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Comparative Analysis of Current NASA Human Mars Mission Architectures

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The human Mars Mission architectures recently under study are compared. The evolution of mission modes utilizing indigenous resources for propellant production is described. A historical summary of human Mars architecture design is given as a way of introducing some of the challenges inherent to mission planning. A Mars architecture option based on minimum development costs and risks, rather than minimized mass in low Earth orbit, is presented. A multiyear Mars vehicle set, characterized by chemical propulsion transfer stages (with aerocapture for Mars orbit capture) is described. It is maintained that, for the current Mars Design Reference Mission, the development of more costly advanced propulsion technologies (such as nuclear thermal propulsion, nuclear electric propulsion, or solar electric propulsion) can be considered as later, evolutionary steps from this initial, less costly, all-chemical approach. Emphasis is given to the minimization of new technology development programs and the associated impacts on total Mars mission costs. Analysis is based on the availability of an 80-metric-ton-payload-class launch vehicle. An important ground rule is the preclusion of on-orbit assembly, allowing only for the joining of complete and fully fueled transfer stages in orbit.

Introduction

OVER the past three decades, a range of space transportation architectures has been proposed as the means to realize human exploration of Mars. The choice of trajectory type, mission duration, mission frequency, fleet size, payload requirements, abort modes, required precursors, vehicle reuse options, and other concerns together constitute the architecture. The selection of the propulsion system type for the Mars transfer vehicle is one of the key decisions that must be made in such analyses. The propulsion system's performance, reliability, reusability, and cost all enter into that decision. In the Mars mission design studies[1,2] of the 1960s, nuclear thermal propulsion (NTP) was almost uniformly selected as the propulsion system of choice for the Mars transfer vehicle. Typically, the large majority of a Mars vehicle's mass is propellant, which for NTP systems would be cryogenic hydrogen, the least dense of liquids. Consequently, NTP Mars vehicles are dominated by large tankage systems. In these early designs, usually the large hydrogen tanks were to be delivered to orbit separately and attached together end-to-end, with the pressurized tanks themselves serving as support structure, carrying the engine thrust loads during the relatively short thrusting periods. Many vehicle designs of this period were alike in their multistage, multiengine approach; typically, each stage had a nuclear engine or cluster of engines that was to be jettisoned with the propellant tank set when emptied. As an individual propulsion unit, a modular NTP stage was envisioned to accomplish lunar transfer and Earth orbit transfer missions as well. Designs of that era represent the first-generation vehicle archetype, which remained essentially unaltered until analysts devoted attention to configuration issues in the late 1980s time period of renewed interest in human Mars missions.

Serious technical literature describing Mars missions begin to surface in the late 1940s. Wernher von Braun's *Das Marsprojekt* study was published in 1949 and eventually gained much popularity when recast in the *Colliers* magazine articles in 1952, in part because of the spectacular spacecraft and planetary scene illustrations of Chesley Bonestell. For this early work, von Braun chose chemical propulsion (nitric acid and hydrazine) for the Earth–Mars transfer vehicle.

Mars Architectures: First Wave of Development

In-depth technical conceptual development of Mars transfer vehicles began in the later 1950s. Krafft Ehriche led investigations of Mars mission strategies, producing a variety of technical papers on interplanetary flight, beginning in 1957.[3] Ehriche published additional material in the late 1950s and early 1960s as a study contractor in one of the earliest series of funded manned Mars mission studies, entitled the "EMPIRE" studies.[4–6] In 1953, Project ROVER was begun to develop nuclear propulsion. This program reached fruition in the mid and late 1960s when the NERVA (Nuclear Engine for Rocket Vehicle Application) series of nuclear thermal engines were built and tested at a Nevada test facility.[7]

The early EMPIRE studies were followed in the mid and late 1960s by a variety of studies. Several aerospace companies conducted comprehensive evaluations during this period.[8–11] At this time von Braun was an influential advocate of human Mars exploration and of nuclear propulsion. In 1969 the Space Task Group[12] presented its recommendations[13] to Congress for a manned Mars mission as a subsequent focus after the Apollo program. Information presented was based on a NTP manned Mars vehicle flying a short-stay-time opposition mission as an initial Mars mission. (Another long-range planning study of 1969 made a similar recommendation.[14]) This proposal was rejected, however, and Mars mission planning was soon dropped, and the NERVA test program was canceled. The last of the in-depth Mars vehicle design studies[15] concluded in 1973. It would be 15 years until further in-depth sponsored contractor studies would resume.

