BENJAMIN D. DONAHUE M/S ZA-06 256/971-3257
benjamin.b.donahue@Boeing.com

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HUMAN MARS MISSIONS: COST DRIVEN ARCHITECTURE ASSESSMENTS

Benjamin Donahue*

Boeing Space Systems, Huntsville, AL 35824

ABSTRACT

This report investigates various methods of reducing the cost in space transportation systems for human Mars missions. The reference mission for this task is a mission currently under study at NASA, called the Mars Design Reference Mission, characterized by *In-Situ* propellant production at Mars. This study mainly consists of comparative evaluations to the reference mission with a view to selecting strategies that would reduce the cost of the Mars program as a whole. One of the objectives is to understand the implications of certain Mars architectures, mission modes, vehicle configurations, and potentials for vehicle reusability.

The evaluations start with year 2011-2014 conjunction missions which were characterized by their *abort-to-the-surface* mission abort philosophy. Variations within this mission architecture, as well as outside the set to other architectures (not predicated on an abort to surface philosophy) were evaluated. Specific emphasis has been placed on identifying and assessing overall mission risk. Impacts that Mars mission vehicles might place upon the Space Station, if it were to be used as an assembly or operations base, were also discussed. Because of the short duration of this study only on a few propulsion elements were addressed (nuclear thermal, cryogenic oxygen-hydrogen, cryogenic oxygen-methane, and aerocapture). Primary ground rules and assumptions were taken from NASA material used in Marshall Space Flight Center's own assessment done in 1997.

INTRODUCTION

This report was written in support of the Boeing NASA/MSFC studies relating to Affordable In-Space Transportation (AIST) concept definition in 1998. Various methods of reducing the cost in space transportation systems are investigated. The reference mission for this task is a human Mars mission currently under study at NASA. The current Mars Design Reference Mission is used for reference purposes and as a basis for comparison to alternative transportation systems and architectures. One of the objectives of this report is to understand the implications of certain Mars mission architectures, transportation vehicle configurations, mission enhancing technologies and mission modes. The primary approach is to start with evaluations for conjunction missions with abort to the Martian surface, studying the 2011-2014 opportunity. Further evaluations were done for variations within this mission set, as well as outside the set to other architectures that are not predicated on an abort to surface philosophy.

STUDY OBJECTIVES

This short study centered mainly around the Mars mission elements. The Mars Design Reference (DRM) was used as a starting point. It will be described, along with its significant elements, so as a fair evaluation and comparison can be made to other missions. This is done primarily with a view to determining cost savings that may be made across the DRM architecture or by adopting a different Mars mission architecture. Variations are proposed at specific points as a means of contrasting the relative cost of adopting new technologies into a mission set. It is the goal of this study to focus on a few significant transportation system elements (such as propulsion type, use of aerocapture, abort philosophy, capture orbits, etc). Primary ground rules and assumptions were taken from NASA material used in MSFC's own assessment of the DRM done in 1997.

MARS DESIGN REFERENCE MISSION DESCRIPTIONS

In the Mars DRM, an Earth return transfer propulsion and habitat system is prepositioned on Mars and refueled by methane and oxygen produced from Mars atmosphere (with the aid of a modest amount of hydrogen brought from Earth). The refueling process uses automated propellant production and relatively simple robotics. The mission crew transfers to Mars, bringing their transfer habitat to Mars surface via aerobraking for use on the surface during the 500-day stay. With a large enough launch vehicle, no Earth orbit assembly is required. Also, no Mars orbit operations are required. Production of propellants from Mars' atmosphere is a greatly simpler proposition than production from Lunar regolith. An adequately robust habitation facility on the surface of Mars is more reachable as a safe haven than return to Earth for some portion of almost any Mars mission profile. The reference architecture employs three launches from Earth; one to Mars orbit and one to Mars surface for cargo delivery, and one to Mars surface for personnel delivery. See Figure 1 for a graphical representation of the three mission scenario. During the crew mission launch period, two cargo launches to support the following crew mission are also launched. In addition to surface cargo, the cargo missions deliver the crew ascent vehicle and the Earth return vehicle. The former is fueled with *in-situ* propellants after landing

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*Systems Configuration Designer

(but prior to the crew departing Earth) and the latter is parked in Mars orbit, fully fueled, awaiting use at the completion of the crew mission. Both the crew ascent vehicle and the Earth return vehicle use methane and oxygen propellants. Methane was selected for the Earth return for engine / propulsion commonality with the ascent vehicle and because the storage temperature for methane is about 70 K warmer than for hydrogen. The manned mission transfer module also goes on a direct trajectory to a Mars landing. The mission vehicle includes an aerobrake for Mars arrival, a transfer / surface habitat, other small surface cargo such as a rover, and a descent stage. The transfer habitat becomes part of the surface infrastructure.

The habitat for the return transfer is already parked in Mars orbit before the crew leaves Earth because it is part of the Earth return vehicle. The Earth return vehicle and the ascent vehicle for the subsequent mission opportunity are delivered to Mars as cargo, arriving about the same time as the current crew. These are available as backup for the return trip. Several variations on this reference architecture have been described, but all are operated on this same basic concept. Inherent in the concept is the idea of a robust and redundant surface architecture that can be depended on as a safe haven. However, the arrangement of the architecture eliminates several abort modes. These will be detailed in a later section. The architecture introduces several new technologies at one time which must be regarded as risky from the program management point of view:

(1) Aerocapture at Mars
(2) *In-situ* propellant production
(3) Robotic assembly of the initial base under conditions of severe communications delay
(4) Nuclear thermal propulsion

Mars atmosphere consists mainly of carbon dioxide. Three means for propellant production are possible. One is to dissociate carbon dioxide into carbon monoxide and oxygen. Both can be liquefied and burned in a rocket engine with an estimated specific impulse of about 250 seconds. The second is to use the oxygen from this dissociation process with hydrogen or other fuel brought form Earth. The third is to react hydrogen brought from Earth with carbon dioxide to produce methane. Oxygen is a byproduct, and additional oxygen can be produced to obtain the optimum mixture ratio for an oxygen-methane rocket engine. These process have been demonstrated in the laboratory environment, and some are industrial processes on Earth.

MSFC MARS DESIGN REFERENCE MISSION DATA

Mission ground rules and assumptions as proposed by MSFC are given in Figs. 2-5. Fig. 6 lists MSFC DRM vehicle weights (from Ref. 1) for the two 2011 cargo flights and the single 2014 crew transfer flight.

BOEING MARS DRM DATA COMPARISON TO MSFC DRM DATA

Boeing vehicle design and weights estimating was done to match this 2011- 2014 mission set. Boeing vehicle weights data for the major vehicle elements are listed along side the MSFC estimates. Fixed payload values are circled. The Boeing generated values are given in bold print. For the transfer vehicles the Boeing values are only slightly higher (due to a more conservative allotment of propellant reserves), the Boeing ascent and descent stage dry mass estimates are heavier however (see "Mission 2" column of Fig. 6). The difference lies in the fact that the Boeing weights code sizes the ascent and descent stage load bearing structure (frame and landing legs) to support the *in-situ* ascent stage *after it is fully fueled* on the Martian surface. (Its fully fueled weight is four times its empty weight.) Figure 7 illustrates the 2014 piloted vehicle with the surface habitat module integrated into the descent vehicle.

COMPARATIVE EVALUATIONS

This DRM vehicle set and payload delivery capability form the baseline from which alternative concepts and approaches are presented in the remaining portion of this document. Understanding and quantifying the implications of other Mars mission architectures in relation to the DRM is important to identifying cost reduction techniques. Finding ways to reduce mission risk and to increase vehicle robustness is also a an objective. This study investigated variations to ascent vehicle propulsion type, mission mode type, transfer vehicle propulsion type, transfer vehicle configuration type, and propulsive vs aerocapture at Mars. Transfer vehicle recapture at Earth for reuse was also evaluated. A comparison was made of abort strategies. The topic of Mars vehicle on-orbit assembly is addressed, as well as potential impacts to the ISS, should it serve as an on orbit support station for assembly. The task work statement stipulates that NTP be used for the reference mission. Alternate systems offering potential for reduced risk, lower cost, lower mass, simplified operations, or some other benefit were considered. For this short study, the overall goal was to understand what elements constitute a flexible, cost effective, evolutionary in-space transportation program for NASA, and provide information necessary to proceed with initial system definition and planning. Only one Mars mission opportunity was evaluated, the year 2014 opportunity for the manned portion with 2011 being the date for launch of cargo flights for preemplacement of surface infrastructure. No trajectory optimizations were done to those supplied by NASA; all vehicles were sized based on the same mission delta velocities, trip times, boiloff rates, engine specific impulse, etc.

Variations to the Mars DRM included evaluating three other ascent propulsion systems vs the baseline in-situ O2/CH4 system. Another option considered the compression of the reference fleet size from 3 transfer vehicles to 2. The current DRM is not well suited to reuse, so an alternative mission mode was selected for reusability evaluations. Some of the concepts and descriptions are taken from the authors work on Boeing's Space *Transfer Concepts and Analyses for Exploration Mission* study. (Ref. 2)

ASCENT STAGE TRADE

The utilization of in situ propellant production (ISPP) for reducing the total payload mass required in Earth orbit has been advocated for Mars exploration missions since the 1970's (Ref. 3). ISPP received renewed interest in the early 1990's (Ref. 4, 5) and is a prominent feature of the present DRM. In the ISPP scenario, liquid oxygen propellant processed out of Martian atmospheric carbon dioxide (CO_2), and liquid methane (CH_4) propellant formed by combining the carbon extracted from the atmospheric CO_2 separation process with H_2 carried with the vehicle, would be supplied to an ascent stage on the surface. This vehicle would be deployed previous to the arrival of a crew on a separate lander stage. Once fueled autonomously on the surface, the vehicle is to be used to return the crew after their surface stay to Mars orbit for rendezvous with a Mars-Earth transfer stage. Since no ascent stage propellant need be transported from the Earth, the total payload required of ETO and Earth-Mars transfer systems can be reduced. Other variations to this scheme have been considered, including those in which only O_2 is produced at Mars as oxidizer to supply vehicles utilizing Earth supplied hydrogen, hydrogen-beryllium, monomethyl hydrazine or other fuel. Elements necessary for processing the O_2 and CH_4 out of the atmosphere include the in-situ propellant plant, surface power for the plant, seed H_2 for CH_4 production, tanks for the seed H_2, and support structure for these elements on the descent stage. The estimated weight for these elements are given in Table II below.

TABLE II IN SITU SYSTEMS

1.	In Situ plant	= 4.3 mt
2.	Seed H2 for CH4 prod	= 4.5 mt
3.	Seed H2 tanks	= 0.4 mt
4.	Lander support struct	= 0.4 mt

Once in Mars orbit, the crew and surface payload is emplaced on the surface by one or more landing craft. All but about 1000 m/s of the descent maneuver ΔV can be taken out aerodynamically by the aerobrake. The only large ΔV maneuver remaining to be considered is the Mars ascent ΔV; it is the most energy intensive of all the Mars maneuvers, requiring 5625 m/s to ascend to a 250 km periapsis by 1 Sol parking orbit for rendezvous with an orbiting transfer stage. This dV value exceeds the ideal escape velocity (5000 m/s) for Mars due to the excess velocity needed to offset gravitational and atmospheric drag losses. Ascent and descent stage performance and mass as a function of a variety of mission, vehicle, and subsystem related variables were estimated for five different ascent propulsion systems. Three of these use only Earth supplied propellants and 3 utilize ISPP. Included in the first group are the nuclear H_2, the chemical O_2/H_2 and O_2/H_2-Be stages. In the latter group are the O_2/H_2 and O_2/H_2-Be stages which benefit from ISPP O_2 oxidizer, and a O_2/CH_4 stage which utilizes ISPP oxidizer and fuel. These options are listed in Table III. Engine performance characteristics are listed in Fig. 8 for each propellant combination. Other assumptions are given below:

General Assumptions

- 5.5 mt ascent crew cabin mass
- 0.1 mt returned samples mass
- Aeroshell mass set to 16% (cargo) or 21% (piloted) of the total decelerated mass
- Single stage
- Weight growth of 15% for stage dry mass

Nuclear Ascent Stage

- Single nuclear engine utilized with a 1 mt dedicated radiation shield mass
- In line primary H_2 tank configured for supplemental shield radiation attenuation
- Engine jettisoned before rendezvous with orbiting transfer vehicle; remains in long lived Mars orbit

In Situ Ascent Stages

- Propellant plant and seed H_2 of 9.6 mt delivered with the ascent stage. This mass is counted as part of the descent stage in the weights calculations. Ascent tanks are topped-off prior to lift off to negate boiloff.
- Ascent and descent stage load bearing structure (frame and landing legs) sized to support fully fueled vehicle on the surface.

