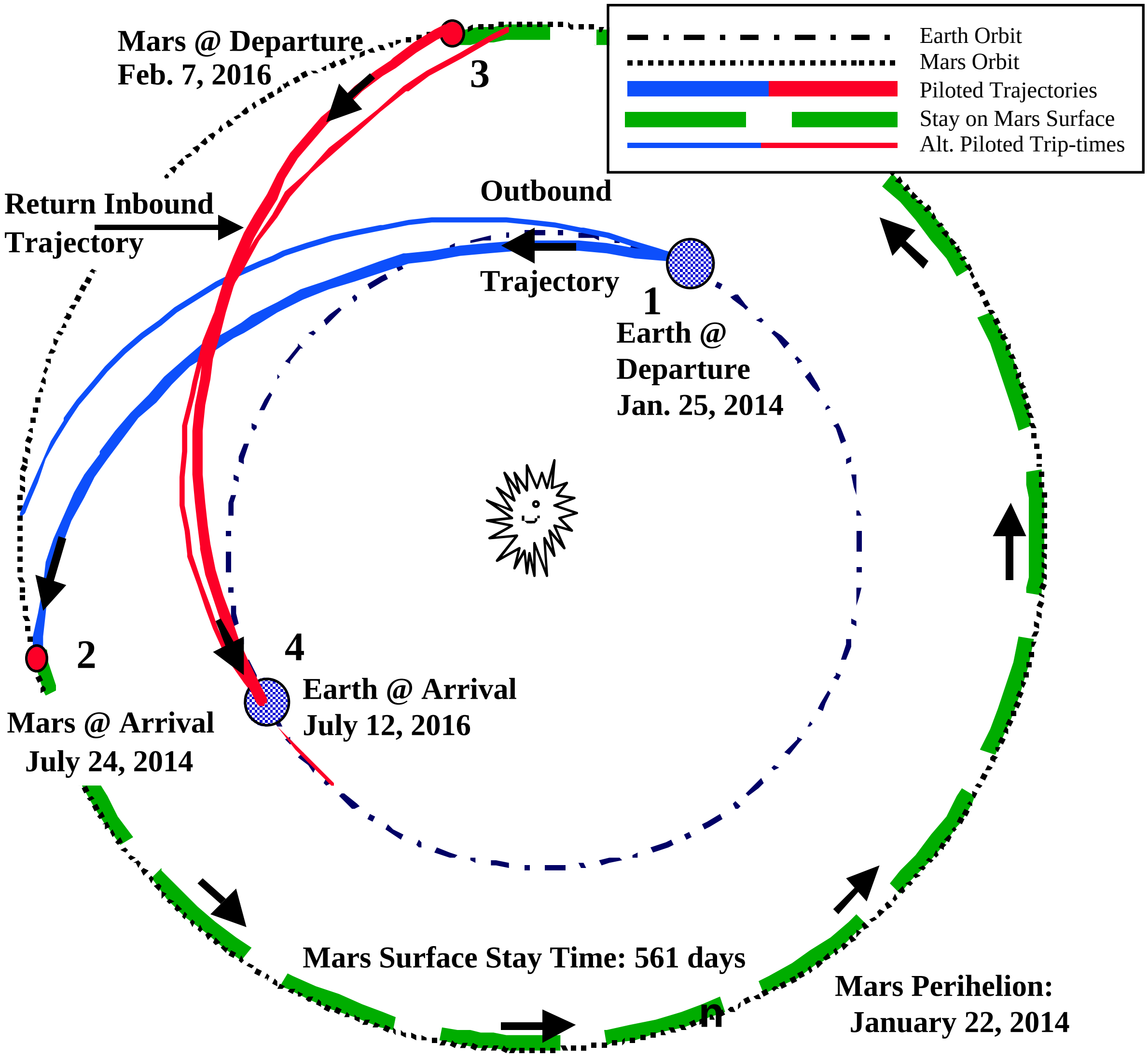

Figure 3 – *HMM 2014/2016 Piloted Trajectories*

TRANSPORTATION ELEMENT DESCRIPTIONS

The following sections describe in more detail each of the major transportation elements to the Design Reference Mission. Note that all stages are still in their conception, evolution and sizing phases. The three options for TMI are the main discriminating feature between the various architectures. Other differences include TEI propulsion type and propellant, NSP design, and descent and ascent propellant loadings.