Mars Architectures: The Second Wave

In the late 1980s plans for a renewed program of human exploration were announced; subsequently, renewed evaluations of manned Mars missions were begun. Major technical studies were done in the 1980s and early 1990s; vehicles in this period were typically configured to fly what is referred to as "split–sprint" mission profiles. The missions were split into two major elements, the first being a cargo-only mission, with the payload consisting of the crew's surface infrastructure (surface habitat, consumables and science equipment) launched on a low-energy, relatively long trip

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time (over 200 days) and one-way trajectory. After a successful autonomous checkout of the landed surface systems, a second, piloted mission would follow, utilizing a higher-energy, faster "sprint" trajectory. The piloted vehicle would carry the crew's transfer habitat, Mars ascent stage, and an Earth crew return vehicle. After arrival on Mars, the crew would land close to the habitat and transfer to the quiescent surface habitat. Trajectories were of either opposition or conjunction types, featuring 30- to 60-day or 500-day surface stays, respectively. Typically, mission planners followed a strategy that kept for the crew the option for a propulsive abort path for the return to Earth, should the surface habitation system malfunction during the crew's outbound transit.

Comprehensive mission scenarios were envisioned that would enable missions within 15 years given proper funding. Four transfer propulsion technologies were deemed viable for the missions under investigation: NTP, nuclear electric propulsion (NEP), solar electric propulsion (SEP), and chemical propulsion with aerocapture at Mars. The most comprehensive of these late 1980 era studies was *Space Transfer Concepts and Analysis for Exploration Missions Study*,[16] which ran from 1989 to 1994. Analysts also investigated lunar scenarios as precursors to, and as technology and operability demonstrations for, the follow-on Mars missions. Some studies at this time emphasized conceptual development of NTP and NEP for the Earth–Mars transfer stages. The favored NTP systems of the past began to evolve into systems in which the reactor would provide both thrust and electrical power. Termed the "bi-modal" NTP, this technology showed promise for applicability to all mission opportunities throughout the Earth/Mars 15-year cycle.

Another emphasis at this time involved understanding the dependencies of the transfer stage design to packaging constraints imposed by the Earth-to-orbit launch vehicles (LVs). During this interval several suitable heavy-lift launch vehicle concepts would come under study. These included the National Launch System, the Advanced Launch System, the Shuttle-derived "Shuttle-C" concept, and several others. Launch vehicles with payload capabilities of on the order of 250 metric tons to low Earth orbit (LEO) were considered. Means of minimizing Mars transfer stage on-orbit assembly were developed.[1]

Though the split–sprint mission architecture received much of the emphasis during this period, other innovative architectures were evaluated as well, including those conceived for efficient Mars base resupply and personnel rotation missions (the "dash–flyby" mode[17]), those conceived for absolute minimum initial mass in LEO (IMLEO), and those for far-term, human-settlement scenarios in which continual human presence on Mars was the objective and settlement self-sufficiency was the eventual goal. Some authors would arbitrarily intimate that human settlement was the best motive for Mars exploration. Other rationales, such as for economic, political, and educational advantage, were also discussed. Several recent architecture evaluations have set forth rationales as preludes to the follow-on technical analysis. In contrast, the earlier studies of the 1960s tended to view human Mars missions as exploration for exploration's sake and therefore tended to place value on the scientific scope and return of the missions. Regardless of what rationale is championed, mission analysts typically present a sequence of missions designed to gradually build up assets on the Martian surface so that at some point the basic infrastructure could be in place to support a permanent presence on Mars.

Architecture Evaluation Criteria

Assessing a particular architecture as superior to another depends on how the benefits and advantages of each are valued. A significant part of the task of architecture evaluation rests on determining and prioritizing the relevant criteria. For this study, the criteria used for assessing human Mars missions are given in Table 1.

Current Design Reference Mission: Third Wave of Architecture Development

The reference Mars mission architecture recently studied at NASA is called the Mars Design Reference Mission (DRM). This architecture is characterized by its in situ propellant production (ISPP) feature. Primary ground rules and assumptions were taken from NASA material generated by in-house exploration teams. These architectures (summer of 2000) are evolutions of DRM assessments initially begun in 1992.

Table 1 Criteria for evaluation of mars architectures

No.	Criterion
1	Minimum major technology development programs
2	Acceptable development and operational risk
3	Low or reasonable first mission cost
4	Commonality of overall architecture with other space activities
5	Evolution to low recurring cost missions
6	Multiple-use technology developments, benefiting society on Earth
7	Minimum mass in LEO

1990 Mars Direct Architecture

The Mars Direct architecture was introduced in 1990 (Refs. 18 and 19) by Zubrin and Baker. Mars Direct emphasized in situ propellant production, and an "abort-to-Mars-surface" philosophy, which will be discussed in more detail later. Also, elimination of Mars orbit rendezvous of stages was desired. [In situ propellant production for Mars missions had been presented as early as 1978 (Ref. 20).] In the Mars Direct plan, an unmanned, combination ascent stage/Earth return vehicle is sent out and is prepositioned on Mars. It carries an inbound transfer habitat for the return leg. Also sent on this mission is an ISPP plant and a nuclear power source. Using the Mars atmosphere, the ISPP plant begins to manufacture the oxygen and methane propellant that will be required to eventually deliver the crew back to Earth. This production is completed within approximately one year, which is several months before the crew's scheduled departure from Earth. Excess oxygen and methane can also be used to power rovers for extensive surface exploration.