Ascent Stage Performance Results

Vehicle mass statements for the different ascent concepts are listed in Figure 9. This data is shown graphically in Figure 10, where the concepts are shown in ascending order by total lander weight. Three NTP ascent stage options are listed, showing stage weight variation with engine t/w. The NTP concept described herein was only analyzed utilizing H_2; other propellants could be used, including those that are in-situ produced. ISPP for NTP ascent warrants further study. A 22.8 mt descent payload was carried to the surface in addition to the ascent stage. In Fig. 9, ascent stage weights are given at Earth departure and Mars liftoff (columns 9 and 10).

TABLE III ASCENT STAGE TYPES

1.	Earth Supplied Oxidizer And Fuel		
	Chemical	H2	O2
	Chemical	H2-Be	O2
	Nuclear	H2	n/a

TABLE III ASCENT STAGE TYPES

2.	ISPP Oxidizer and Earth Supplied Fuel		
	Chemical	H2	O2 ISPP
	Chemical	H2-Be	O2 ISPP
3.	ISPP Oxidizer and Fuel		
	Chemical	CH4 ISPP	O2 ISPP

ISPP offers significant mass savings over the Earth supplied options when ascent stage comparisons are made at the "*Earth Departure*" mass level (Column 9 of Figure 9). Production of both oxidizer and fuel on the surface would allow the O_2/CH_4 stage to be transported to Mars with its propellant tanks empty, at one fourth the mass of its fully fueled O2/H2 Earth supplied counterpart. At 9.7 mt, this all ISPP concept proved to be the lightest of all those ascent stages evaluated. After fueling on the surface, however, the ISPP O2/CH4 craft becomes the heaviest of those evaluated, weighing 47.9 mt when fully fueled, quintupling its empty delivered mass (38 mt of this amount is propellant).

It is important to note that the ascent stage load bearing and descent stage load bearing frame and landing leg structure needed to support this amount is heavy, even in the reduced gravity environment on Mars. This additional descent stage structural mass, (along with the 9.6 mts additional ISPP hardware and H_2 seed propellant that must be carried to the surface), is enough to change the rankings of the weights comparison when the concepts are assessed at the "*total lander mass*" level (last column of Fig. 9). The descent stage required for the in-situ O2/CH4 ascent stage is as heavy as the descent stage required for the all Earth supplied O2/H2 ascent stage, and 10.3 mt heavier than that descent stage required for the ISPP O2/ Earth supplied H2 ascent stage, and 12 mt heavier than the descent stage required for the nuclear ascent stage (eng t/w = 9).

Consequently, the total lander mass (ascent, descent and surface cargo) associated with the nuclear ascent system (eng t/w=9) is only 4.1 mt more than that of the ISPP O_2/CH_4 system (91.0 vs 86.9 mt), though its ascent stage mass (at Earth departure) is 15.9 mt heavier. This total lander mass value is more relevant to this investigation than merely the ascent stage value, as it is the complete lander system that must be boosted to Mars. At 81.8 mt, the total lander vehicle associated with the ISPP O_2/Earth supplied H_2 ascent vehicle is the lightest overall. The NTP ascent stage is illustrated in Figure 11.

Minimum mass, however, cannot be treated as the only criterion for evaluation; operational differences must also be assessed. In this regard there exits a notable difference between the Earth supplied and the ISPP vehicles. The former do not require the additional complexity associated with autonomous vehicle predeployment and operation on the Martian surface previous to crew arrival. Also, the non-ISPP landers have the capability to effect an abort during the descent burn if necessary. In such an event the ascent stage can separate from the descent stage and burn its propellant to return to Mars orbit. The ISPP vehicles are delivered to Mars with their ascent tanks empty and inherently have not this capability. However, the ISPP strategy does offer a secondary benefit for scenarios in which extensive use is made of surface rovers. If an ISPP system is emplaced and operational, additional propellant beyond that required for the ascent craft can be used for powering rovers and other surface base vehicles.

NUCLEAR PROPULSION ASCENT STAGE

Recent work in the nuclear thermal propulsion field (Ref. 6, 7) indicates that both a high engine t/w and Isp performance level from a relatively small, 20 klbf thrust class engine is possible for the short burn time required of a Mars ascent maneuver. For a vehicle capable of ascending to a 250 km by 24 hour period orbit, the nuclear ascent stage (25.7 mt for an engine t/w of 9, 27.8 mt for an engine t/w of 6) provides a sizable reduction in mass as compared to a chemical O_2/H_2 vehicle of the type typically utilized in Mars mission studies. The all ISPP O_2/CH_4 ascent stage provides the lowest delivered mass of the concepts considered. However, other factors raise its descent stage mass to a level such that the total lander mass associated with the ISPP O_2/CH_4 ascent stage is only slightly lighter then the lander mass associated with the non-ISPP nuclear ascent systems. What is noteworthy here is that the nuclear stage is competitive with the ISPP concepts without requiring autonomous pre-deployment and fueling at Mars previous to crew arrival, and without sacrificing descent abort capability. This single tank, single propellant system requires no ignition or combustion, and without a throttling, gimbaling or engine restart requirement could be made highly reliable. The complication imposed by its radiation source can be countered by shielding, which involves the addition of no moving parts.

The nuclear engine is basically a heat exchanger and not a combustion device. Mixing of oxidizer and fuel in an injector is unnecessary, and neither ignition nor combustion take place within the reactor. Typically, propulsion system reliability is achieved though engine redundancy; contemporary man rated vehicle concepts are in most cases designed with an engine out margin. The relative operational simplicity of the nuclear engine allows for an exception in this case, however. Though not a nuclear stage, the Apollo Lunar Module ascent stage can be cited as an example of a single engine manned ascent stage; high confidence in this propulsion system was achieved through simplicity; no igniter was used (hypergolic, or self igniting fuels were used), and no engine throttling, gimballing or restart was required. These simplifications reduced the number of credible failure modes and justified the choice of a single engine. Likewise, a single engine nuclear system would require no igniters, nor would it have a throttling, gimballing or restart requirement. Unlike the Apollo system, however, the nuclear system would require a hydrogen turbopump, which represents an additional failure mode. The utilization of dual turbopumps, however, would allow continued operation in the event of a single pump failure. Rocketdyne designed, built and tested a dual turbopump system for the NERVA series of nuclear engines in 1967 (Ref. 8).

A single engine is utilized for the nuclear concept. Major subsystems of this craft are labeled from one to seven on the cut away view contained in Figure 11. These elements are the (1) ascent crew cabin, (2) the primary ascent stage H2 propellant tank, (3) the descent stage structure, (4) the descent stage propellant tanks (total of four), (5) the descent stage engines (total of four), (6) the ascent stage secondary (conformal) tank and (7) the ascent stage nuclear engine. (This and other lander configuration types are discussed in detail in Ref. 9.) Vehicle design analysis included consideration of the unique issues associated with the use of nuclear propulsion, including radiation shielding and engine disposal.

A NTP system emits about 1 percent of its energy in gamma and neutron radiation through the reactor pressure vessel (Ref. 10). This high radiation environment in the vicinity of the reactor may produce potentially lethal radiation doses in the vehicle crew compartment, may possibly damage sensitive components, and heats the surrounding structure and propellant. The addition of a radiation shield between the reactor and the tank bottom attenuates the energy disposition into the vehicle. The thickness of the shield for a given energy attenuation is a function of reactor power and also an inverse function of separation distance. The cumulative shielding capability provided by the dedicated external shield, the internal engine forward support plate, the propellant, and other spacecraft and crew cabin hardware mass serves to keep the crew from receiving radiation exposure beyond acceptable limits. Because liquid H2 has good neutron absorption capabilities, the design takes advantage of positioning the H2 propellant to supplement the direct line shielding ability of the radiation shield. The shield design must take into account the supplemental shielding ability of the other attenuating elements, and the overall geometry of the system including ducts, voids and other features.

Radiation shield effectiveness calculations are quite complicated, and require an extensive accounting of a variety of geometrical elements that can contribute to attenuation or exacerbate the problem through the generation of secondary sources of radiation (Ref. 11). A radiation assessment was not done for this study; the estimate of 1.0 mt of shield mass used in the analysis was scaled (based on power level) from a shielding study (Ref. 12, 13) done for a larger class of engines. Previous to the ascent burn, the engine contains no radioactive fission products. Over the course of the 13 minute burn time about 5 grams of fission products would have accumulated in the reactor; this is roughly 1/100000 of the amount generated by a typical 3000 MW terrestrial nuclear power plant in one year.

CONSOLIDATION OF CREW TRANSFER VEHICLES TRADE

Consideration was given to the compression of the reference fleet size from 3 transfer vehicles to 2. Rather than splitting the Earth-Mars-Earth transfer mission into two elements as in the DRM, with its separate Earth-Mars crew transfer vehicle and TEI transfer vehicle, a single transfer vehicle was configured to consolidate the function of the two into one. The MTV is round trip capable. This consolidation eliminates the need for autonomous placement of the TEI transfer stage in Mars parking orbit 4 years ahead of its eventual utilization. However this mode retains the DRM characteristic of pre-emplacing an empty *in-situ* lander, as well as utilizing the surface habitat as the crews outbound transfer habitat. This full round trip capable transfer stage benefits from the utilization of NTP propulsion for the MOC and TEI in addition to TMI. The DRM utilizes less efficient O2/CH4 propulsion for TEI (because it requires TEI stage aerocapture at Mars.). Figure 12 presents a vehicle set mass comparison between this 2 vehicle variation and the 3 vehicle DRM. All 2011-2014 mission objectives planned for the 3 vehicle DRM can be achieved by the 2 vehicle set for 39 mt less IMLEO. (Total mission IMLEO's are listed in the last column; 458 mt for the DRM, 420 mt for the consolidated set.) The single round trip capable MTV of this consolidated set would consist of 3 pieces joined together (with minimal assembly) in Earth orbit prior to TMI. The mass of these 3 pieces are shown as boxed values in Figure 12, bottom row.

These 3 elements compare to the 4 elements necessary for the reference DRM (two each to make up each of the two transfer vehicles). The weights for these DRM elements are also given in Figure 12 (boxed values in rows 1 and 3) and in Table IV below. The six ETO flights required for the DRM set could be reduced by one, to five ETO flights by this consolidation.

TABLE IV MISSION ELEMENTS

DRM type	Lander asc stg surf p/l	Lander surf hab	TMI stg	TEI stg
cargo	n/a	n/a	69.7	72.3
cargo	86.9	n/a	80.0	n/a
manned	n/a	69.0	80.3	n/a

DRM VARIATION type	Lander asc stg surf p/l	Lander surf hab surf p/l	TMI stg	MTV stg
cargo	86.9	n/a	80.0	n/a
manned	n/a	69.0	92.0	91.7

NON-ISPP. MARS MISSION ARCHITECTURES

This non-ISPP architecture is characterized by a mission set consisting of cargo and piloted vehicle pairs. The piloted craft is a round trip capable transfer vehicle carrying the lander to Mars orbit. The lander is fully fueled at Earth departure. The cargo vehicle delivers the surface habitat and infrastructure on a low delta velocity trajectory previous to crew arrival. The difference from the 2 vehicle consolidated set (variation) described earlier, is the lack of ISPP, with the pre-emplacement of the surface hab, rather than the ascent stage, previous to crew arrival. This mission mode has been called the "Split - Sprint" mode in past studies. The mission is split into the cargo and piloted components, with the piloted MTV utilizing a faster trip trajectory (sprint) as compared to the slower (lower delta velocity) cargo trajectory. A variety of configuration options exist which could be utilized for this architecture type.

CONFIGURATION OPTIONS

The available propulsion and aerocapture options for manned Mars transfer vehicles are presently graphically in Figure 13. The four configuration variations illustrated are described below.

Type 1:

This vehicle utilizes NTP exclusively. The same NTP engine cluster and core transfer stage is used for the TMI, MOC and TEI burns, while the lander is propulsively captured with the transfer stage. This option eliminates the need for aerocapture into Mars orbit. A low L/D, descent only aerobrake is shown on the lander of the *Type 1* vehicle in Figure 13.

Type 2:

The *Type 2* vehicle is an all NTP transfer vehicle with separate aerocapture of the lander. Same as *Type 1* above, with the exception that before Mars encounter, the lander separates from the MTV and either aerocaptures into Mars orbit or effects a direct entry to the Mars surface. The same NTP core stage is used for TMI, MOC and TEI. A biconic aerocapture brake is shown on the *Type 2* vehicle of Figure 13. Since the transfer stage payload to be decelerated at MOC is reduced by the mass of the lander, MOC propellant is reduced compared to the *Type1* vehicle. An increase in mission risk is incurred due to the incorporation of aerocapture technology into the program. (Mars orbit to surface descent only aerobrake technology entails much less risk). Figure 14 shows vehicle packaging in an ETO launch vehicle for a representative *Type 2* configuration.