Trans-Mars Injection (TMI) Stages

Solar Electric to HEO w/ Chemical TMI Assist

The SEP HEO stage delivers the Mars payload to the 800 x 70,761 km highly elliptical Earth orbit. The high thrust LOx/liquid hydrogen (LH_2) chemical TMI stage performs the last portion of the TMI ΔV to boost the stack to sufficient energy (C_3) to achieve the same interplanetary trajectory as the high thrust TMI options. The chemical stage provides the last 848 m/sec to 1,398 m/sec burn to reach the same C_3. The chemical stage is baselined to use RL10B–2 type engines at a thrust level

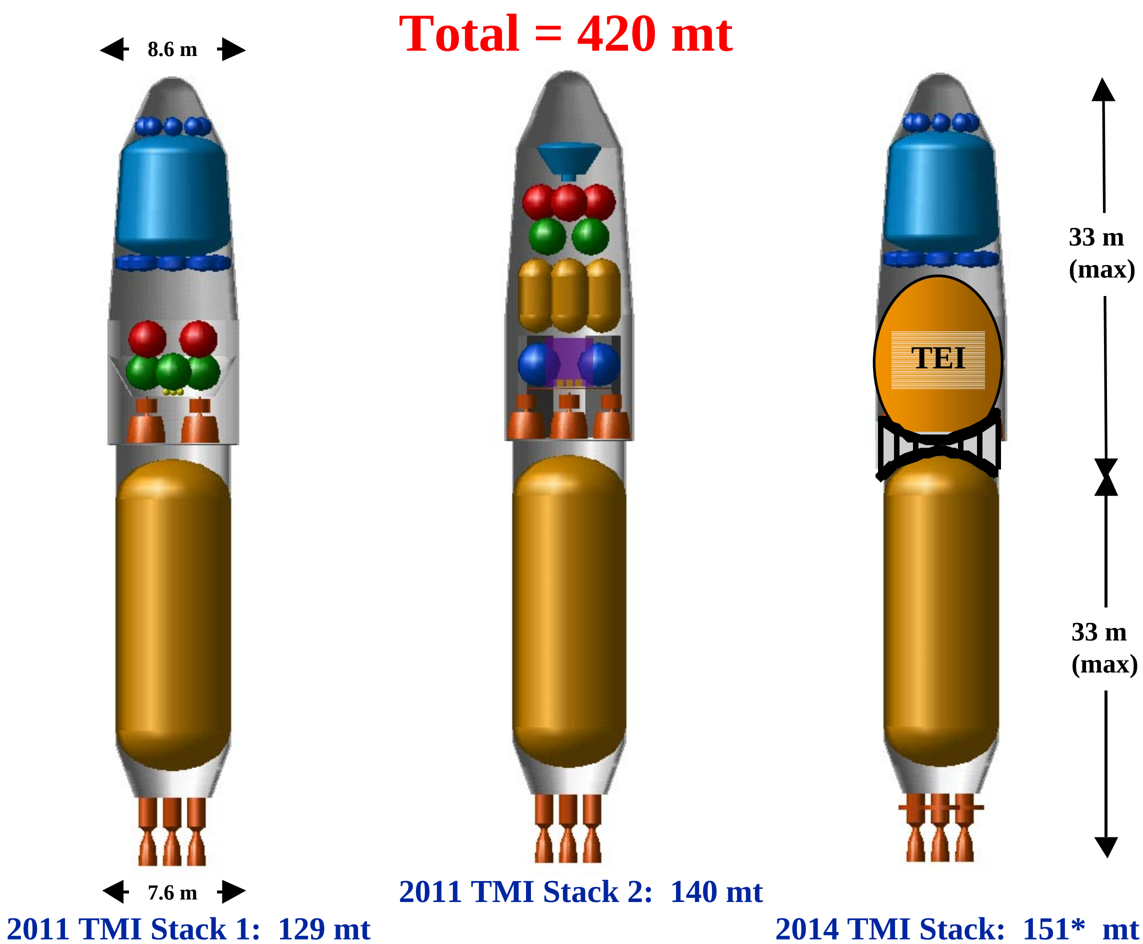

Figure 4 – *Transportation Element Illustration*

of 24,750 lb_f, with an I_{sp} of 466 sec at an oxidizer to fuel (O:F) mixture ratio (MR) of 6:1, and a nozzle area ratio of 285. An illustration of this engine can be seen in figure 5. The boil-off rates for all the chemical stages used beyond LEO is less than 0.1% of the initial propellant loading per month (%/mo). The same engines are also used in the high thrust chemical TMI option in the following section.

The SEP stage description will be published later in more description, however, an initial sizing has just recently been made available. The spiraling stage will have a peak specific impulse of 2100 sec, 800 kWe power to provide the thrust, and uses Xenon as the propellant for the electric thrusters. The stage delivers both the cargo in 2011 to HEO using most of the 60 mt of propellant, returns to LEO with that remaining, and then once resupplied with another 47 mt of propellant, spirals the crew out to the HEO for their TMI departure. The dry mass of the SEP stage is estimated to be approximately 16 mt. A preliminary artist's conception of this stage is shown in figure 6. This is not shown to scale, as the solar arrays would extend just as far in the horizontal direction as they do in the vertical direction as depicted. The shape of the array is also still currently being designed, so the purpose this drawing is just meant to show both the low thrust and high thrust portions together on one TMI stack.