Mars atmosphere consists mainly of carbon dioxide. Several options exist for propellant production. One option is to dissociate carbon dioxide (CO_2) into carbon monoxide (CO) and oxygen (O_2). Both can be liquefied and burned in a rocket engine with an estimated specific impulse I_{sp} of about 250 s. The second is to use the O_2 from this dissociation process with hydrogen (H_2) or other fuel brought from Earth. The third is to react liquid H_2 brought from Earth with Martian CO_2 to produce methane (CH_4). Oxygen is a byproduct, and additional O_2 can be produced to obtain the optimum mixture ratio for an O_2–CH_4 rocket engine. These processes have been demonstrated in the laboratory environment, and some are industrial processes on Earth. The O_2–CH_4 engine would have a maximum I_{sp} of about 380 s.

The difficulty of storing liquid hydrogen on Earth arises because liquid hydrogen is cold enough to liquefy Earth's major atmosphere constituents, whereas liquid oxygen and methane are not. Therefore, insulation systems are required to keep air away from hardware in contact with liquid hydrogen; otherwise, heat leaks are catastrophic. On Mars, however, any of these three cryogenic propellants (H_2, O_2, and CH_4) are cold enough to freeze atmospheric CO_2 and so they are, in one respect, qualitatively the same. Without the qualitative difference between H_2 and CH_4 that exists on Earth, the difficulty of storage is probably best measured by the amount of heat input needed to evaporate and warm a unit volume of liquid. By this measure hydrogen is more difficult than methane by a ratio of 7:5, i.e., not much worse. When the thermodynamic advantage of hydrogen for supplying vapor-cooled shields as an insulator for ascent stage tankage is considered, methane may not have significant storage advantage. Hydrogen needs more energy to liquefy and refrigerate, if that is required, but much less hydrogen is needed. By selecting methane fuel, Mars Direct gives up a significant I_{sp} advantage (about 100 s) and much O_2/H_2 engine technology heritage.

The in situ refueling process would use automated propellant production and relatively simple robotics. After fueling is completed on the surface of Mars for the first vehicle, the second, a piloted, one-way transfer stage, would depart Earth, travel to Mars, and use direct entry aerobraking to land near the quiescent ascent stage/Earth return vehicle. (The crew's outbound habitat also serves as their surface habitat.) The return vehicle must function autonomously for 4 years, which is the period from its Earth departure to the crew's Mars departure. One of those years would be dedicated to robotic propellant production, which would depend on nuclear surface power, and which would be autonomously deployed and operated. This high level of autonomy represents a challenge to implementing this architecture. See Fig. 1 for an iconographic depiction of the mission sequence. Table 2 lists the major payload elements transported on each vehicle. Chemical propulsion or NTP can be used for trans-Mars injection (TMI). Chemical O_2/CH_4 is used for the Earth return vehicle. No element is parked in either Mars or Earth orbit. Because it was a goal to eliminate Earth orbit assembly and operations, each of the two transfer stages would be launched directly into TMI atop a large launch vehicle.

Because of the nature of Mars Direct, some abort options are compromised. Should the descent stage or the surface habitat develop a malfunction while the crew is enroute to Mars (either failure would