Type 3:

A NTP core stage is used for TMI and MOC propulsive capture only. The lander is propulsively captured with the transfer vehicle into Mars orbit. A separate chemical stage is used for TEI. The *Type 3* vehicle illustration in Figure 13 shows a chemical propulsion stage integrated with a spherical crew habitat. The NTP core stage is left in Mars orbit. The TEI stage is expended at Earth arrival.

Type 4: The *Type 4* vehicle is a NTP / Chemical / Dual Aerocapture Vehicle. A single use NTP stage is used for TMI and then continues on past Mars without capture. A chemical O_2/H_2 or O_2/CH_4 propulsion system is small enough to be packaged together with the transfer habitat module into a biconic brake of the same dimension as that used by the lander. The chemical TEI stage and the lander separate upon arrival at Mars and each aerocaptures. Both are shown with biconic brakes in Figure 13. The TEI stage drops its aerobrake once in Mars orbit.

PERFORMANCE ASSESSMENT

Each of these four configuration types were assessed for the same 2014 mission trajectory utilized in the DRM analysis, carrying equivalent surface payloads, with the exception that the full round trip vehicle carries a heavier crew transfer hab than that of the DRM TEI stage (27 vs 21 mt). For these Type 1 - 4 vehicle missions ISPP was not utilized; the NTP (eng t/w=6) ascent stage was delivered fully fueled with the crew on board to Mars surface. A summary weight statement is presented in Figure 15 for each configuration. Three *Type 4* vehicles are listed, differentiated by TEI propellant type (O_2/H_2, O_2/CH_4 or O_2/H_2-Be). Fleet IMLEO varies from 229 to 292 mt. The lander utilizing the NTP ascent stage was used for IMLEO's shown in Figure 15.

The *Type 1* configuration required an IMLEO of 292 mt. Elimination of aerocapture by this configuration offers risk reduction, has a single propulsion technology (NTP) is used exclusively. Used with the NTP ascent stage, NTP would be the sole propulsion technology development required for the program (with the exception of the lander descent engines, which could be existing RL-10's).

The *Type 2* all NTP vehicle with independent lander aerocapture IMLEO is 250 mt, a reduction of 40 mts compared to the *Type 1* (rows 1 and 2 of Figure 15). This savings is in MOC and TMI propellant, and is one objective way of measuring the benefit of independent lander aerocapture. Consideration must be given to the cost of developing and testing a man-rated aerocapture brake into Mars orbit (rather than a brake suited merely for descent). This development cost must be compared to the extra launch cost associated with putting 40 mt of H_2 into LEO.

The *Type 3* NTP / Chemical TEI vehicle IMLEO is 247 mt. Like Type 1, this mode does not require aerocapture.

Three *Type 4* configuration vehicles were evaluated, differing only in TEI stage propellant type; O_2/H_2, O_2/CH_4, and O_2/H_2-Be. The masses for these three 2014 transfer vehicles are 236, 247 and 229 respectively. The *Type 4* craft utilizes aerocapture at Mars.

Payloads for all these transfer vehicles included a 27 mt transfer hab and the lander, which weighed either 52 mts if it had to independently aerocapture, or 47 mts if it was propulsively captured with the MTV.

Fleet comparison to the baseline DRM

A comparison between the DRM and this non-ISPP architecture using the *Type 2* and *Type 3* transfer vehicles is shown in Figure 16. For each, the same total surface payloads are delivered so as to make for a fair comparison. In Figure 16, note columns relating to total fleet IMLEO and ETO flights required. Total mission IMLEO is roughly the same for all three modes. However, the six ETO flights required for the DRM set would have to be increased by one, to seven for the other two modes. When overall program cost is considered however, the *Type 2* mission offers potential for significant benefit - elimination of autonomous, predeployed ISPP. The *Type 3* set achieves this benefit but provides a third: elimination of Mars aerocapture.

REUSABILITY

A comparative evaluation of the propellant weight penalties for recapture of the MTV at Earth for reuse was done. Evaluations were done for both recapture to a High Earth Elliptical Orbit (HEEO) (800 by 45000 km 14 hr orbit period) and recapture to LEO (407 km by 407 km). All recaptures were effected propulsively by reusing the "core" stage (MOC/TEI/EOC stage).

The *Type 2* configuration was used in this assessment. Mars vehicle IMLEO and major component weights are given in Figure 17. The 2014 mission only was evaluated, in this case with a Mars-Earth inbound leg characterized by a 5.2 km/s Earth arrival velocity (V infinity). The MTV, which carries a larger 30 mt crew transfer habitat in this instance and a 52 mt lander, has one NTP engine cluster, utilized sequentially for the TMI, MOC, TEI and EOC burns. The lander aerocaptures at Mars independentlyof the transfer stage, this is the *Type 2* characteristic.

(1) Expendable case for reference
(2) Recapture into HEEO
(3) Recapture into LEO

Recapture into LEO

Vehicle propulsive capture down to LEO for reuse is expensive in terms of the added propellant required. IMLEO is higher by about 75% for the 2014 mission. (442 vs 250 mt). 40 mt of Earth Orbit Capture (EOC) propellant is required.

Recapture into HEEO

For recapture into HEEO the penalty is much less severe; 13.5 mt of EOC propellant is required; IMLEO increases by 65 mt. A summary of major element weights is contained in Table V.

TABLE V REUSABILITY RESULTS
Mission Elements for 2014 *Type 2* round trip transfer vehicle. Departs LEO

Return to	Lander asc stg	TMI prop tankage	MTV stg	Delta mass
expendable	52.0	107.2	90.9	--
HEEO	52.0	134.7	129.2	+65
LEO	52.0	187.6	202.9	+192

Reuse of Habitat and NTP systems

It is desirable to reuse the expensive core propulsion stage and the transfer crew habitat. After recapture into HEEO, the vehicle is refueled, a new lander is attached, and the vehicle is utilized on the next piloted opportunity. In Figure 17, two potential 2016 reuse missions are listed below the heading "High Elliptical Earth Orbit," and the 2014 mission which returns to HEEO.

The first 2016 option is to depart from HEEO, conduct a new mission and again return the transfer stage to HEEO (for yet another potential reuse). The second option is similar but no recapture of the vehicle is attempted after this its second use. Element weights for these two options are shown as rows 3 and 4 on Figure 17. For the latter option (row 4) the major elements necessary for refueling and resupply to enable a second mission are listed in Table VI. 127.7 mt of mass would have to be boosted to HEEO for attaching the TMI fuel tank, transferring MOC/TEI propellant, and attaching a new lander payload to the vehicle. Compare this to 250 mt required for a totally new 2016 vehicle. Not only is 122 mt saved, the procurement of one habitat, core stage, and NTP engine cluster is now unnecessary. The life expectancy of the transfer crew habitat is greater than the duration of a single mission, and the NTP system would also contain sufficient operational life left for at least one additional mission, perhaps two.

TABLE VI REUSE REQUIREMENTS
Reuse of the 2014 *Type 2* round trip vehicle in 2016. Departs from HEEO

(1) TMI propellant and tankage:	37.1 mt
(2) MOC/TEI/boiloff propellant:	38.6 mt
(3) Lander system:	52.0 mt
	127.7 mt

NTP engine life expectancy (on the order of 10 hours of burn time) exceeds the cumulative burn time of the first mission, which is about 3-4 hrs. The cost savings associated with reuse of these elements must be evaluated against the expense of the upper stage needed to ferry the lander and reuse propellant to HEEO. If there already exists a large upper stage (for LEO-GEO, or LEO-Moon) that could be utilized for this purpose then the expense for development could be avoided. Such was not defined for this study. It is recommended that a HEEO mission OTV system be defined, and to identify possible synergism with other mission types.

ABORTS COMPARISONS

We return to the subject of abort modes mentioned earlier. The Mars DRM represents one of several possible mission modes for manned exploration of Mars. This unique planetary transfer architecture offers some interesting benefits to Mars mission planners but also presents some difficulties. Elements of the DRM abort strategy must be considered if a complete evaluation is to be made. Difficulties are quantified by comparison with the abort strategies of the more traditional mission profiles. (See Figure 18) Rationales put forward by proponents of the DRM are identified, as are other issues associated with the "abort-to-surface" philosophy that is advanced when discussing this unique mission mode. The importance that comprehensive abort planning has on manned interplanetary mission design must not be overlooked. Gaining an understanding of the DRM's abort strategy and how it affects the probability of mission and crew loss was accomplished via comparison to several alternative mission modes, including the variation to the DRM described earlier.

To facilitate this comparison, references will be made to a series of graphical flowcharts called "Mission Abort Flow Diagrams." (These are given in Figure 19-22, one each for five different Mars architectures) Abort related difficulties, in which mission events have no recovery capability should an anomaly occur, are represented on the Mission Abort Flow Diagrams as shaded circles appearing under specific mission events (enclosed by unshaded boxes). These may be thought of as representing "holes" in the abort strategy. Reference to these "no-abort available" events will help facilitate a quantitative, objective approach to estimating a mission failure probability. Some descriptive repetition will occur as the issue is addressed in the following section.

DRM ABORT PHILOSOPHY

The Mars DRM is a special mission mode that emphasizes the pre-emplacement of major mission hardware components at Mars previous to the departure of the crew. In this respect it is not different from other mission modes investigated in past studies. What differentiates it from these other modes is what is pre-emplaced. The DRM pre-emplaces an empty *In-situ* lander on the Martian surface and an unmanned crew return stage in Mars parking orbit. (For the traditional modes the surface habitat system is pre-emplaced). The crew then departs in an one-way only (Earth-Mars) capable transfer vehicle, living out of the surface habitat in route.

This piloted transfer vehicle contains no return propellant. Upon arrival into Mars orbit, the crew descends to the surface in the surface habitat living out of it during the period of the surface stay, typically about 500-600 days. At the end of the surface mission, the crew then traverses across the Mars surface some distance to the ascent stage, boards and readies the craft, then ascends to Mars orbit for rendezvous with the TEI stage, which has been waiting in Mars orbit for four years. (This pre-emplaced TEI stage contains crew consumables sufficient for the return leg only.)

Along with its resource placement strategy, the elimination of on-orbit assembly is a key characteristic of this architecture. This is achieved by splitting the mission into three transfer vehicle components as outlined earlier, two of which serve as crew transfer vehicles, one outbound, one inbound. This necessitates two distinct crew transfer habitats, propulsion stages, power systems, aerobrakes, etc. for what is intuitively a single manned Mars mission. By separating the crew traveling on the outbound Earth-Mars leg from the Earth return stage, the option of effecting a Mars swingby abort (rather than Mars capture) for an immediate Earth return is lost (in case of a return stage, or surface system malfunction). Also lost is the crew's option of effecting a descent maneuver abort, since the DRM's ascent stage is not carried on the descent stage. The crew descends with their outbound transfer habitat to the surface, having only terminal descent propellant onboard.

Resource Location

Typically, for full-round trip capable vehicles flying conjunction trajectories, some additional propellant is carried onboard for affecting (if the need arises) either an immediate Earth return via Mars swingby, or an earlier than planned Earth return maneuver. In either case, more propellant would be required than the nominal non-abort conjunction mission return to Earth would require, hence it can be said that resources (in this case TEI propellant) are directed to transfer stage capability to provide an opposition mission like abort return to Earth capability.

The DRM philosophy, in contrast, directs all resources to Mars. In cases of an outbound system's difficulty, these assets, (pre-emplaced on the surface), are to be used as a haven for the crew until such time as a later rescue mission can reach them. The utilization of these assets in this way, is explained as differing in no significant way from their original, intended use as outpost assets (that is, habitation and exploration activities). Thus no additional assets beyond that originally slated for the nominal mission would be required to cover an abort situation, as they would in the previously mentioned case where some extra propellant is necessary to provide for a swingby abort or an early Earth return TEI burn. (This is the characteristic of the Earth return abort mode philosophy) It should be noted that, for the Earth return philosophy, the added resource is only propellant; which entails no change in the vehicle operation, nor introduces any new technology into the mission, and requires very little change in the vehicle design (except for stretching tanks to accommodate additional propellant).

Abort Philosophies

Several abort philosophies will be referred to in the following, including "Abort-to-Surface," "Abort-to-Orbit," Flyby, and "Return-Next-Chance." Conjunction class missions are generally planned with *Abort-To-Surface* and *Abort-to-Orbit* modes. *Return-Next-Chance* with *Abort-to-Orbit* is the typical abort planning scheme for an opposition class mission. A strict *Abort-to-Surface* is a mode of abort that was adopted by the Mars DRM. *Abort-to-Surface* has the characteristics of devoting as few resources of the mission as possible to up-front contingencies for abort. If some event occurs that normally requires abort, the strict *Abort-to-Surface* mission philosophy dictates that the crew go to surface and if possible return to Earth at the end of the mission or continue on the surface until a rescue can be mounted. This rescue could use the next nominal mission reconfigured to perform the necessary rescue function. If a catastrophic event occurs that precludes the crew from remaining on surface to End Of Mission (EOM), the mission and crew will be lost. The *Abort-to-Orbit* mission mode is usually in combination with the other abort modes. *Abort-to-Orbit* simply allows for sufficient consumables on-board the MTV or the ETV to allow the crew to weather certain abort events. Thus, if some event dictates that the crew cannot go to the surface after a nominal arrival, they can remain on-orbit until an EOM return. Another example that would call for an *Abort-to-Orbit* could be an early return to orbit, requiring an EOM return or a rescue mission.