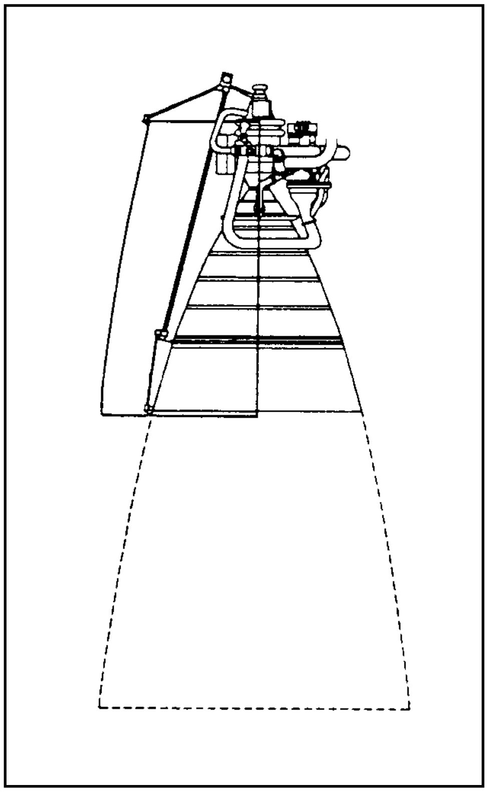

Figure 5 – *HMM LOx/LH$_2$ Chemical Engine*

Chemical Propulsion TMI

As in all following high thrust TMI options, each TMI burn is done during two perigee passes. In the case of the chemical TMI, the total TMI is done with a two stage TMI element. The first stage pushes the entire stack through a burn duration that is approximately half of the total TMI burn time. This stage drops off, i.e. it is "staged", in the interim departure orbit. This orbit typically has an apogee on the order of thousands of kilometers. The period of this intermediate coast orbit is three to six hours. The resulting ΔV is less than 38-42% of the total, because the stack mass is significantly less for the second stage of the TMI element. The second TMI element also does nearly half of the total TMI burn time and sends the payload out onto the proper interplanetary trajectory.

The first-burn chemical TMI stages use four RL10B–2 type LOx/LH$_2$ engines, giving a total thrust level of 99,000 lb$_f$ initially for each TMI stack. The second-burn chemical TMI stages use three RL10B–2 type LOx/LH2 engines, giving a total thrust level of 74,250 lb$_f$ for each TMI sub-stack. These chemical stages are also baselined to use RL10B–2 type engines and have the same thrust level, I$_{sp}$, and nozzle area ratio mentioned above.