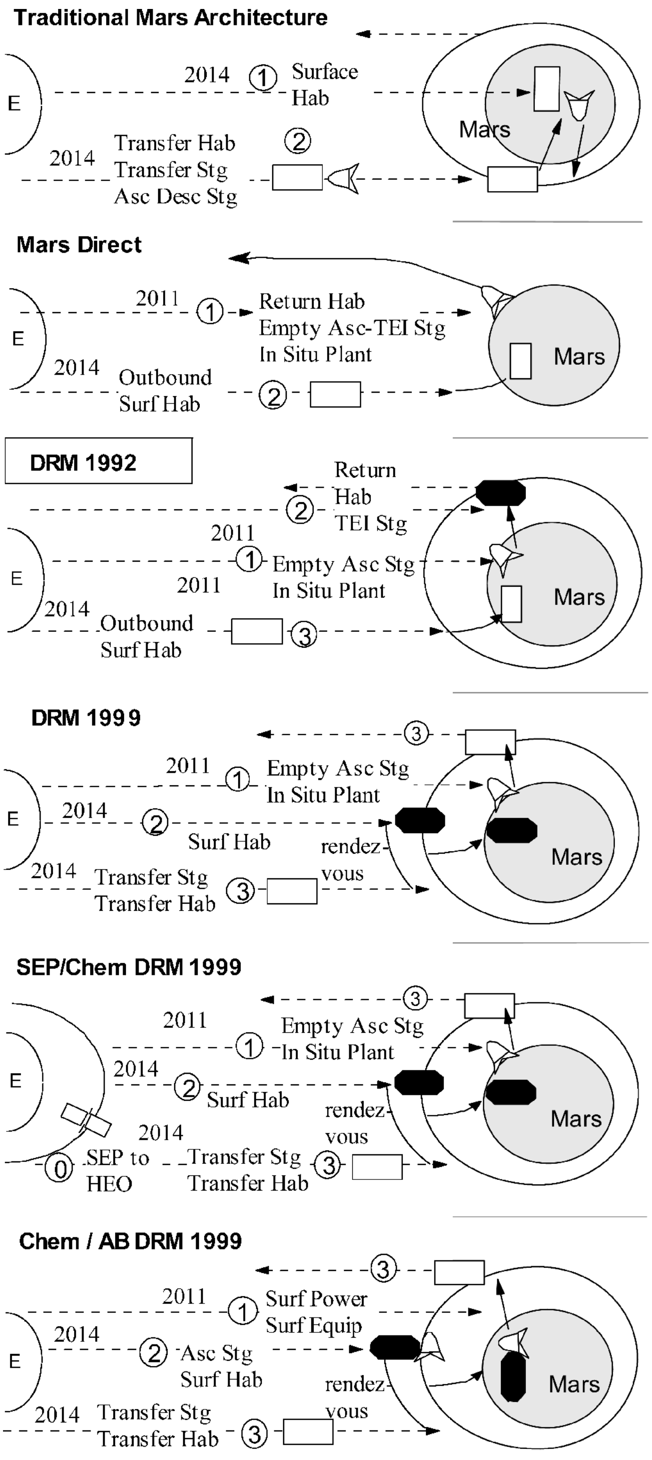

Fig. 1 Architecture mission sequence diagrams.

Table 2 1990 Mars direct mission

Mission name	TMI/MOC propulsion	Major payload elements	Launch vehicle	Year
Cargo 1	NTP/Ab	Empty ascent/TEI stage ISPP + power Liquid H_2 cache Return trip habitat	1 @ 120 metric tons to LEO	2011
Piloted	NTP/Ab	Transfer/surface habitat	1 @ 120 metric tons	2014

Table 3 1992 Mars DRM

Mission name	TMI/MOC propulsion	Major payload elements	Launch vehicle	Year
Cargo 1	NTP/Ab	TEI stage return trip habitat	1 @ 200 metric tons to LEO	2011
Cargo 2	NTP/Ab	Empty ascent stage ISPP + power Liquid H_2 cache	1 @ 200 metric tons to LEO	2011
Piloted	NTP/Ab	Transfer/surface habitat	1 @ 200 metric tons	2014

preclude a landing), the crew would be without the option of effecting a propulsive burn at Mars to return to Earth. (The piloted Mars transfer stage does not contain Earth-return propellant.) Mars Direct requires a Mars-free return trajectory to cover such a contingency. In such cases the vehicle, with the aid of a (nonpropulsive) Mars swingby gravity assist, would eventually intercept the Earth, though the return trip time can be excessive when compared to a nominal return. Free return inbound legs of the order of 750–900 days are typical. Also problematic are the resulting increased Earth atmospheric entry velocities. By separating the crew from their ascent stage, some additional risk is incurred. The ascent stage waits on Mars two years before the crew arrives, and then the crew must land near the ascent system to survive. A guidance or navigation error on descent can be fatal, and abort from the descent is not possible.

1992 DRM: Addressing the Deficiencies of Mars Direct

In 1992, NASA adopted a modified version of Mars Direct as a reference mission for conducting mission planning and trade studies.[21] This reference mission contained most of the primary elements of Mars Direct, including an ISPP ascent stage, methane fuel, abort-to-surface, and the lack of Earth orbit operations. Launch vehicle payload capabilities were similar to those required of Mars Direct, on the order of 120–200 metric tons.[22] Unlike Mars Direct, however, the Earth return habitat is not available to the crew during their 1.5-year surface stay. The single ascent/Earth return stage of Mars Direct was dropped in favor of separate ascent and Mars departure stages; thus, Mars orbit rendezvous is necessary [between the ascent and trans-Earth injection (TEI) stages].

This 1992 DRM proposed three transfer stages rather than two as in Mars Direct. Two cargo flights depart for Mars during the first opportunity, one to Mars orbit and the second to the surface, in preparation for the crew during the following opportunity. Each TMI stack was to consist of a cargo/payload portion and a propulsion stage for performing the departure delta velocities dV to get on to the appropriate Mars trajectories. For this architecture, three 66,700-N (15-klbf) thrust, 960-s I_{sp} NTP engines were utilized with a single H_2 tank for the TMI stage for each stack.[23] The first proposed cargo mission delivers a TEI stage and the crew's return habitat to Mars orbit, the second an in situ ascent craft to the surface, and a third the crew to the surface in the combination transfer/descent stage carrying an outbound/surface habitat (Fig. 1 and Table 3). (The Earth return and ascent stages would operate in an autonomous mode for about 4 years until the time of their utilization.) Methane fuel was selected for TEI to achieve engine commonality with the ascent vehicle. Another two cargo missions, delivering TEI and ascent vehicles for the next mission opportunity, arrive at about the same time as the first crew vehicle. These two vehicles are available as backup units to the crew. At this point in the timeline, within 3 years of the first cargo vehicle's departure, three vehicles, each carrying transit habitats, are on Mars.