Full round trip capable missions generally employ the *Return-Next-Chance* abort strategy. With full round trip capable mission contingencies, the crew can stay in orbit, on the surface, or in some cases immediately return, depending on the time and characteristics of the abort event. The *Return-Next-Chance* strategy can increase the number of possible ways to recoup from an abort event, and therefore the *Return-Next-Chance* strategy may be more flexible.

An event leading to an abort can result in several outcomes. First, corrections could be made with no mission or crew loss. For example, the crew could *Abort-to-Surface*, the mission could be completed, and a successful rescue is undertaken. The second case to consider consists of a loss of mission, but the crew is returned safely to Earth. An example of this kind of abort could entail the following scenario: the habitat on Mars is remotely detected as having irreversibly malfunctioned, the crew conducts a Mars flyby, and the subsequent return to Earth is successfully completed. This abort scenario entails a mission loss but the crew is safely returned to Earth. A third case that is considered could be a crew loss (e.g. ascent vehicle misses rendezvous with Earth return vehicle).

ABORT FLOW FOR TYPICAL MISSIONS

In this section, abort flows for five architectures are discussed and compared.

1. Round trip capable (non-*in-situ*) modes
2. Mars direct mode
3. Mars DRM baseline
4. Consolidated DRM variation (*in-situ*) mode
5. Generic electric mission

The discussion will address only the manned vehicle portion of the mission. Each assumes some pre-emplacement on the surface prior to crew arrival. The DRM requires two pre-emplacement missions, the others require one. Figure 18 lists what is pre-emplaced for each architecture. The round trip capable and generic electric modes can employ either conjunction or opposition style transfers. The Mars Direct, DRM and consolidated DRM variation are conjunction only modes.

Opposition trajectories

Opposition class missions are characterized by Mars stay times of 30 to 90 days. Opposition missions are possible every two years and therefore, rescue missions can be mounted on two year intervals. The standard abort strategy chosen for these architectures is the *Return-Next-Chance* mode. If an abort event that is not related to main vehicle propulsion occurs before the nominal Mars departure point (within 30 to 90 days of Mars arrival), the opposition class vehicle has adequate fuel to depart early. If the abort event precludes an early departure, the crew can go to the surface or remain in orbit until the following departure opportunity or wait until a rescue mission arrives.

Conjunction trajectories

Conjunction class missions are characterized by stay times of 500 to 600 days. The missions are approximately 2 years apart with long stay times between return opportunities. One of the disadvantages of the conjunction style mission is related to rescue opportunities. The return opportunity falls several months before the next mission arrival from Earth. This return constraint is related to the physics of interplanetary transfer. Thus, conjunction arrival / return opportunity constraints aggravate the abort scenario by requiring additional living space and consumables for the rescue crew over the duration of another opportunity (approximately 2 years, including transfer time). Alternatively additional TEI propellant can be carried on board to furnish the extra delta velocity necessary to depart for Earth immediately and return on an off-nominal trajectory. The propellant penalty, which can be excessive, is dependent on the time Mars departure takes place.

The round-trip capable mission modes

This mission type can be broken into eight primary events: TMI, Early Trans-Mars Coast, Late Trans-Mars Coast, MOC, Prepare for Descent, Descent, Surface Mission, Ascent, and TEI. There is nothing special about this delineation of the mission events, but this set of events was chosen for convenience to illustrate certain abort modes. See Figure 19 for the Abort Flow

Diagram for this mission mode. Under each of the primary events shown in Figure 19 are one or two typical abort events. For example, under *the Early Trans-Mars Coast* event box is a circle that indicates that an anomalous event has occurred, namely the transfer habitat malfunctioned, precluding long term use. The abort mode for this event dictates immediate return to Earth. Note that this event could not be a propulsion problem because there would be no immediate return without propulsion. Each circle under a primary event is either shaded gray or is not shaded. A non shaded circle indicates that there exists a way to abort the anomalous event, and the shaded circles indicate that no way to abort has been made possible. For all subsequent abort flow charts, the shading convention will hold true. This discussion does not purport to exhaust the possible abort related events. Further, the anomalous events designated as "No Abort" cases were assumed not to have an abort because of the prohibitively high cost in delta-V required to correct the trajectory, target an Earth return trajectory, or the abort event has no known way of escape.

For this round trip capable transfer vehicle mission, there are three primary events that have no abort scenarios. First, if the MOC maneuver fails to occur correctly, the vehicle could fail to capture and the mission and crew would be lost. Second, if ascent failure occurs after lift off the crew could crash or miss rendezvous with the return vessel, again resulting in mission and crew loss. Third, if the TEI fails in such a way that the vehicle is placed on an interplanetary trajectory that does not intercept Earth, the mission and crew will be lost. These are identified in Figure 19 as the shaded circles. Since this mode can fly both conjunction and opposition trajectories, some mission events show two abort circles associated with them, one corresponding to the conjunction, and one for opposition missions.

Mars Direct mode

Both the Mars Direct and DRM mission architectures employ *Abort-to-Surface* in their mission design. Mars Direct and the Mars Reference Design Mission Baseline have an abort mode of strictly *Abort-to-Surface*.
The Mars Direct mode places the Earth return crew habitat on the ascent stage, rather than reemplacing a separate TEI stage in Mars orbit. The ascent stage/crew habitat ascends to a direct to Earth departure from the Martian surface. Only 2 vehicles are used in the Mars Direct mission mode. The manned outbound stage is identical to that of the DRM, as it goes direct from Earth to the surface of Mars. See Figure 20 for the Abort Flow diagram for the Mars Direct mission; the primary events are basically identical to the corresponding events of the round trip capable mission shown in Figure 19, but the secondary abort events are different. Note the increase in anomalous events in which no route to recoup by abort are available (more than twice as many.) An explanation of this reduction in number of events that have aborts is found in the abort philosophy of the reference mission: *Abort-to-Surface*. This mission places almost no abort contingencies in the manned phase of the mission, other than on the surface. Thus for the first three primary events, *TEI, Early Trans-Mars Coast, and Late Trans-Mars Coast*, there is no CRV on the outbound vehicle, nor MOC or TEI propellant, precluding an Earth return. For *MOC* and *Prepare for Descent*, there are not adequate consumables on board the outbound vehicle for a stay in orbit until a rescue could be mounted on the next opportunity. The surface phase of the mission assumes *Abort-to-Surface*. This abort philosophy is inherently an effective abort approach once the crew is on the surface. (It should be pointed out, however, that there are not adequate consumables on the Earth return vehicle in the event that the crew is forced to go to orbit early to await rescue or EOM return.) Given the strict split strategy of the Mars Direct mission, there is no ascent vehicle on the manned outbound mission to Mars, resulting in a "No Abort" for the descent phase of the mission.

Mars DRM

The present DRM is a modification of the Mars Direct Mode. See Figure 21 for the Abort Flow diagram for this mission. The inbound vehicle leg now consists of the *in-situ* ascent stage flight to Mars parking orbit and rendezvous with the waiting inbound habitat / TEI stage, which departs for Earth. This allows for the option of the outbound crew vehicle to capture at Mars and rendezvous with the inbound stage without descending to the surface. However, the piloted outbound stage has no main propulsion system to effect orbit raising and plane change maneuvers (its TMI stage was jettisoned immediately after Earth departure). Also, both it and the TEI stage would have to jettison their aeroshells successfully before the two could rendezvous. In the DRM flow the primary events are also basically identical to the corresponding events of the round trip capable mission, but again the secondary abort events are different, like those of the Direct mode. Notice again the anomalous events in which no route by abort is possible (Figure 21). Though they are fewer than the Mars Direct mode, they are more numerous than the round trip capable mission mode. Because the outbound crew has no main propulsion available it is impossible for the vehicle to conduct a Mars swingby, if a surface system, ascent stage or TEI problem is detected on the outbound journey. As mentioned earlier, rendezvous with the TEI stage for Earth return may be possible. Again, there is no ascent vehicle on the manned outbound vehicle, hence a "No Abort" for the descent phase of the mission.

The consolidated variation to the DRM

See Figure 22 for the Abort Flow Diagram. The abort situation for the full round trip transfer vehicle variation DRM represents an improvement over the baseline DRM. Propulsive capture of the habitat and core vehicle means that MOC and TEI are on board the vehicle and can be utilized to conduct a Mars swingby return to Earth in the event of surface systems or lander systems malfunction. In this respect it is similar to the traditional round trip mode discussed earlier. Since it retains the *In-situ* lander component however, it, like the DRM, can offer no recovery from a descent burn malfunction.

The generic electric mission

The generic electric mission falls into the category of conjunction or opposition type mission because of the flexible characteristics of electric propulsion. An electric mission can be either nuclear powered or solar powered. The abort approach employed in nominal electric mission is *Abort-to-Surface* or return on reduced power. A reduced power return will entail a longer return trip, however, the windows of opportunity are significantly wider than a conventionally powered mission. If an *Abort-to-Surface* is required, then the electric mission will incur the same consequences as described earlier. There are fewer primary events for an electric mission than a conventional mission.

ABORT COMPLICATIONS OF THE DRM

Abort complications arise for the DRM due to the dividing of what would intuitively appear to be a single manned transfer vehicle into two transfer spacecraft (one of which must operate autonomously for four years pervious to its utilization). If the abort anomaly that occurs during the outbound Earth-Mars transfer is related to this quiescent Earth return stage, then there are no assets that could cover such an abort. That is, unless a second TEI stage is also pre-emplaced. This is a consideration of the present DRM, to pre-emplace a *second* TEI in orbit as a backup return vehicle should the first TEI stage fail. (This second TEI stage, would serve in the nominal case as the return vehicle for the crew flying the next opportunity in 2016.) At this point one must consider the total number of habitat transfer vehicles allocated to insure the crew's safe return; at this point we have noted that DRM planning calls for 2 transfer vehicles to be autonomously emplaced in Mars orbit, previous to crew arrival; this seems excessive and runs counter to the objective of designing for low cost missions.

In cases of an indicated anomaly in the descent stage occurring during its outbound journey, again little recourse is available to the crew for a return to Earth since the in orbit TEI stage is only outfitted for, and capable of, a prescribed conjunction return at a point in time at least 500+ days into the future (the duration of the planned surface stay). It therefore has not the propellant to affect an early Earth return maneuver, nor are the 500+ days of consumables required for the crew, available in the TEI stage for them to wait in orbit until the correct departure date. Compare the number and type of "no-abort available" events (shaded circles) of the DRM (Figure 21) to that of Figure 19. It is evident from an examination of the Abort Flow Diagrams, that the DRM does not adequately provide for all credible failure paths, and thus levies serious additional risks to the mission. The strict philosophy of "*aborting to the surface*" does not adequately provide a viable abort path for the full complement of mission events that the crew will be subjected to. The multiple difficulties associated with any actual implementation of the DRM will translate into a greater probability of both mission and crew loss than other mission modes. (Reconfiguring the DRM to the "consolidated" DRM variation mode helps somewhat, compare Figure 22 to Figure 21.)

Proponents of the DRM strategy, however, have argued that total mission risk is actually decreased. The following paragraph is quoted from the document "*Mars Exploration Strategies: A Reference Program and Comparison of Alternative Architectures*" (Ref. 14) published in 1993.

> " The provision of a robust surface capability is fundamental to the reference mission philosophy employed in this study. Assets are focused at the planetary surface because that is where the goals of Mars exploration can be achieved. Although efficient and reliable space transportation elements are critical component of any planetary exploration strategy, the exploration goal adopted in this study suggests the need to be able to "live off the land."
>
> Thus, the surface capability must provide a comfortable, productive, reliable, and safe place for the crew. This, in turn, changes the risk perspective with respect to previous studies by relieving the pressure on the space transportation systems to resolve any and all contingencies. Whereas in previous studies, many mission contingencies resulted in trajectory aborts (direct returns Earth) another option exits in this reference mission, namely, abort to Mars' surface. This allows the mission design to focus on the surface capability, not on the provision of costly propulsive performance increases and redundant systems to be used in the unplanned and relatively improbable even to system failure in flight.
>
> Unlike in Apollo and other strategies for returning humans to the Moon, free-return abort and power abort maneuvers do not come for free at Mars. The goal of the human portion of the space transportation function should be to deliver the crew to and from Mars with the least reasonable achievable exposure to the hazards of the space environment. Trajectory aborts, far from being presumed requirements for human missions to Mars, should have to fight their way into a reference mission as a last resort. By emphasizing the capabilities available to the crew on the surface of Mars, it, not the interplanetary space environment, becomes the most secure, reachable place for the crew in the Solar System after the completion of the TMI burn."