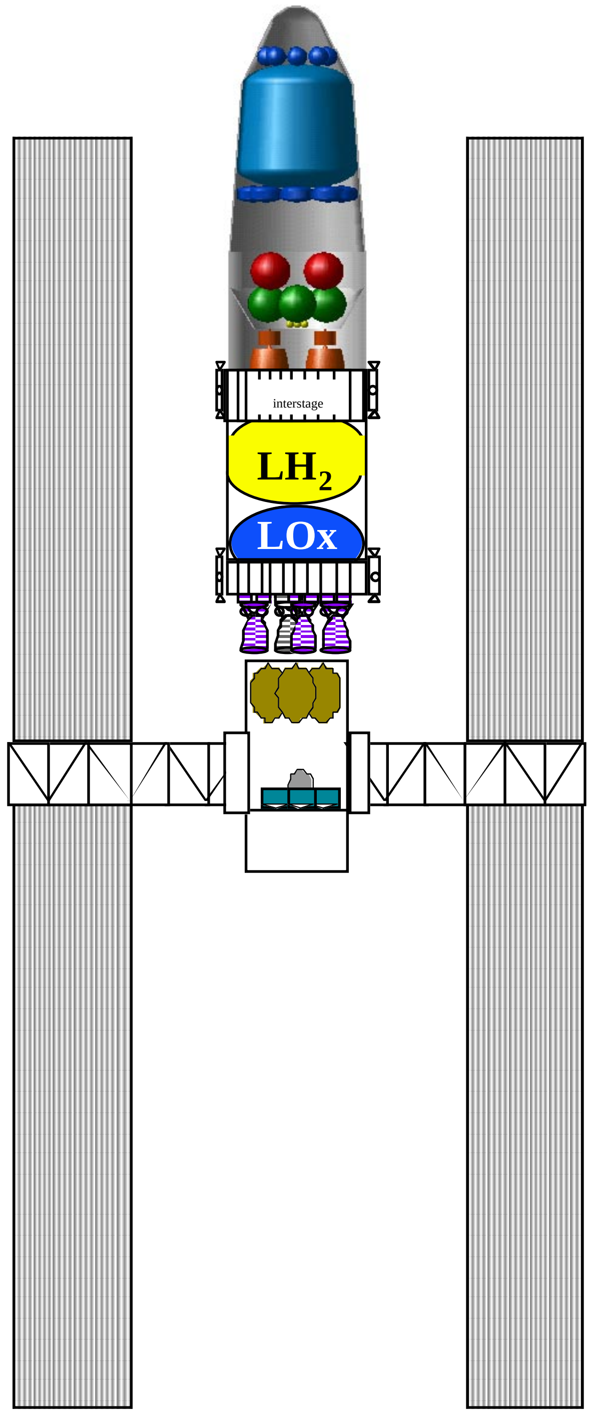

Figure 6 – *SEP TMI Stack Illustration*

To reduce the TMI propellant loading, it is necessary to reduce the gravity losses and the propellant boil-off. The TMI maneuver is done over two separate perigee passes to minimize gravity losses and can typically benefit the TMI delta velocity (ΔV) budget by 100 m/sec for the cases of the chemical TMI architecture. Second, the boil-off rates for all the chemical TMI stage propellant tanks are approximately 0.3% of the initial propellant loading per month (%/mo.) when using passive cryogenic fluid management (CFM). This drives these elements of the TMI stacks to be the last pieces launched into LEO to minimize the time spent there. By using passive CFM, the stack does not have to carry the mass of a larger active CFM subsystem through the large TMI ΔV.

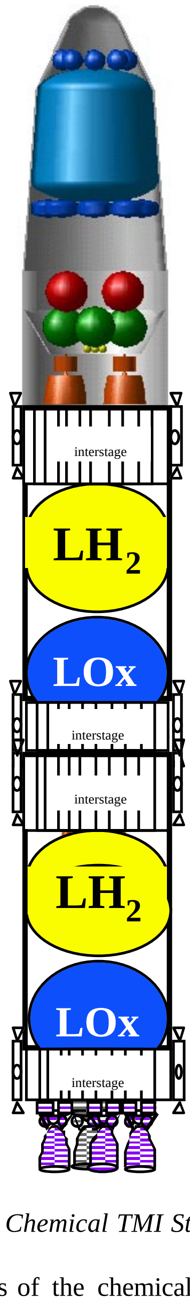

Figure 7 – *Chemical TMI Stack Concept*

The two stages of the chemical TMI element can either be designed to be identical, with the same propellant load in each as shown in figure 7, as is currently being done, or they can be tailored to maximize the utilization of the launch vehicle. Note that in this case the second stage still has one less engine. Both of these design options have advantages and disadvantages. The development cost for two differently sized stages will be greater than the cost for a single design with a single qualification process and used for both burns. Unique designs for each burn can be done to fully utilize the delivery capability on the MLV with the design for the first TMI stage, therefore minimizing the second TMI stage. This minimization allows for case where a portion of the cargo can be integrated onto the launch vehicle. This will be more costly, but can possibly lower the number of Magnum launches required.

Nuclear Thermal Propulsion TMI

The NTP TMI stage uses three 15,000 lb_f thrust nuclear engines, giving a total thrust level of 45,000 lb_f thrust for each TMI stack. The thrust-to-weight of the engine itself is approximately 3.1 and the propellant used will be LH_2. To minimize the corresponding LH_2 TMI propellant load two things are done. First, to reduce the gravity losses, the TMI maneuver is again done over two separate perigee passes. This benefits the TMI ΔV budget by typically 150 m/sec or more. Second, the boil-off rates for all the nuclear TMI stage hydrogen tanks are approximately 1.9%/mo, driving these elements of the TMI stacks to be the last pieces launched into low Earth orbit (LEO) to minimize the time spent in LEO. The longest nominal duration that a TMI stage will be in LEO in this architecture is approximately 37 days, requiring a sufficient amount of the passive thermal protection system to minimize the LH_2 boil-off. Active CFM thermal control methods were not selected for the TMI LH_2 tank, since the estimates for the mass of the CFM subsystem are similar to that of the estimated boil-off mass. This again allows the mass for CFM to not have to be carried through the large TMI ΔV. Note that the MLV shrouds are required to launch the TMI stages to protect the sensitive insulation required for a large LH_2 tank. The diameter of these stages is reduced to 7.6 m to be accommodated by the inner diameter of the shroud support rings. In one option, the NTP TMI is baselined to be dropped immediately after TMI, while the bimodal option would carry it longer so that it could be used in other mission phases as another resource.