Table 4 1999 Mars DRM

Mission name	TMI/MOC propulsion	Major payload elements	Launch vehicle	Year
Cargo 1	NTP/Ab	Surface habitat	2 @ 80 metric tons	2011
Cargo 2	NTP/Ab	Empty ascent stage ISPP + power Liquid H_2 cache	2 @ 80 metric tons to LEO	2011
Piloted	NTP/NTP	TEI stage Transfer habitat	2 @ 80 metric tons to LEO	2014

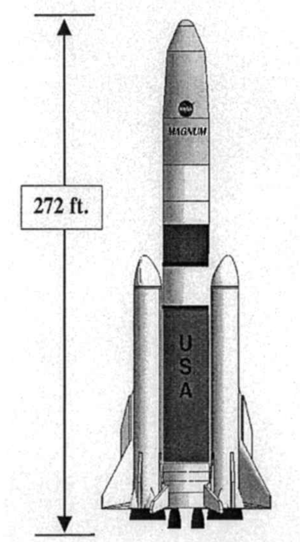

Fig. 2 Magnum launch vehicle.

1997 DRM: Further Refinement

In 1997, the DRM was again modified.[24] Removing the prohibition against Earth orbit operations allowed a repackaging of mission elements with a reduction in the required payload capability of the launch vehicle. Delivering the TMI stage and its Mars cargo (with its aeroshell) to LEO separately allowed the change to a more modest 80-metric-ton class launch vehicle. These two elements would need only joining, not assembly, once on orbit. While doubling the number of Earth-to-orbit (ETO) launches, the strategy eliminated the high costs of developing a much larger launcher. The *Magnum* shuttle-derived launch vehicle concept (Fig. 2) was developed to boost 85 metric tons to a 400-km orbit at 28.5-deg inclination. The *Magnum's* payload bay size was 7.6 m in diam × 28 m in length. Those ETO flights carrying the Mars payload element would utilize the Mars aerobrake as their shroud (the biconic shape of the aerobrake being very similar to the desired shape of a shroud otherwise required for ETO ascent). The aerobrake performs the Mars orbit capture (MOC) maneuver, which would entail its second use. In the case of the surface payloads, a subsequent aerodescent is done with the same shell, i.e., its third use.

1999: DRM with Round Trip Capable Transfer Stage

In a 1999 revision, a significant change occurred via combining the once-separate outbound and inbound transfer functions. Now a single, round trip capable crew transfer stage was baselined (Fig. 1 and Table 4). Where before the crew was to travel out in the surface habitat and conduct a direct entry aerodescent to the surface, now the crew transfer stage propulsively captures into Mars orbit then would rendezvous with an earlier-arriving, coorbiting transfer stage; its payload is a surface habitat/descent-stage combination. The crew would then transition to the habitat and descend to the surface to land near the ascent stage. For this architecture the piloted vehicle uses NTP for each major burn: TMI, MOC, and TEI. For the two cargo missions, NTP is used only for TMI, whereas MOC is done with aerocapture.

This 1999 revision, current as of this writing, is much like the traditional split–sprint architecture described earlier. Several of the major features of the Mars Direct architecture, except in situ propellant production, have been dropped; the abort-to-surface philosophy, one of the most distinctive of its elements, is no longer a feature. With a propulsive Earth return capability for the crew, free-return trajectories would no longer be mandatory (but nevertheless might still be retained). Earth orbit operations and Mars orbit rendezvous are now baselined. Overall, six launch vehicles put up three TMI stages and three payload elements. These six elements are joined on orbit, forming three vehicles. Total fleet IMLEO for these three vehicles is 419 metric tons. Although NTP was baselined, two other propulsive options were evaluated in 1999, a combination SEP/chemical propulsion option and an all chemical-aerobrake (Chem-AB) option.

1999 DRM Solar Electric Propulsion/Chemical Option

In the SEP option, efficient electric propulsion is combined with solar power technology to provide a partial TMI boost to each of the three departing transfer stages (Cargo 1, 2, and Piloted; Fig. 1). A SEP stage would do the bulk of the LEO to TMI transfer, boosting the Mars payload to a final high Earth orbit (HEO). A small chemical stage would provide the remaining dV to surpass Earth escape velocity and reach required TMI velocity. Each of the transfer vehicles would aerocapture at Mars. Because low thrust transfers to HEO take several months, the crew would not board the transfer stage until the higher orbit is reached. A small, high-thrust chemical propulsion space taxi stage[25] would deliver the crew to HEO. After rendezvous, the SEP stage separates and returns to LEO, is refueled in orbit, and performs one additional LEO–HEO transfer during another opportunity before it is expended. A 800-kW power system and 2,500-s I_{sp} thruster performance characterize the SEP stage. Fleet IMLEO for the required SEP and chemical Mars transfer stages is 410 metric tons; six launch vehicles are required to place these in LEO.