By the term "trajectory aborts" the authors are referring to crew return to Earth via a Mars swingby or powered Mars maneuver. The above document speaks subjectively of the benefit of the abort-to-surface philosophy, but fails to make an objective accounting of all the specific abort related events involved. The document does speak specifically of reducing the exposure to the hazards of the space environment, which in this context refers to the detrimental effects of exposure to zero gravity and solar and galactic radiation (which are lessened when the crew is on the Mars surface.)

But when one considers the importance that comprehensive abort planning must have on mission design, it appears that this topic as been somewhat overlooked in past evaluations of the DRM. All elements of its abort strategy must be considered if a satisfactory evaluation is to be made. Only after an rigorous accounting of the risks involved can one make a case for a realistic mission failure probability.

In this study, the approach has been to consider all mission events, determine if there is a recovery path available should a vehicle or surface systems malfunction occur during that event, and then compare the number and severity of these events to similar evaluations of other mission modes. Though more analysis is needed, this comparative evaluation indicates the following order of failure probabilities from highest to lowest:

TABLE VII MISSION FAILURE PROBABILITY
Ranking from highest to lowest

1.	Mars Direct
2.	Present DRM
3.	Round Trip Capable
4.	Generic

VEHICLE ASSEMBLY IN ORBIT

Mars vehicle configuration issues often revolve around inefficiencies associated with launch vehicle packaging and on-orbit assembly. Difficulties in these areas were addressed in the self-assembling Mars transfer vehicle configurations utilized in this study. This configuration family was conceived from the beginning to reduce on-orbit assembly tasks to a minimum. The design is characterized by its modular in-line integrated truss/tank elements and its self-assembly capability. There is no requirement for any tank-to-propellant line or tank-to-truss connection assembly operations on-orbit. Vehicle tanks are pre-integrated with propellant lines, tank gas pressurant lines and other hardware into a standardized tank/truss module as a single pre-assembled unit for packaging into the launch vehicle and berthing on-orbit. On-orbit assembly dedicated hardware and tasks are reduced over previous designs because the design was conceived from the outset to act as its own assembly platform. A 'core' transfer vehicle would consist of two integrated element modules. The first element, a transfer habitat module and Mars excursion vehicle, is integrated onto a forward spine rigid truss structure section. The second element consists of a engine / thrust structure system, radiation shield, cylindrical aft propellant tank, and a reaction control system (RCS) assembly, all integrated onto the aft truss structure section. These two element core vehicle modules could be identical for all missions, regardless of opportunity year. Subsequent ETO flights would deliver the necessary TMI and MOI propellant tank / structural modules; each would consist of a large propellant tank pre-integrated with propellant lines onto identical truss sections. Variations in mission delta velocity or payload requirement would then only impact the length of the non-core TMI and MOI tanks. All tanks would have the maximum diameter that the ETO vehicles shroud envelope will allow. A lunar transfer version would utilize the same two element core (which has the single aft tank), though with smaller crew habitat, lander system and a single, TLI tank. For each modular element a truss runs atop the propellant tanks to which they are attached; the non-core tanks can be jettisoned, by truss mounted release mechanisms after use to reduce mass for subsequent maneuvers on missions where this would be beneficial. Once on-orbit, the modular elements would be attached end-to-end. At the front and aft ends of each element are interfaces where propellant, gas pressurant, communication and power lines would be joined. All subsystems, propellant lines etc. are pre-integrated and located on the structural truss to which the tank is attached. Positioned above the truss are one or more RMS arm units that can traverse the length of the truss for the purpose of joining sections together. RMS units traveling across the truss top have access to these subsystems for joining, inspection, repair or change out; they could accommodate suited personnel to facilitate these operations if required.

Assembly consists of attaching the common tank/truss elements at their end-to-end interconnect points as shown in Figure 23-24. Rather than sending up to orbit a separate platform prior to the delivery of the spacecraft components, a single or dual RMS operates from the first element delivered to orbit. The forward habitat / truss element segment acts as the 'assembly platform' for the remaining elements. Utilizing this element's RCS system, translational and attitude control maneuvering to within RMS arm capture distance of the second co-orbiting element is accomplished. Moving along the top rails of the rigid truss section, the autonomous (or crew assisted) RMS captures and pulls to an aligned position the second element and connects the two at the end-to-end interconnect point. (This could also be accomplished by a small one or two man crew pod craft - this vehicle will be discussed in detail in the next section).

This first interface consists only of a quick-connect communication/power lines interface. Once connected, secured and inspected, the RMS then moves onto the second element, travels the length of its truss rail, reaches the unconnected end and repeats the capture and connection process for the next element (again with the first element's RCS providing the maneuver control to get within reach). For the second-to-third element connection, the quick-connect interface includes propellant and gas pressurant line link ups. This process is repeated until the vehicle is complete. The elimination of the requirement for a dedicated co-orbiting assembly platform, would offer a significant cost benefit to any Mars program. The number of connection operations would always be one less than the total number of elements delivered to orbit; i.e. for the four element vehicle pictured in Figure 23, three capture and connection operations would be required. For a three element Lunar vehicle, two would be required. A reduction in connection operations could only be achieved by increasing the lift capacity of the launch vehicle so that the vehicle would consist of fewer (but larger) modules. Furthermore, if vehicle reuse is desired, refurbished core vehicle elements could be joined to new non-core tankage modules, and a new landing craft on-orbit, allowing for the economical reuse of the expensive habitat and propulsion elements.

CONCLUDING REMARKS

The Mars DRM, in its present form, introduces additional risk due to difficulties associated with its abort strategy. In addition to its abort related challenges, the DRM introduces many new technologies at one time. Several options are available to address these challenges. Adopting the "consolidated" DRM mission variation would alleviate a portion of this risk while still retaining the DRM's ISPP component (if that is desired). The sum of the IMLEO's of the consolidated mission set is 8% less than the IMLEO sum of the DRM. The six ETO flights required for the DRM could be reduced by one to five of about the same capability. The single round trip capable MTV could consist of three elements ranging from 69 to 92 mt each, mated on orbit (or with minimal assembly). This compares to four elements (ranging from 69 to 87 mt) for the unaltered DRM. The cargo vehicle, with ISPP ascent stage, would be the same for both the DRM and the consolidated set requiring 70 and 72 mt pieces for launch.

Another approach for reducing risk would be to eliminate ISPP altogether. Mission IMLEO for an equivalent 2011-2014 mission could be kept at the same level by this approach if a suitable NTP ascent stage is utilized. Four year early, autonomous pre-emplacement would be unnecessary, and the crew transfer vehicle would be carrying a functional ascent stage allowing for a contingency descent abort. ISPP vehicles cannot accommodate descent abort. The total lander stage fitted with a non-ISPP, NTP ascent stage weighs about the same (within 5%) as the lander fitted with the empty O2 / CH4 ISPP ascent stage. It would require no autonomous predeployment, autonomous propellant production, nor the dedicated surface power needs of ISPP.

The present DRM is dependent on Mars aerocapture, which represents an additional risk as compared to propulsive capture. It also necessitates a significant technology development program. This expense can be avoided altogether by planning to reuse the NTP TMI engines again for MOC. Results showed that the mission objectives of the year 2011-2014 DRM can be done exclusively with NTP (excepting RL-10's for descent), without aerocapture, without ISPP, and without 4 year autonomous pre-deployment, at about the same IMLEO as the present DRM. (*Type 3*). In light of these factors, this study suggests that the DRM (in its present form) adds additional risk without providing significant benefit when compared to either the special risk-mitigating variation of the DRM, or to an appropriately designed non-ISPP alternative mission mode. It is a recommendation of this study that the risks associated with the present DRM be further quantified. It is also recommended that more technical detail be generated for the alternate modes and vehicles which appear to offer equal benefit at lower cost and risk.

SPECIFIC COST REDUCTION TECHNIQUES

(1) Reduce the number of new technologies to the DRM

(2) Eliminate the separate 4 year early predeployment of the DRM TEI stage by consolidating its function with the 2014 crew vehicle. The DRM ISPP philosophy (if that is desired) would still be retained.

(3) Utilize NTP for more than the TMI burn, specifically propulsive capture at Mars.

(4) Eliminate ISPP dependency.

(5) Utilize the self-assembly technique for Mars transfer vehicles, rather than rely on a DRM that is predicated on elimination of on-orbit assembly.

(6) Utilize the *Type 2* or *Type 3* configurations as a means of implementing measures (1) - (5) above. This would entail abandoning the DRM characteristic.

(7) Recapture of a round trip capable MTV into a HEEO may offer some potential for cost savings, if a suitable, preexisting OTV capability is available. A cost reduction via reuse of the NTP propulsion system/core vehicle and the transfer habitat may be achieved. The IMLEO penalty is modest, but the cost of the OTV system needed to boost propellant and payloads to HEEO must be considered. This warrants further investigation.