A bimodal concept for NTP stage would provide power to the TMI stack for an extended time, and therefore the TMI stage would be carried nearly all the way to Mars, with the cargo TMI stages being disposed of shortly before the Mars orbit capture maneuver. Although this has some design impact to the rest of the TMI stack, it will not be addressed at this time. See reference seven (Borowski, et. al., 1998) for specific design and data on this TMI stage concept.

Aerobrakes

The aerobrake (A/B) elements of the transportation are very crucial items to the DRM architecture. First, the aerobrakes are multi-use, and second they enable part of the precision landing requirements. The Earth-to-orbit (ETO) launches with cargo utilize the A/B as the launch vehicle shroud. The triconic shape of the desired mid-to-high L/D, typically on the order of 0.5 – 0.9, aerobrake is very similar to that required for the MLV ascent shroud. The aerobrakes are heavier than the MLV shroud, but they still have to be packaged on and lifted to orbit by the MLV in some way. The aerobrakes perform the Mars orbit capture maneuver as their second use, with the current baselined limitation for the peak capture velocity of 7.4 km/sec. In the case of the surface payload and the piloted mission, a subsequent aeroentry is done with the same shell, this becoming the third use of the A/B. The sizing methods used for these aerobrakes indicate that the cargo aerobrakes will have a mass fraction of approximately 12% as a result of the A/B scaling equation, while a piloted two-aeropass A/B will have at least a mass fraction of 16%. In most cases with Mars payloads and in nearly all cases when the payload's destination is the surface, use of the A/B is more mass efficient than doing the Mars orbit capture with a propulsive stage. The aerobrakes for this architecture have masses between 9.5 mt and 13.8 mt.

A schematic of the aerobrake is shown in figure 8. This figure shows the maximum length A/B, but in practice the cylindrical section will always be shortened to the appropriate length to accommodate packaging the payload. This customizing will help minimize the mass of the A/B. All other A/B dimensions are fixed due to the diameter being fixed to match up with the MLV core section outside diameter of 8.6 m and the structure interface diameter of 8.38 m. As shown in the aerobrake scaling equation, $M_{A/B} = \sqrt{(M_{PYLD})}*(a + b*V_e) + M_{STR}$, the structural mass is a term added in directly to scaling equation rather than scaling with Mars entry speed parameter. This is due to the structural g-load environment at liftoff, aerocapture, and aeroentry. This structural mass has been scaled as a function of the A/B length with a near linear curve fit.

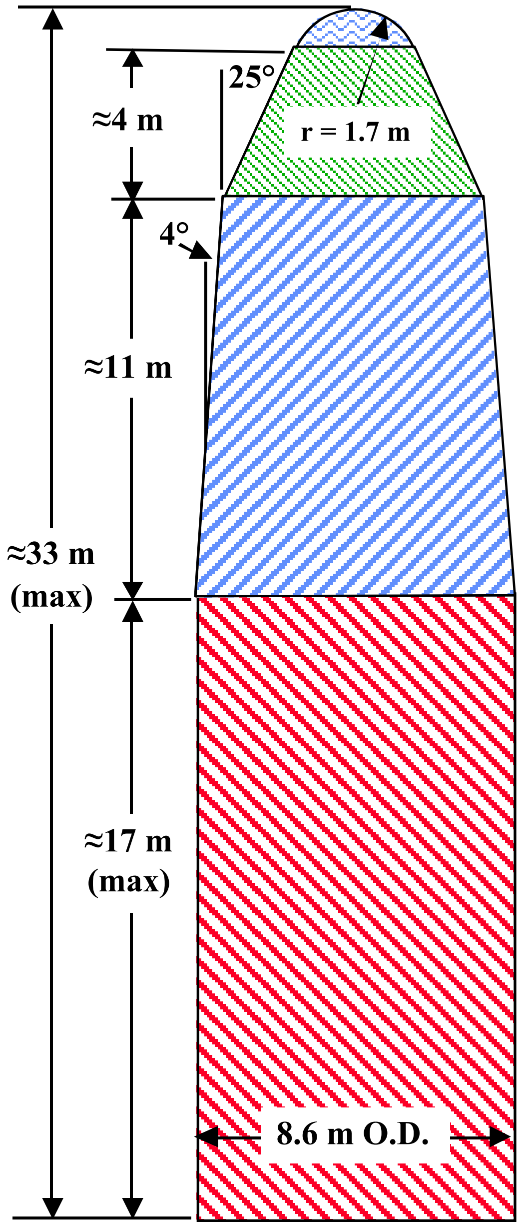

Figure 8 – *Aerobrake Configuration*

Descent-Only Stage

The descent-only stage is the stage used for the piloted landing on the surface of Mars. The stage includes six 22,000 lb_f LOx/LCH_4 engines, and has a landing ΔV budget of 632 m/sec. These engines have an I_{sp} of 377 sec at an O:F MR of 3.5:1, and a nozzle area ratio of up to 400. The engine chamber pressure is approximately 600 psi (4.1 MPa). This stage requires only four of the commonly sized tanks to accommodate the piloted descent propellant loading. The DRM version 2.0 mission used a common descent stage for both cargo and crew with a different ascent stage, for a total of two different vehicles needing to be developed and designed for this mission. The current mission design still requires only two different vehicles to be developed, although the two landers are now subtly different. The descent only stage and the surface hab stack is the first element launched to LEO for the piloted mission of the 2014 opportunity.