1999 Chemical-Aerobrake Version of DRM

Chemical/aerobrake architectures have been studied to some detail in the past. This implementation of the current DRM was configured to lower the total program costs by reducing the number of required new technology developments. The Chem-AB option selects one propulsion technology for all propulsive maneuvers. Current RL-10 O_2/H_2 engine technology (466-s I_{sp}) is baselined. The development of more costly advanced propulsion technologies, such as NTP, NEP or SEP, could be considered later, as evolutionary steps from this initial, less-costly approach. The mission plan is much the same as just described for the 1999 DRM; however, the first ascent stage to be delivered to Mars goes out fully fueled. This change represents a significant operational difference because the first mission to Mars, in this strategy, would not be dependent on predeployed autonomous ISPP. Subsequent ascent stages would have oxygen supplied via ISPP. In fact, the first in situ plant would arrive on Mars while the first crew was on the surface. There the crew would monitor the process of filling a second ascent stage for utilization on the next piloted opportunity. For this option H_2 fuel is brought from Earth; methane is not used. For the Chem-AB option the first two cargo missions departing in 2011 consist of a single TMI propulsive stage that boosts a one-way payload enclosed in a biconic aerobrake. The third cargo and piloted transfer vehicles (illustrated in Fig. 3) departing in 2014 have greater payloads such that two of the common propulsive stages are utilized for TMI. See Fig. 4 for a mission timeline. The TMI stages are designed to the

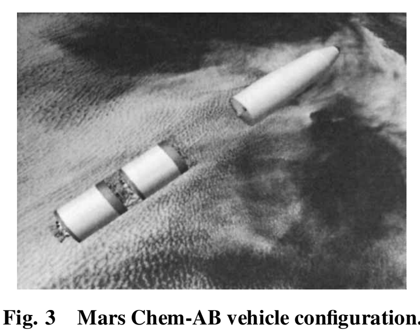

Fig. 3 Mars Chem-AB vehicle configuration.

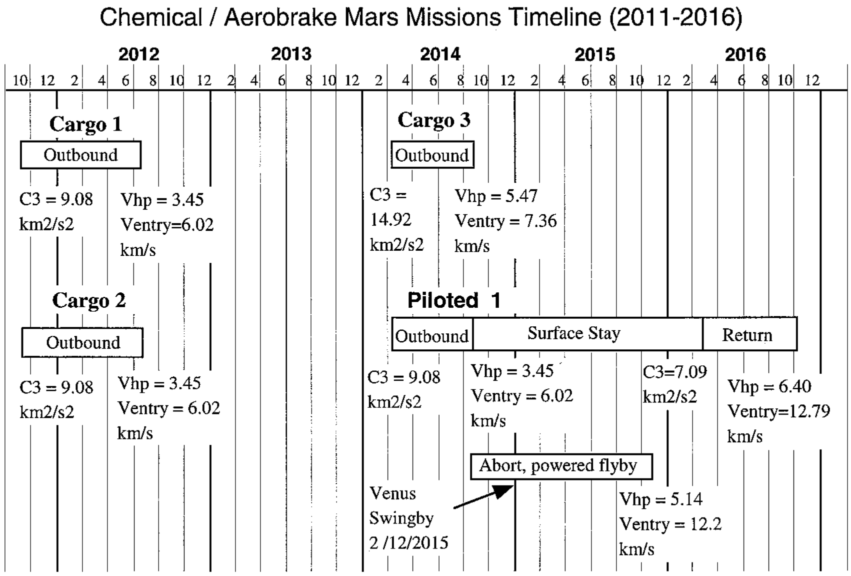

Fig. 4 Mars mission timeline 2011–2016.

Table 5 1999 Chem-AB DRM

Mission name	TMI/MOC propulsion	Major payload elements	Launch vehicle	Year
Cargo 1	B stage/Ab	Power, surface payloads	1 @ 80 metric tons	2011
Cargo 2	B stage/Ab	Power, surface payloads	1 @ 80 metric tons	2011
Cargo 3	A + B/Ab	Surface habitat Full ascent stage	3 @ 80 metric tons to LEO	2014
Piloted	A + B/Ab	TEI stage Transfer habitat	3 @ 80 metric tons to LEO	2014

Table 6 Chem-AB propulsion stage weight

	Mass, kg			
Element	TMI A[a] RL-10C[b]	B RL-10B2	TEI RL-10B2	Ascent RL-10B2
Structure	4,239	4,239	2,048	2,048
Tankage/insulation	2,959	2,959	1,235	1,235
Engines (3)	1,512	884	884	884
Propulsion related	645	437	419	419
Reaction control system (RCS) related	n/a	319	319	319
Avionics/power	475	475	200	n/a[c]
Weight growth	1,074	1,074	432	432
Total dry weight	10,903	10,387	5,537	5,337
RCS propellant	n/a	955	923	195
Reserves/residuals	2,060	2,031	884	638
Main propellant	67,982	67,007	29,166	17,755
Stage gross weight	80,944	80,380	36,510	23,923

[a]Stage type. [b]Engine. [c]Not applicable.

capability of the launch vehicle; their maximum mass is 81.5 metric tons. Because the triconic aerobrake can be utilized as part of the launch vehicle shroud, an estimated 2.5-metric-ton increase in capability of the launch vehicle (to 84 metric tons) can be realized for the payloads encapsulated in the dual use shroud/aeroshell. Payloads are delivered in the order given in Table 5.