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LAUNCH SEQUENCE DIAGRAM

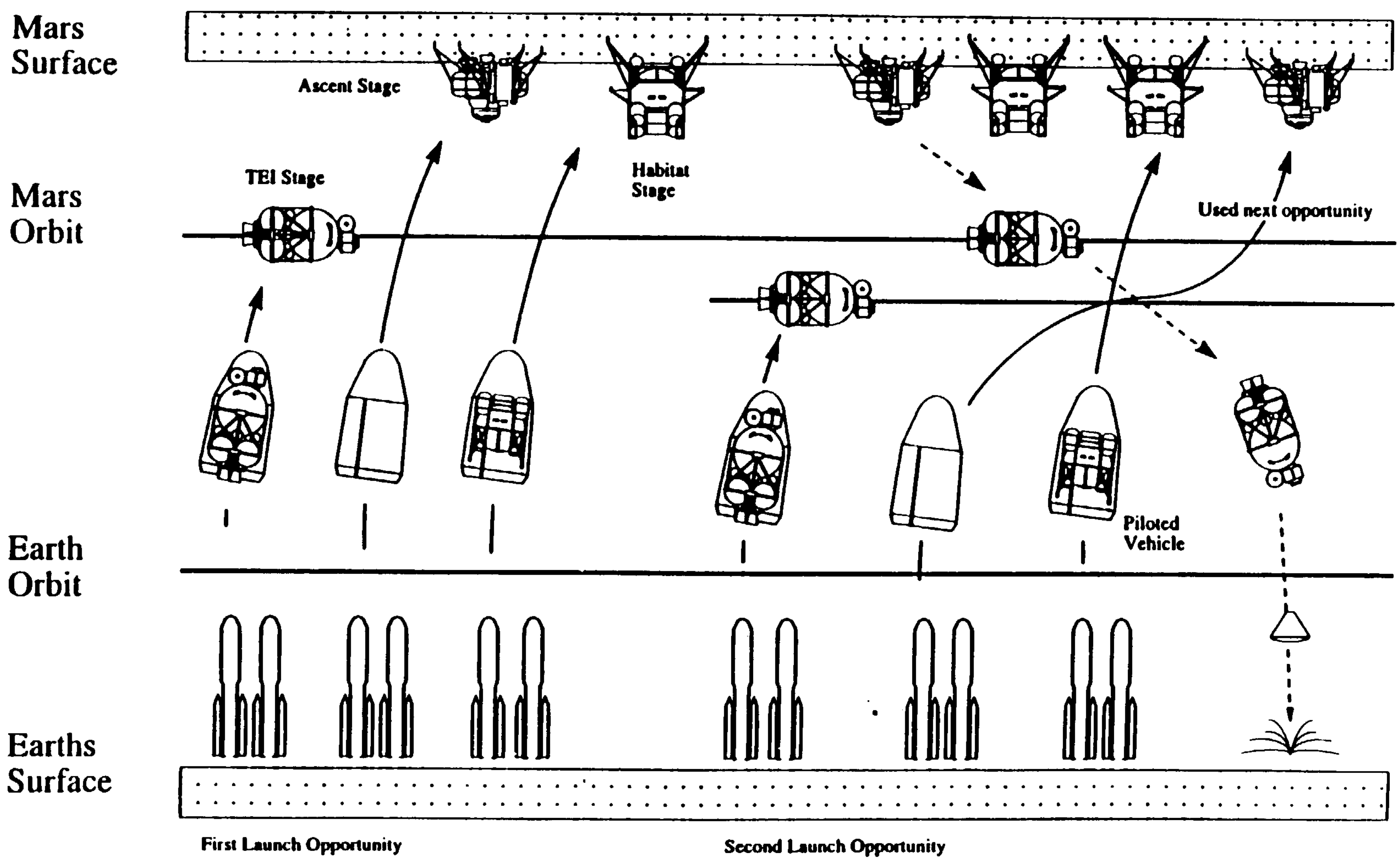


Fig. 1

Mars Design Reference Mission Timeline

2011 - 2014 Opportunity

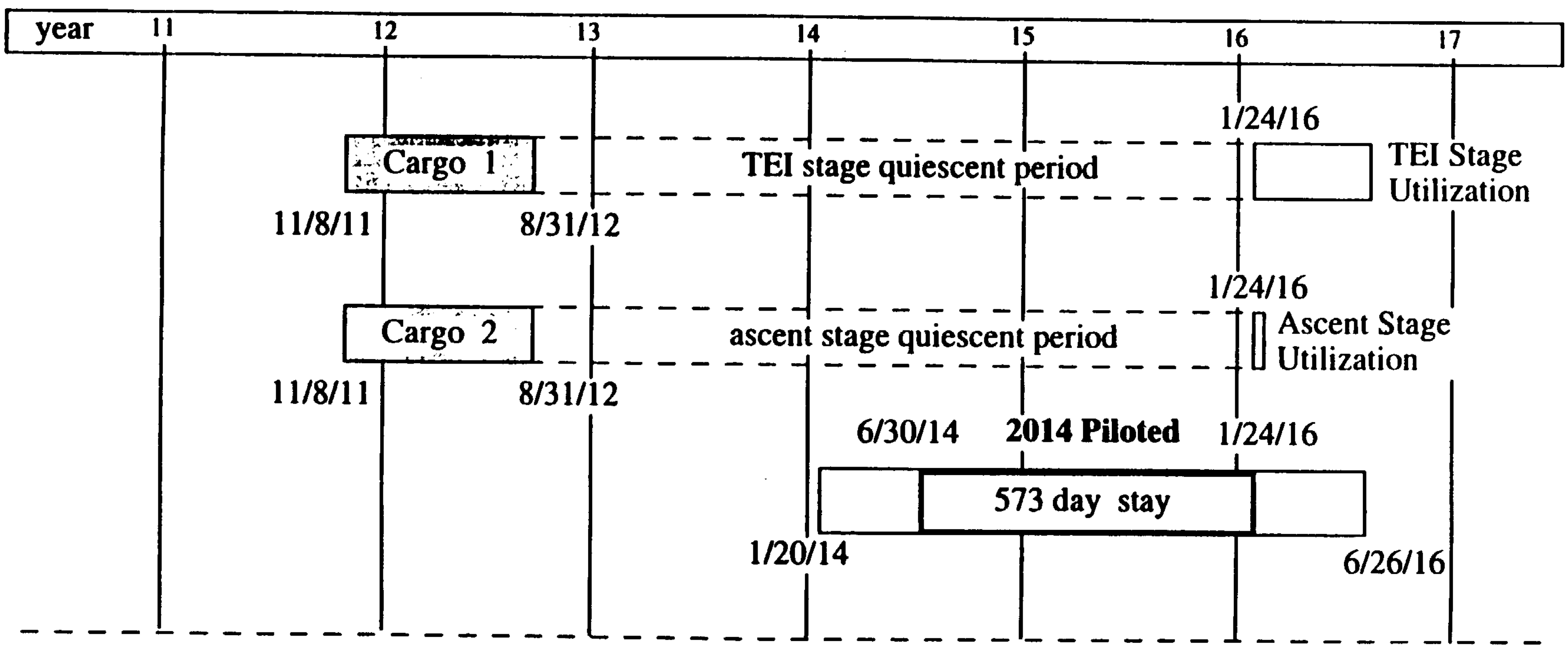


Fig 2

Human Mars Mission (HMM) Study
Ground Rules for DRM

- **Split Mission Scenario (2 cargo missions in 2011, 1 piloted mission in 2014)**
- **Earth Orbital Vehicle Assembly**

  Each transfer vehicle is delivered by 2 Earth-to-orbit launches (req. 1 AR&C)
  Launch vehicle lift capability to LEO ($\approx 400^{+}$ km): 80 mt
  Launch vehicle shroud size: 8.6 m x ~28 m

- **Aerocapture At Mars Arrival into Mars Parking Orbit**
- **Mars Parking Orbit: 250 x 33,849 km Altitude (1 Sol Period), ≈40° Inclination**
- **Aerobrake / Terminal Propulsion Used for Descent to Mars**
- **Automated, Precision (1 x 3 km) Mars Landings, w/ Landing Site Latitude ≈30°**
- **Baselines In-Situ Resource Utilization (ISRU) for Ascent Propellant**
- **Mars Ascent Capsule also Serves as Earth Crew Return Vehicle (ECRV)**
- **ECRV Performs a Hyperbolic Reentry at Earth Arrival**
- **Mission Abort Strategy:**

Outbound:	Abort to Mars surface
At Mars:	Backup Return hab and backup propellant production plant
Inbound:	Piloted transfer stage can perform TEI (w/ AFIS & TEI stage propellant)

Larry Kos
MSFC/PD32
9/16/97

Fig 3

Human Mars Mission (HMM) Study

Ground Rules for the Mars Transportation System

- **Trans-Mars Injection (TMI) Stage**

  Nuclear Thermal Propulsion (LH_2), I_{sp}: 960 sec (931 sec)
  Thrust: 45,000 lb_f (three 15 klb_f thrust nuclear engines)
  T/W ratio: 3.1
  Boil-off rate: 1.64% per month

- **Mars Descent Stage (Lander)**

  LOx / Methane propellant (LOx/CH_4), I_{sp}: 379 sec, MR: 3.5
  Thrust: 60,000 lb_f (four 15 klb_f thrust engines)
  Boil-off rate: LOx: 0.26%/mo; CH_4: 0.37%/mo

- **Piloted Descent/Ascent Stage**

  Utilizes Mars in-situ produced propellant
  LOx / Methane propellant (LOx/CH_4), I_{sp}: 379 sec, MR: 3.5
  Thrust: 30,000 lb_f (two 15 klb_f thrust engines)
  Common ascent crew capsule / ECRV
  Boil-off rate: LOx: 0.27%/mo; CH_4: 0.31%/mo

- **Trans-Earth Injection (TEI) Stage**

  LOx / Methane propellant (LOx/CH_4), I_{sp}: 379 sec, MR: 3.5
  Thrust: 30,000 lb_f (two 15 klb_f thrust engines)
  Boil-off rate: LOx: 0.14%/mo; CH_4: 0.16%/mo

- **Aerobrake (Integral w/ Habs) Sizing**

Cargo orbit capture only or descent only:	16% of entry mass
Cargo capture and descent:	21% of entry mass
Piloted capture and descent:	22% of entry mass

Fig 4

'97 DESIGN REFERENCE MISSION (DRM) ΔVs

HMM Split Scenario: 2 Cargo Flights & 1 Piloted Flight

	Aerobrake (A/B) Trajectory	A/B + Descent Trajectory	
Cargo TMI, 297 day trip (November 8, 2011)	3,581	3,581	m/sec
1% ΔV Margin	36	36	"
Gravity Losses	92	113	"
Midcourse	50	50	"
Aerobraking ΔV Assist / Propulsive ΔV	0	0	"
Post-Aerobraking Plane Change	60	60	"
Atmospheric Entry	0	15	"
Descent/Landing	± 0	±1,000	"
subtotal	3,819	4,856	"
Piloted TMI, 161 day trip (January 20, 2014)		3,887	m/sec
1% ΔV Margin		39	"
Gravity Losses		132	"
Outbound Midcourse		50	"
Aerobraking ΔV Assist / Propulsive ΔV (June 30, 2014)		0	"
Post-Aerobraking Plane Change		60	"
Atmospheric Entry		15	"
Descent (July 2, 2014)		1,000	"
Ascent (January 22, 2016)		5,625	"
Mars Orbit Plane Change & ECRV Rendezvous w/ TEI Stage		500	"
TEI, 154 day return (January 24, 2016)		1,476	"
Inbound Midcourse		± 50	"
subtotal		12,834	"
3 Mission / Flight TOTAL:		21,509	m/sec

Fig 5

Mars Design Reference Mission Comparison

NASA DESIGN REFERENCE MISSION (MSFC Sept 1997) comparison of wts data generated by: **Boeing Nov 1997 - Bold Font** MSFC Sept 1997 - Plan Font		2011 Mission 1 CARGO place TEI stg in Mars Orbit 4 y previous to crew return		2011 Mission 2 CARGO place asc stg on Mars 4 y before crew departs Mars		2014 Mission 3 PILOTED place crew & outbound/surf habitat on Mars surface	
TEI Stage LOX/CH4 Isp=379 sec	Aerobrake Mass (t)	**11.6**	10.7				
	Return Habitat (t)	21.6					
	Burnout Mass (t)	**4.6**	4.7				
	Propellant Mass (t)	**34.6**	31.4				
Ascent Stage LOX/CH4 Isp=379 sec							
	ECRV / Ascent Capsule (t)			5.5			
	Burnout Mass (t)			**4.2**	2.6		
	Propellant Mass (t)			***34.5**	*35.1		
	Returned Science P/L Mass (t)			0.1			
Descent Stage LOX/CH4 Isp=379 sec							
	Aerobrake Mass (t)			**18.4**	16.0	**15.3**	14.0
	Crew (t)					0.5	
	Surface/outbound Hab Mass (t)					19.3	
	Surface payload Mass (t)			32.5		9.8	
	Burnout Mass (t)			**8.9**	4.2	**6.9**	4.2
	Propellant Mass (t)			**17.3**	17.1	**17.2**	17.3
	MASS SUBTOTAL (t)	**72.3**	68.4	**86.8**	77.9	**69.0**	65.1
TMI Stage LH2 Isp=960 sec	Burnout Mass (t)	**23.2**	22.4	**25.3**	22.4	**27.9**	25.6
	Propellant Mass (t)	**46.5**	46.5	**54.7**	50.6	**52.3**	51.6
	TOTAL TMI STG MASS (t)	**69.7**	68.9	**80.0**	73.0	**80.3**	77.3
	TOTAL IMLEO (t)	**142.0**	137.3	**166.8**	150.8	**149.3**	142.4

* Asc propel produced at Mars using In-Situ resources

Direct payloads input from MSFC data 9/7/

Fig. 6

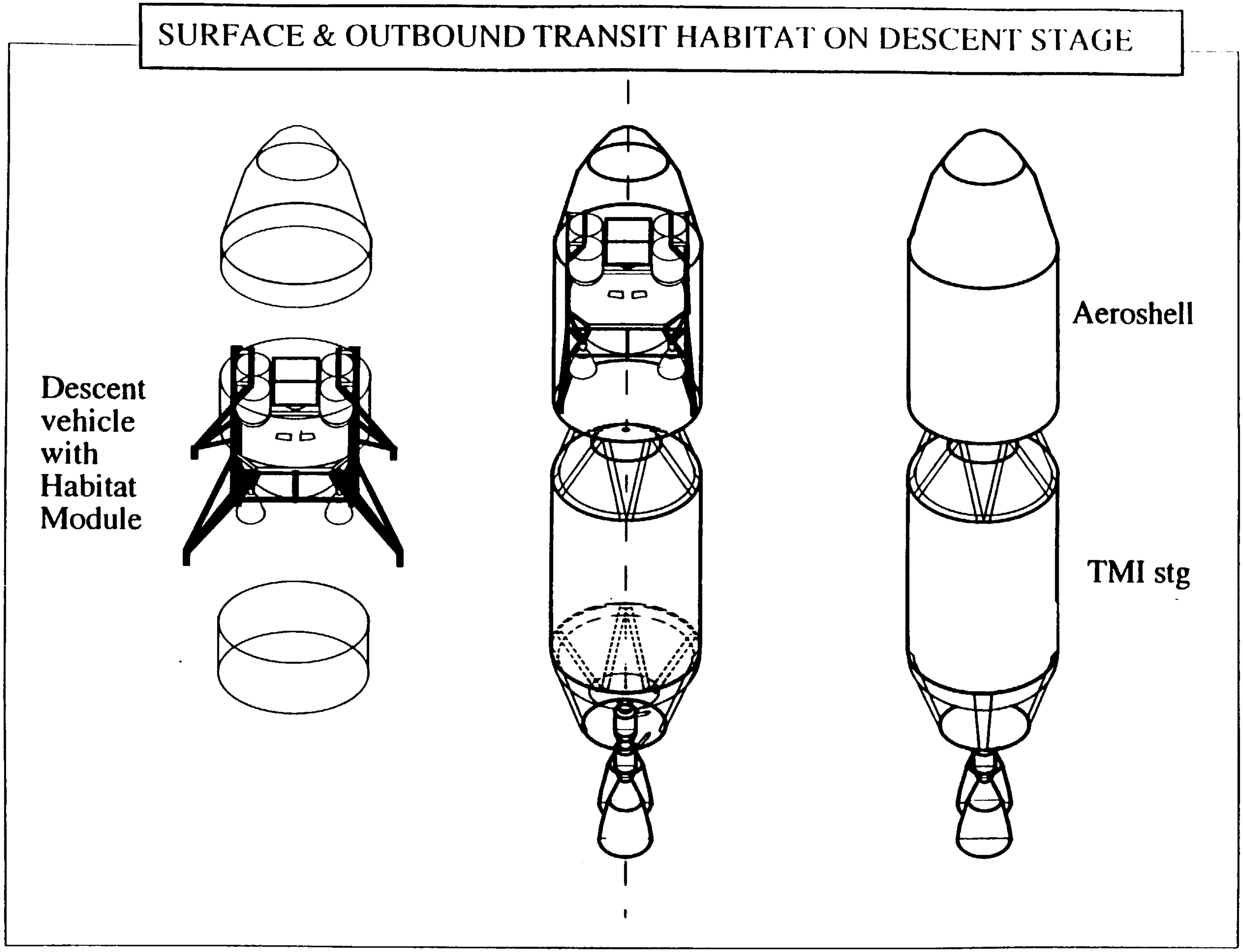

SURFACE & OUTBOUND TRANSIT HABITAT ON DESCENT STAGE
Descent
vehicle
with
Habitat
Module
Aeroshell
TMI stg