A major change from the DRM version 3.0 that affects the stack packaging is that the touchdowns are being done with the stacks landing horizontally. This allows for much easier deployment of the NSP and the communications system, and allows the height of vehicles above the ground to be minimized. However, engine and tank arrangements are now a critical design concern, especially in the case of the descent / ascent stage. Also of concern will be dropping portions of the aerobrake, which was previously retained as integral unit. Center of gravity (c. g.) location is still as critical as it was before with the vertical landing concept.

Descent/Ascent Stage

The descent/ascent stage is a combined version of the previously separate descent and ascent stages. The driver for this design was the minimization of the length of the payload for operations in ground facilities and for operations on Mars' surface, and minimizing the mass associated with the transportation (for example engines, tanks, etc.…). This descent/ascent stage and the surface cargo is the second element launched into LEO for the 2011 opportunity. All cargo elements are sized at 8.6 m outside diameter to match up with the corresponding dimension of the MLV. This stage also uses six of the 22,000 lb_f LOx/LCH_4 engines.

The stage is structured such that the NSP unit, approximately 1.5 m tall and 3 m in diameter, can be deployed off one end that is already relatively close to the surface. Later, after the crew reaches the surface, the portion of the stack not required during or after ascent (i.e. the four liquid hydrogen seed tanks), can be detached to allow the ascent event to be performed without interference. The landing ΔV budget for the stage is also 632 m/sec, with approximately 30% of the total propellant capacity of the tanks being used for descent. However, the ascent maneuver requires all eight tanks to be fully loaded at the time of crew departure from the surface Mars, and of course, is the driver for the tank sizing. Of concern, is the location of the stack's c. g., especially during the aerocapture, the aeroentry, landing, and the ascent events of the mission. The engine arrangement at landing with six engines must also provide the proper c. g. location for the ascent stack when two of the six engines are used (or reused, as the case may be) for the ascent maneuver. The ascent maneuver is the largest single ΔV performed during the entire mission scenario and is budgeted at 5625 m/sec.

Trans-Earth Injection (TEI) Stage

The trans-Earth injection (TEI) stage is the stage used for the crew's return to Earth. The stage is responsible for aligning its orbit plane about Mars above the crew's landing site prior to the ascent from the surface. A plane change maneuver has been included in the stage propellant load sizing and design, and will be controlled externally from the TEI stage. This will minimize the amount of propellant necessary to be made on the surface (with the ISPP processes) and help minimize performance requirements for translational ΔV on both the ascent stage and the ECRV. The stage also includes two 24,750 lb_f LOx/LH_2 engines and a ΔV budget of 2152 m/sec. The propellant capacity for the stage is set to be 27 mt and the dry mass is approximately 6.4 mt.

An example of an architecture mass manifest is shown in figure 9. This table summarizes an architecture at its system (highest) level. There are versions of this table that show detail down to the subsystem level and component level as well for some of the studied architectures that have reached these lower levels of development and design. The example shown is the summary for the SEP architecture.

An alternative to the LOx/LH_2 TEI stage above is a TEI stage that would includes two LOx/LCH_4 22,000 lb_f engines and a similar ΔV budget. The propellant capacity for the stage will be 8 to 12 mt heavier, depending on whether the CFM option selected is active control or passive control. This version of the TEI stage would have dry mass of up to 7 mt.