The Chem-AB DRM begins with the placement of dual 60-kW nuclear power generators, along with other surface equipment with the 2011 Cargo 1 and 2 missions. Alternatively, these missions could deliver a 24-kW solar electric power system to Mars, providing minimal power for ISPP. There remains the question of how to robotically deploy an array of solar cells of many thousands of square meters. During the 2014 opportunity, a third cargo mission is launched, carrying the surface habitat and a fueled ascent stage.

The common TMI chemical propulsive stage weighs 81.5 metric tons fully loaded with propellant. For each of the two cargo missions of 2011, only one TMI stage is required. These departures do not

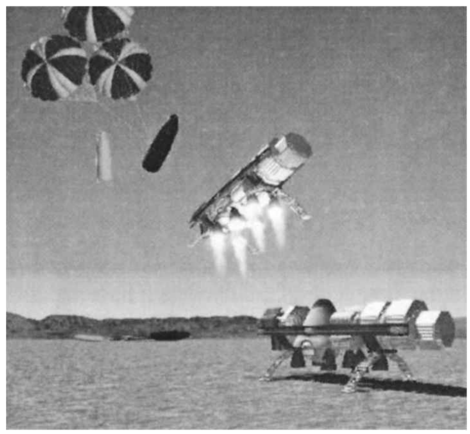

Fig. 5 Mars lander illustration.

require a full propellant load, and the tanks are delivered to orbit off-loaded. In this case the TMI stage weight is 58.0 metric tons. For the heavy 2014 Cargo 3 and piloted missions, two fully loaded stages are used in tandem; these are identical with the exception of the engines. The first to burn ("A" stage) utilizes three 222,400-N (50,000-lbf)-thrust RL-50 engines and the follow-on "B" stage that completes the TMI burn utilizes three 110,000-N (24,700-lbf)-thrust RL-10 B2 engines. See Table 6 for stage A and B weights.

The TEI stage is sized for the worst-case Earth return opportunity of 2022. The tanks and engines are common with the ascent stage. The descent stage uses the aerobrake during the first portion of descent, a supersonic multichute parachute system to reduce the velocity further, and RL-10 engines to provide the final 632 m/s (Ref. 26) of dV to hover and touchdown. Because the descent stage is packaged in a conical aerobrake, a horizontal configuration is baselined (Fig. 5). (An alternate lander configuration that features a vertical stacking arrangement is shown in Fig. 6.) A comprehensive list of system weights and payload elements is given in Table 7. The Chem-AB option requires eight launch vehicles, which together deliver a total of 668 metric tons to LEO.

In Table 8, the Chem-AB option is compared to the NTP and SEP alternatives. The comparisons are based on several criteria, including the number of *Magnum* launches and IMLEO required for one complete 2011–2014 mission. General propulsion-related advantages and disadvantages of each are listed in Table 9. Chem-AB has the advantages of utilizing off-the-shelf propulsion and lower development costs but requires more IMLEO than the others. SEP offers good reusability but requires a large area for its arrays, which would suffer some decay in performance as a result of Van Allen

Table 7 Chem-AB mission system masses

	Description for mission				
Parameter	Cargo 1	Cargo 2	Cargo 3	Piloted 1	Total
Earth departure date	8 November 2011	8 November 2011	27 December 2013	10 January 2014	
TMI dV (with g losses) (km/s)	3.605	3.605	3.661	3.768	
C3 with 1 engine out (km^2/s^2)	9.08	9.08	9.03	10.98	
Outbound trip time (days)	265.5	265.5	207	193	
Mars arrival Vhp/(allowed km/s)	3.45/5.42	3.45/5.42	5.42/5.42	5.42/5.42	
TEI to ascent rendezvous dV (km/s)	n/a[a]	n/a	n/a	0.500	
Mars depart date/dV (km/s)	n/a	n/a	n/a	December 2015/1.615	
Earth arrival date/Vhp/allowed (km/s)				12 July 2016/6.4/6.9	
Total mass in LEO (metric tons)	84.0	84.0	245.0	254.6	667.6
Magnum launches	1	1	3	3 + shuttle	8
Transfer stages	B	B	B + A	B + A	4B + 2A
Trans Mars injection stage, metric tons					437.4
Stage 1 dry weight	10.9	10.9	10.9	10.9	43.6
Reserves/residuals	1.0 B off loaded	1.0 B off loaded	2.0 B	2.0 B	6.0
Usable propellant	45.1	45.1	68.0	68.0	226.2
Stage 2 dry weight	n/a	n/a	10.9	10.9	21.8
Reserves/residuals			2.0 A	2.0 A	4.0
Usable propellant			68.0	68.0	136.0
Payloads of Mars orbit	27.0	27.0	83.2	92.8	230.0
Stages	n/a	n/a	27.7 (ascent)	36.5 (TEI)	64.2
Stages	8.0 (descent)	8.0 (descent)	20.7 (descent)	n/a	36.7
ECRV + crew	n/a	n/a	n/a	5.8	5.8
Aeroshells	5.5	5.5	13.4	13.4	37.8
Habitats	n/a	3.0 (lab module)	15.5 (surface)	20.8 (transit)	39.3
Consumables	3.6	3.9	0.5	12.2	20.2
Power	5.3 (nucl surface)	5.3 (nucl surface)	4.5 (solar surface)	3.5 (transit)	18.6
Science (surface/transit)	1.1 (surface)	n/a	0.7 (surface)	0.6 (transit)	2.4
Other	1.2	1.3	0.2	n/a	2.7
Rovers	2.3	n/a	n/a	n/a	2.3

[a] Not applicable.