Fig. 7

Engine Performance Characteristics

	Chamber		Nozzle	Propellant	Impulse	Impulse	Density	Density	Mixture
	Max Temp (°K)	Pressure (Atm.)	Expansion Ratio (-)	Molecular Weight (AMU)	Specific, Efficiency (η)	Specific, (Secs)	Oxidizer, (kg/m3)	Fuel, (kg/m3)	Ratio, (-)
O2/MMH	3419	68	200	64	0.94	372	1141	870	2.1
O2/CH4	3560	68	200	48	0.94	379	1141	423	3.5
O2/H2	3250	68	200	34	0.98	468	1141	71	6.0
O2/H2-Be (49/51)	3250	68	200	42	0.98	553	1141	140	0.91
Nuclear H2	3000	68	200	2	0.98	960	n/a	71	n/a

FIG. 8

Mars Ascent Stage and Descent Stage Weight Statements

5625 m/s dV ascent, 1000 m/s dV descent, 21% aerobrake

Ascent propellant	Ascent stage: Dry mass: Crew cabin	Ascent stage: Dry mass: Tankage, Structure, Propulsion & RCS	Ascent stage: Propellant: Earth supplied: Ox	Ascent stage: Propellant: Earth supplied: Fuel	Ascent stage: Propellant: Earth supplied: Boiloff	Ascent stage: Propellant: Mars supplied: Ox	Ascent stage: Propellant: Mars supplied: Fuel	Ascent stage: Total: at Earth departure	Ascent stage: Total: at Mars liftoff	Descent propellant	Descent stage: Dry mass: Tankage, structure, propulsion, landing legs	Descent stage: Dry mass: In-situ plant & Seed H2	Descent stage: Dry mass: Aero-brake	Descent stage: Propellant: Earth supplied: Ox	Descent stage: Propellant: Earth supplied: Fuel	Total Desc Stage	Surf Pay-load	Total lander mass, at Earth departure
In-situ O2/CH4	5.5	4.9	n/a	n/a	n/a	29.6	8.5	9.8	47.9	O2/CH4	8.9*	9.7**	18.4	13.3	4.2	54.3	22.8	86.9
O2/H2	5.5	5.2	22.7	3.8	2.8	n/a	n/a	39.5	36.7	O2/H2	10.1	n/a	24.6	16.3	3.4	54.0	22.8	116.
In-situ O2 Earth H2	5.5	5.0	n/a	3.7	1.1	22.4	n/a	15.0	37.4	O2/H2	8.7*	4.5	17.3	11.5	2.4	44.0	22.8	81.8
O2/ H2-Be	5.5	4.7	9.2	10.1	3.7	n/a	n/a	32.9	29.2	O2/H2	9.3	n/a	22.0	14.6	3.1	48.6	22.8	104.
In-situ O2 Earth H2-Be	5.5	4.7	n/a	10.1	3.1	9.1	n/a	23.0	32.1	O2/H2	9.0*	4.5	20.1	13.3	2.8	49.5	22.8	95.3
Nuclear eng t/w = 3	5.5	11.3^	n/a	13.8	4.3	n/a	n/a	35.0	30.7	O2/H2	9.6	n/a	22.8	15.2	3.1	50.3	22.8	108.
Nuclear eng t/w = 6	5.5	7.8^	n/a	11.0	3.4	n/a	n/a	27.8	24.4	O2/H2	8.6	n/a	20.0	13.3	2.7	44.3	22.8	94.9
Nuclear eng t/w = 9	5.5	6.6^	n/a	10.2	3.2	n/a	n/a	25.7	22.6	O2/H2	8.2	n/a	19.2	12.8	2.6	42.8	22.8	91.0

* Descent stage structure and landing legs mass is a function of the load supported, which includes the ascent stage when it is fully fueled.
** seed H2 taken for CH4 production counted as descent payload. ^ includes 1 metric ton dedicated radiation shield. All masses are in metric tons.

FIG. 9

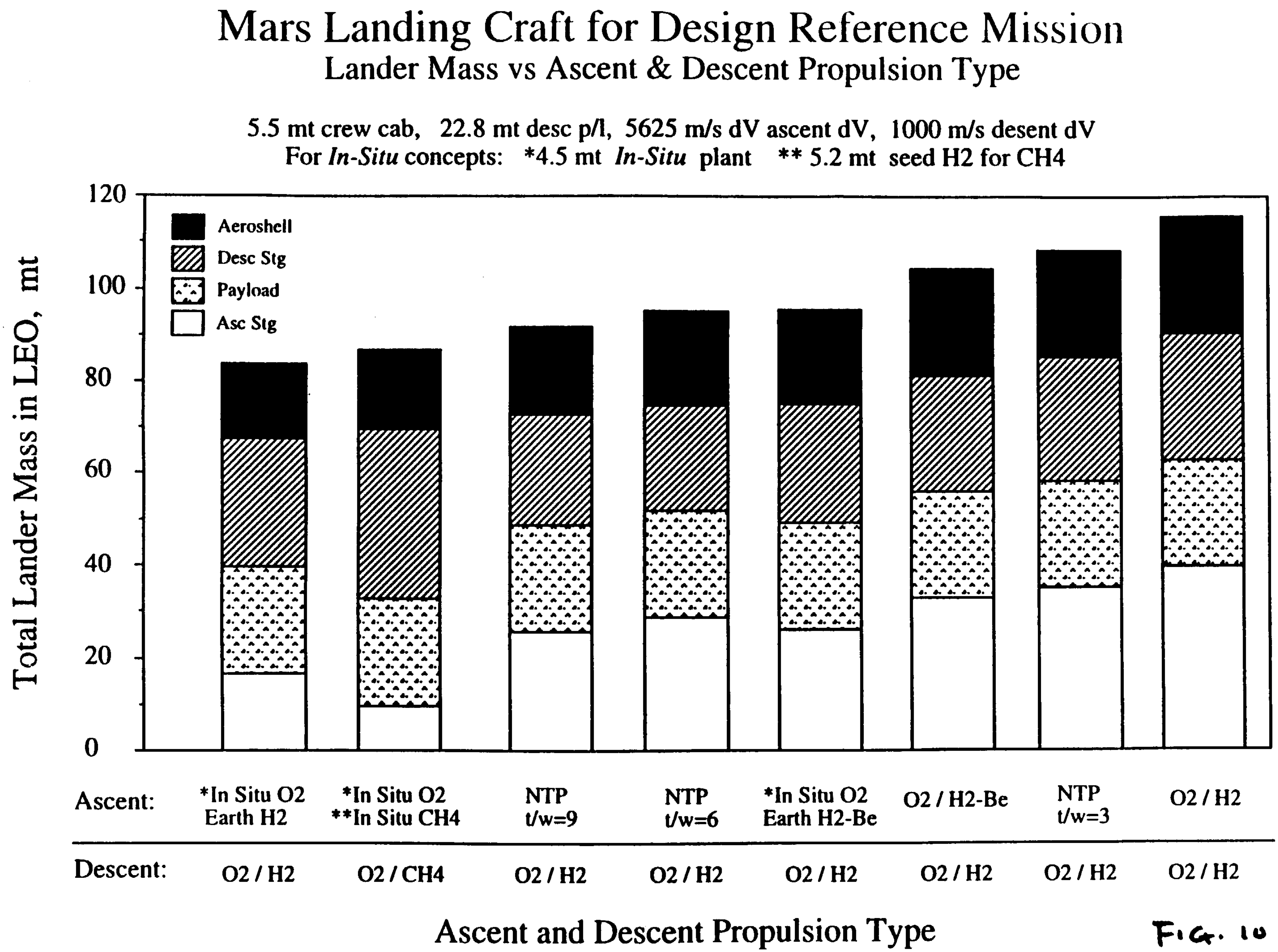
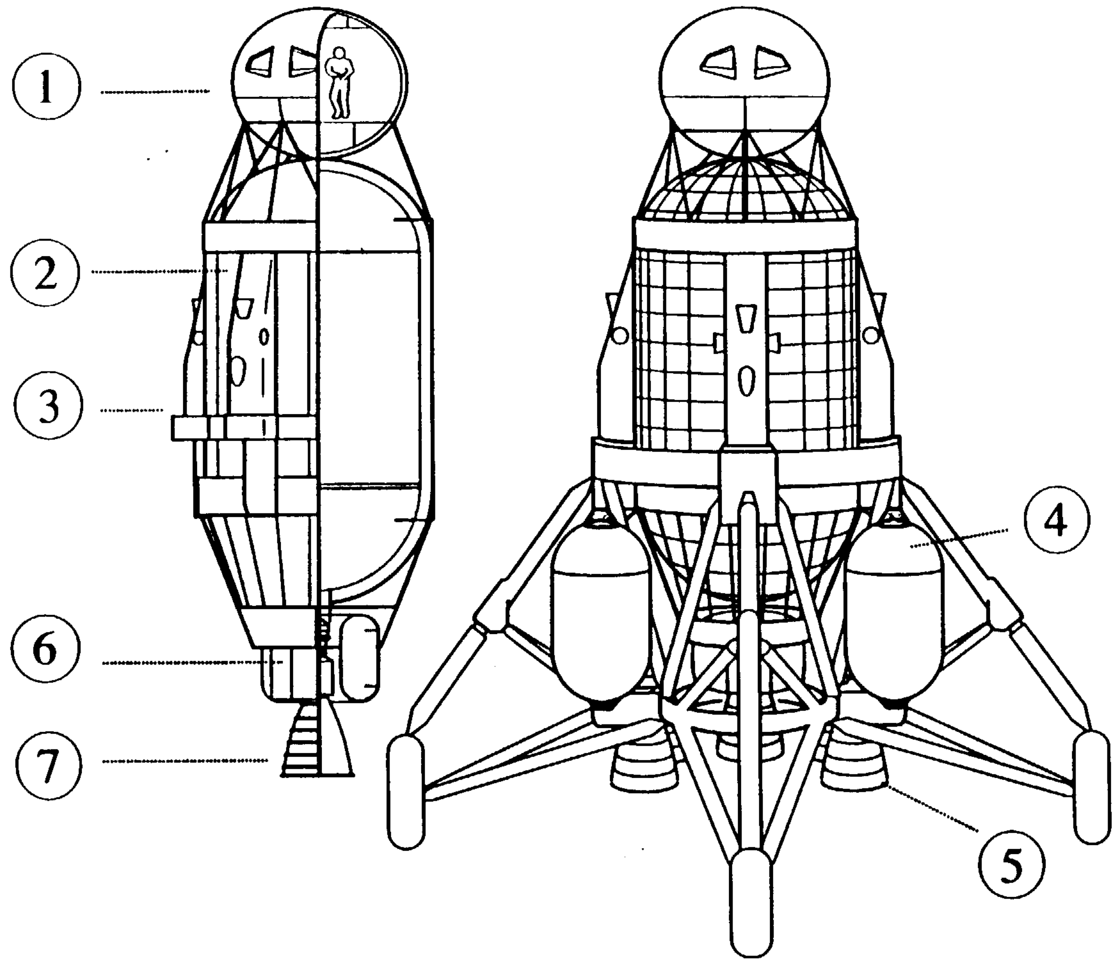

Mars Landing Craft for Design Reference Mission
Lander Mass vs Ascent & Descent Propulsion Type
5.5 mt crew cab, 22.8 mt desc p/l, 5625 m/s dV ascent dV, 1000 m/s desent dV
For In-Situ concepts: *4.5 mt In-Situ plant ** 5.2 mt seed H2 for CH4
Total Lander Mass in LEO, mt
120
100
80
60
40
20
0
Aeroshell
Desc Stg
Payload
Asc Stg
Ascent:
*In Situ O2 Earth H2
*In Situ O2 **In Situ CH4
NTP t/w=9
NTP t/w=6
*In Situ O2 Earth H2-Be
O2 / H2-Be
NTP t/w=3
O2 / H2
Descent:
O2 / H2
O2 / CH4
O2 / H2
O2 / H2
O2 / H2
O2 / H2
O2 / H2
O2 / H2
Ascent and Descent Propulsion Type