	Description	2011 Cargo Flight 1	2011 Cargo Flight 2	2014 Piloted Flight (5)	**Totals**
TEI Stage	Transit Habitat (mt)			28.8	
LOx / LH_2	Burnout Mass (mt)			6.3	
I_{sp} = 466 sec.	Propellant Mass w/ 500 m/s (mt)			27.0	**33**
V_∞^2 = 14.8	Propellant Mass Fraction			0.81	
Ascent Stage	Ascent/Earth Return Capsule (mt)		5.1	5.1	
LOx / CH_4	Burnout Mass (mt)		4.8		
I_{sp} = 377 sec.	Propellant Mass (mt, ΔV_{asc} = 5625 $^m/_s$)		*39.5		
	Return Science Payload Mass (mt)		**0.1		
Descent Stage	Crew (Piloted Mission Only, mt)			0.6	
LOx / CH_4	Surface Habitat Module Mass (mt)	23.7			
I_{sp} = 377 sec.	Surface/On-orbit Payload Mass (mt)	7.8	20.5	3.6	**32**
Nuclear Surf. Pwr.	LH_2 for Water Cache & Asc Prop (mt)		4.1		
	Aerobrake Mass ($\sqrt{M_p}$*(a+b*V_e)+M_s, mt)	9.7	9.8	13.2	**33**
	Descent System (Dry+chutes) (mt)	5.3	5.3		**11**
	Landing Legs (mt)	2.0	2.0		**4**
	Propellant Mass (mt, ΔV_{desc} = 632 $^m/_s$)	11.5	9.9		**21**
Cargo TMIs (Chem)	Burnout Mass (mt)	4.3	4.3		**9**
LOx / LH_2	LOx / LH_2 Propellant Mass (mt)	19.1	18.8		**19**
I_{sp} = 466 sec.	Propellant Mass Fraction	0.82	0.81		
	Payload Mass Subtotal (mt)	**83.5**	**84.7**	**84.7**	**253**
TMI (SEP & Chem)	Burnout Mass (mt)			6.0	**6**
Xe / LOx / LH_2	LOx / LH_2 Propellant Mass (mt)			32.7	**33**
I_{sp} = 2100, 466 sec	SEP Dry & Xenon Prop Mass (mt)	60.0	16.0	47.1	
C_3 = 13.8, 18.4	Propellant Mass Fraction	0.79	0.75	0.85	
	Total Stage & Resupply (mt)	**60.0**	**16.0**	**85.8**	**162**
	TOTAL IMLEO (mt)	**143.5**	**100.7**	**170.5**	**415**
Number of **80** mt LV Flights (75%–100% packing efficiency)		**2**	**1**	**2**	**5**

* Produced at Mars using ISRU, w/o boil-off; ** Gathered samples & rocks; Crew taxi stage: 3.2 mt dry + 12.7 mt prop = 15.8 mt stage.

Figure 9 – *HMM SEP Architecture Mass Manifest*

A second alternate version of the TEI stage, uses up to two of the same 15,000 lb_f NTP engines as the NTP TMI stage. The required TEI propellant drops to 15.5 mt from 27 mt, but the dry mass increases from 25.8 mt (in the case of the NTP engines doing just TMI) to 27.2 mt for the case when both TMI and TEI are performed with the same engines. Part of this increase is the mass for the radiation shielding necessary for the crew during the TEI burn. There is also a decrease of 8 mt in the TMI LH_2 propellant load due to the smaller mass required for TEI. The combined mass of the TMI and TEI dry stages and propellant is 24 mt less in the case of NTP for both maneuvers. This shows both the advantage of retaining and reusing the engines as well as the much higher performance (I_{sp}) of the nuclear propulsion technology.

RESULTS

A summary of the results of these studies will provide a good foundation to achieve exploration of Mars using six or fewer large Magnum-type launch vehicles. These architectures' total mass is designed to be and is less than 480 mt. This summary has been updated on a yearly basis[11, 12] by the intercenter exploration team, and the next one should be this foundation. A backup or preliminary architectural option with the chemical TMI architecture is also shown, and can be done with eight or nine MLVs with a total mass of less than 640 mt. This backup option is designed to capture the initial three opportunities to initiate interest for human exploration of Mars with minimal "high-tech" hardware developments. This paper is a brief preview of this summary that will likely be referred to as "DRM version 4.0".

CONCLUSION

Exploration of Mars can now be undertaken with more robust architectures and transportation systems equally or more capable and less massive than during the SEI studies. The goal of sending a suite of three exploration flights / missions to Mars at a fraction of that proposed before is nearing completion. Hardware and technologies which have been developed in the last decade and technology developments which are currently in progress, for example zero boil-off demonstrations at Lewis Research Center, have dramatically improved both the performance and the cost projections for what interplanetary exploration requires beyond the industry's current capabilities.

The human exploration of Mars can be done with non-nuclear propulsion, but it is at the expense of either developing large and efficient solar electric propulsion or much larger TMI stages. Both options will be very challenging to implement though.

The chemical TMI architecture requires a more launches per opportunity, having implications for both ground and on-orbit operations. However, the current TMI stage designs lend themselves very well to being put up by an 80 mt MLV. The TMI engines use LOx at the 6:1 O:F MR, so the packaging efficiency of the stage is 100%. The TEI stage in this option would also be LOx/LH_2 because the same engines used for TMI could be applied to this stage design as well, thus minimizing transportation stage design and development costs. Of concern will be the long-term active CFM necessary for the LH_2 on TEI stage that loiters in the high Mars orbit for the 18 months the crew is on Mars surface. The actual capability to prevent LH_2 boil-off is not the problem, but making the system perform without failure for a very extend time is the area of concern.