Table 8 Architecture comparison by general criteria[a]

Propulsion system	Number of ETO launches	Total fleet IMLEO, metric tons	Commonality of main propulsion	In situ req't first mission	Years between ascent stage Earth departure and utilization
SEP/Chem-AB	6	410	None	Yes	4
NTP	6	419	TMI-MOI-TEI	Yes	4
Chem-AB	8	668	TMI-TEI-descent-ascent	No	2

[a] Comparison based on a typical Mars mission with 80 mt class payload ETO.

Table 9 Mars transfer vehicle main propulsion comparisons[a]

System	Advantages	Disadvantages
Cryo/AB	**Lower development cost**	**High IMLEO**
	Adequate redundancy	Sensitive to variations in mission profile requirements
	Good reusability potential if operated from L2 node	May require orbital assembly of large aerobrake, with rigorous verification requirements
	A large low-energy aerobrake is required for Mars landing with any propulsion option	Needs accurate terminal navigation at Mars for successful aerocapture
SEP	**Fully reusable**	Susceptible to radiation damage in van Allen belts
	Lower IMEO after first trip	High power levels (MWs) required for reasonable trip times
	Trip times competitive with cryo/aerobrake	**Large area required for solar array**
	May offer low development cost	Array production cost may be too high
	Less sensitive to launch dates, windows	Variable power over trajectory
	Development synergy with SSP systems	Small solar flux for distances between Mars
	Power supply at destination	
	Eliminates risk of large, high-energy aerobrakes	
NTP	**Lower IMLEO**	Nuclear systems in ETO launch
	Trip times about 400–450 days possible all opportunities	**High cost development**
	Broader launch windows than chemical systems	
	Technology demonstrated through ground test	
	Eliminates development and risk of large, high-energy aerobrakes	
	Reusable and restartable	
	Simpler trajectory design	
NEP	**Fully reusable**	Operated from HEO for competitive trip time
	Lower IMEO after first trip	Limited redundancy and long operating times
	Faster trip times at high power, <200 days each way	**Dynamic power conversion required**
	Power supply at destination	Nuclear power requires further technological development
	Eliminates devel and risk of large aerobrakes	High power levels required for reasonable trip times (10 MW)
	Less sensitive to launch dates, windows	Nuclear systems in ETO launch
	Power source independent of solar distance	

[a] Major advantages and disadvantages are shown in boldface type.

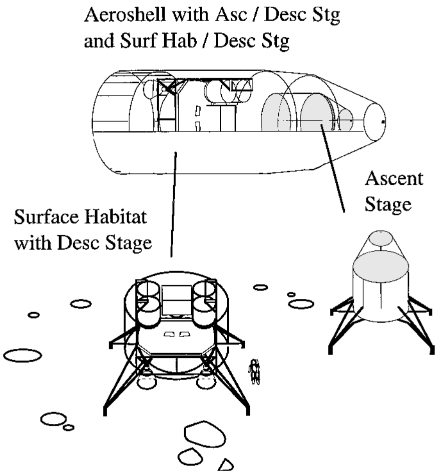

Fig. 6 Alternate lander configuration.

belt passage. Nuclear systems provide both lower IMLEO and faster trip times but may incur high development costs, particularly in the area of ground test facilities. Developing either SEP or nuclear systems would preclude the development and risk of large high-energy aerobrakes.

Conclusions

Current and past Human Mars mission architectures were investigated. For the current DRM, a lower number of launch vehicles is required for the NTP and the SEP variations than for the Chem-AB option, primarily due to the higher propellant requirements of its lower I_{sp} propulsion system, which in this case is relatively inexpensive LO_2 and LH_2. Advantages of the Chem-AB, however, are several. First, fewer new technology developments are required; second, first mission set cost would be lower; finally, off-the-shelf engines would be used for all but Mars orbit capture, precluding the need to develop methane-based propulsion systems as baselined for the other variations. The Chem-AB option proposed here is not dependent on ISPP for the first mission. ISPP is assumed for subsequent missions, and its first use would occur with the crew on site monitoring the operation. If water is easily accessible on Mars, ISPP may be based on water electrolysis rather than methane production from the atmosphere. This water electrolysis scheme for ISPP lends itself to higher I_{sp} ascent propulsion based on liquid oxygen/LH_2. Because the first mission is not ISPP dependent, the crew can descend to the Mars surface in a fully fueled ascent/descent stage, retaining full abort-back-to-orbit capability.

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