The SEP architecture requires long spiral times in Earth orbit and the associated operations for this portion of the mission can be significant. Other concerns that are being addressed are the launch vehicle packaging and on-orbit deployment of a large photo-voltaic array system. Some design work has been done in this area, but the design has not been completed for the current payload manifest. Also required will be rendezvous in a high Earth orbit that is also very elliptical. This is quite different from rendezvous in a low circular orbit. This architecture does look promising by the fewer number of launches that is required to put up the payloads in low Earth orbit. The feature of reusing the SEP stage for the crew payload, after it was used to spiral out the cargo, shows areas where minimizing the number of flight units also helps in reducing recurring costs for this architecture. This architecture has been modeled with both a LOx/LH_2 TEI stage and a LOx/LCH_4 TEI stage since the efficiency of the SEP stage allows for a heavier TEI stage. This trade was concerned with the more difficult problem of LH_2 active CFM, as opposed to the much easier LCH_4 active CFM. This problem comes down to one-stage cryo-coolers versus two-stage cryo-coolers.

The nuclear architecture discussed here is the "comparison case" for the non-nuclear architectures. It is somewhat easier to enable a mission with this technology, however, development costs and public acceptance are of concern in this architectural option.

ACKNOWLEDGMENTS

The author wishes to express his appreciation for the work of the members of the Human Mars Mission Study Team at the Marshall Space Flight Center over the entire 1998 fiscal year. The team produced the dramatic changes in the architectures during a period with highly fluctuating designs to update the assessment of the HMM transportation.

Further, the author would like to acknowledge that this concept also could not continue to be improved without the cooperation and teamwork provided by the intercenter exploration team members at the Johnson Space Center, Ames Research Center, and the Lewis Research Center also currently working on this mission concept.

REFERENCES

[1] Cohen, A., et al., "*Report of the 90-Day Study on Human Exploration of the Moon and Mars*," National Aeronautics and Space Administration Report, U.S. Government Printing Office (GPO), Washington, DC, November 1989.

[2] Stafford, T. P., et al., "*America at the Threshold: Report of the Synthesis Group on America's Space Exploration Initiative*," National Aeronautics and Space Administration Report, U.S. GPO, Washington, DC, May 1991.

[3] Weaver, D. B., M. B. Duke, B. B. Roberts, "*Mars Exploration Strategies: A Reference Design Mission*," IAF 93-Q.1.383, NASA Johnson Space Center, Houston, TX, 1993.

[4] Weaver, D. B., M. B. Duke, "*Mars Exploration Strategies: A Reference Program and Comparison of Alternative Architectures*," AIAA Space Programs and Technology Conference, AIAA 93-4212, NASA Johnson Space Center, Houston, TX, 1993.

[5] Richards, J. S., Calhoun, T. M. "*HMM Exploration Schedule"*, Internal document, NASA Marshall Space Flight Center, Huntsville, AL, January 1998.

[6] Wercinski, P., "*Estimates of Triconic Aerobrake Mass*", Internal Presentation, NASA Ames Research Center, Moffett Field, CA, January 1998.

[7] Borowski, S. K., Dudzinski, L. A., McGuire, M. L., "*Vehicle and Mission Design Options for the Exploration of Mars/Phobos Using 'Bimodal' NTR and LANTR Propulsion*," 34th AIAA/ASME/SAE/ ASEE Joint Propulsion Conference and Exhibit, AIAA-98-3883, Lewis Research Center, Cleveland, OH, July 1998.

[8] Dexter, C. E., Kos, L. D., "*An Overview of Mars Vicinity Transportation Concepts for a Human Mars Mission*," 10th Annual Propulsion Engineering Research Center (PERC) Symposium on Propulsion, October 1998.

[9] George, L. E., Kos L. D., "*Interplanetary Mission Design Handbook: Earth-to-Mars Mission Opportunities and Mars-to-Earth Return Opportunities, 2009-2024*," NASA/TM-1998-208533, July 1998.

[10] Munk, M. M., Debban, T., Graves, C. A., *'Departure Energies and Entry Speeds for Human Mars Missions'*, Internal JSC document and presentation, NASA Johnson Space Center, Houston, TX, July 1998.

[11] "*Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team*", Hoffman, S. J. and Kaplan, D. I., Eds., NASA Special Publication 6107, July 1997.

[12] "*Reference Mission Version 3.0 Addendum to the Human Exploration of Mars: The Reference Mission of the NASA Mars Exploration Study Team*", Drake, B. G., Ed., Exploration Office Document EX13-98-036, June 1998.

[13] Drake, B. G., Personal communication, NASA Johnson Space Center, Houston, TX, October 1998.