FIG. 10

Nuclear Thermal Propulsion Ascent Stage



1 Crew Cab
2 Main H2 Tank
3 Support Structure
4 Descent Stage Tanks
5 Descent Stage Engines
6 Secondary Surround H2 Tank
7 NTP engine

Fig 11

Mars Design Reference Mission Comparisons
"Consolidated" Two Vehicle Set Variation to DRM vs DRM Nominal Three Vehicle Set
2011 Cargo, 2014 Piloted Opportunities

	Payload							Trans Mars Injection			Mars Transfer Vehicle								
											MOC		TEI	EOC	Core Inerts	Transfer Habitat	Total MTV	Total IMLEO
Design Reference Mission Mode	Asc Stage	In-Situ Plant, Seed H2	Surf P/L	Surf Hab	Desc Stage	Lander Aero Shell	Total	Propel	Inerts	Total TMI	MTV Aero Shell	Propel	Propel	Propel	Tankage Engines Truss			
Mission 1 2011 TEI stg	n/a	n/a	n/a	n/a	n/a	n/a	n/a	46.5	23.2^	$\boxed{69.7}$	11.6	1.2	33.4	n/a	4.6	21.6	$\boxed{72.3}$	142.0
Mission 2 2011 Lander	9.7	9.6	22.8	n/a	26.3	18.4	$\boxed{86.9}$	54.5	25.5^	$\boxed{80.0}$								166.9
Mission 3 2014 Crew	n/a	n/a	10.3	19.3	24.1	15.3	$\boxed{69.0}$	52.3	27.9^	$\boxed{80.3}$								149.3
																		Total 458.2
Alternate "Consolidated" Mission Mode Single Round Trip Capable Transfer Stage																		
Mission 1 2011 Lander	9.7	9.6	22.8	n/a	26.3	18.4	$\boxed{86.9}$	54.5	25.5^	$\boxed{80.0}$								166.9
Mission 2 2014 Crew	n/a	n/a	10.3	19.3	24.1	15.3	$\boxed{69.0}$	74.4	17.6*	$\boxed{92.0}$	n/a	25.1**	12.7	n/a	28.0	21.6	$\boxed{91.7}$	252.7
																		Total 419.6

All masses are in metric tons.

* TMI Tankage only

* * Includes 9.4 mt of TMI propellant stored in MOC tanks

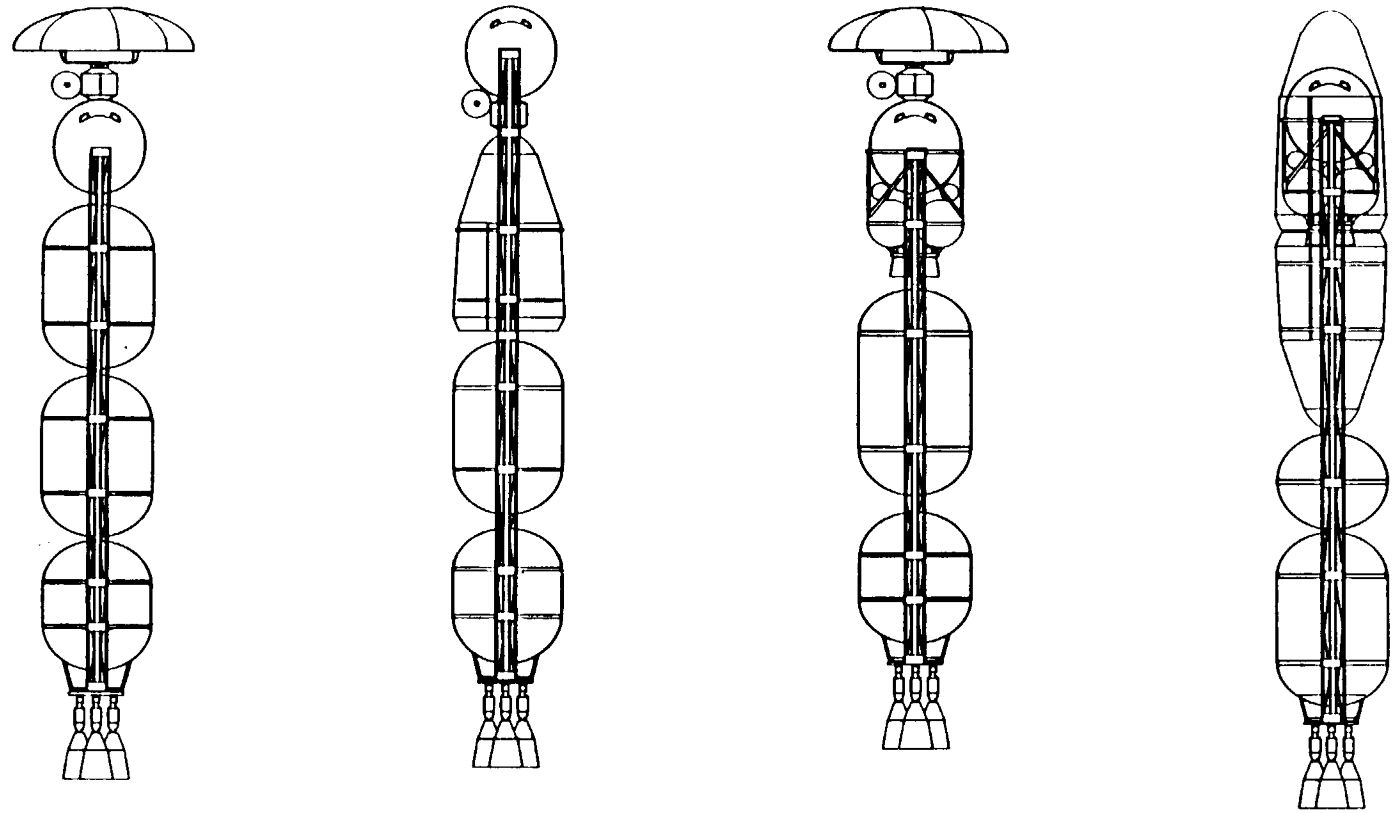
^ Separate TMI stage with its own NTP engine cluster.

$\boxed{}$ Principle vehicle element weights to be put into Earth orbit by launch vehicle

Fig 12

MARS TRANSFER VEHICLE CONFIGURATION OPTIONS

Representative illustrations - tank sizes not necessary to scale



TYPE 1	TYPE 2	TYPE 3	TYPE 4
<u>Transfer Stage</u> TMI - NTP MOC - NTP TEI - NTP	<u>Transfer Stage</u> TMI - NTP MOC - NTP TEI - NTP	<u>Transfer Stage</u> TMI - NTP MOC - NTP TEI - Chemical	<u>Transfer Stage</u> TMI - NTP MOC - Aerocapture TEI - Chemical
<u>Lander Stage</u> Propulsively Captured w MTV	<u>Lander Stage</u> MOC-Aerocapture	<u>Lander Stage</u> Propulsively Captured w MTV	<u>Lander Stage</u> MOC - Aerocapture

FIG. 13

Mars Transfer Vehicle ETO Packaging Illustration
Type 2 Configuration

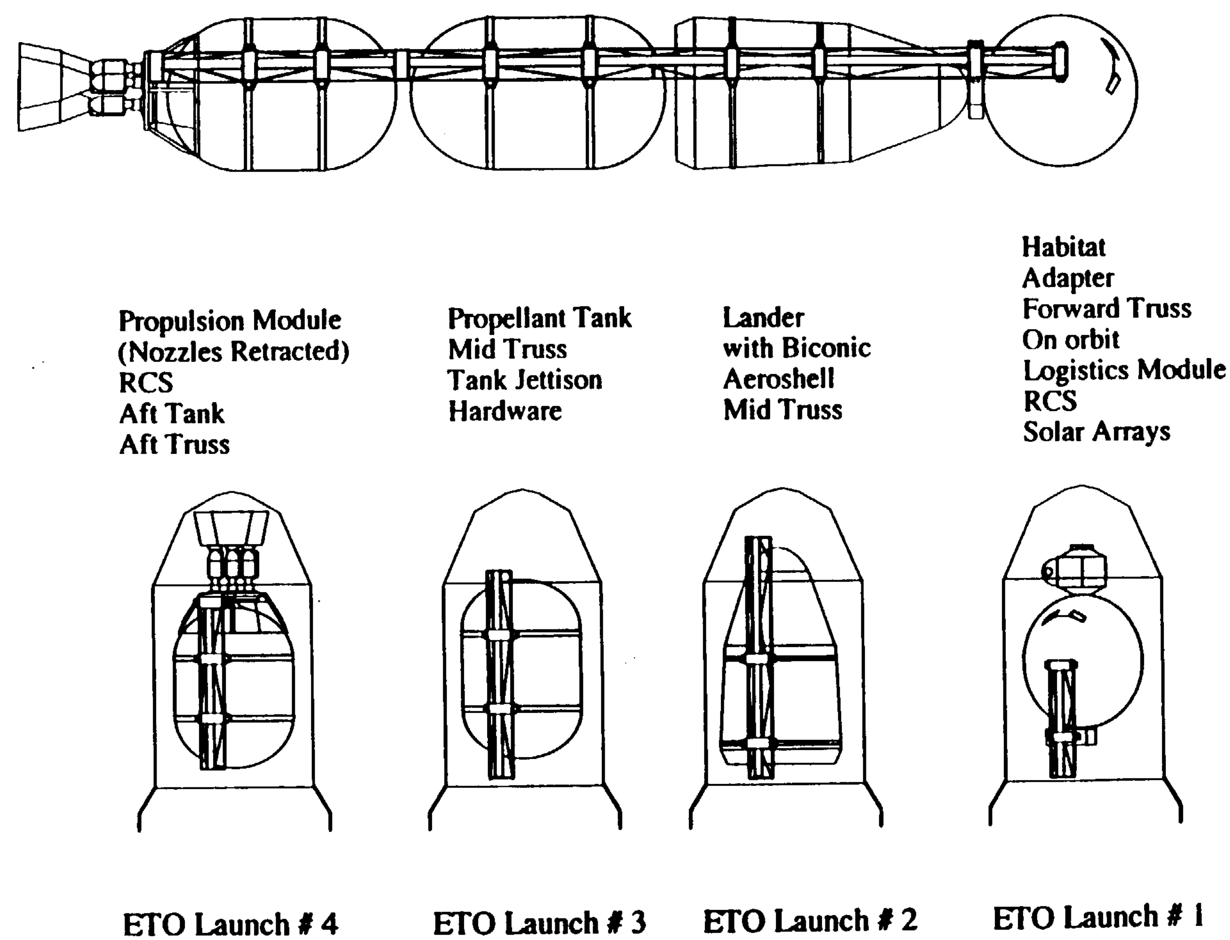


FIG. 14

Mars Transfer Vehicle Propulsion Options Comparison
for Non-ISPP, Full Round Trip Capable Mars Transfer Vehicle with Lander
2014 Opportunity

Configuration Type	Propulsion System				TEI			MOC				Core Mars Transfer Vehicle							Payload	Total IMLEO
	TMI propul	MOC MTV propul	MOC lander	TEI propul	Prop	Stage inerts (or drop tank^)	Total	MOC Prop	Portion of TMI Propel	Sep-arate Tank-age	Total	MTV Aero Shell	TEI Prop	EOC Prop	Core Inerts Tankage Engines Structure	Boil-off	Transfer Habitat	Total Core stage	*Lander Total	
Type 1	NTP	—— NTP ——		NTP	71.0	17.1^	$\boxed{88.1}$	36.6	27.0	10.5	$\boxed{74.1}$	n/a	15.0	n/a	34.6	4.9	27.0	$\boxed{81.5}$	$\boxed{47.8}$	291.6
Type 2	NTP	NTP	Aero capture	NTP	62.1	15.3^	$\boxed{77.4}$	20.9	18.5	7.1	$\boxed{46.5}$	n/a	14.0	n/a	29.1	3.8	27.0	$\boxed{73.9}$	$\boxed{52.0}$	249.8
Type 3	NTP	—— NTP ——		O2/ H2	44.0	28.8	$\boxed{72.8}$	23.0	40.	10.7	$\boxed{73.7}$	n/a	20.5	n/a	4.4	1.2	27.0	$\boxed{53.1}$	$\boxed{47.8}$	247.4
Type 4	NTP	Aero capture	Aero capture	O2/ H2	82.2	32.7	$\boxed{115}$	n/a	n/a	n/a	n/a	14.5	21.4	n/a	4.8	1.2	27.0	$\boxed{69.0}$	$\boxed{52.0}$	235.9
	NTP	Aero capture	Aero capture	O2/ CH4	86.1	33.6	$\boxed{120}$	n/a	n/a	n/a	n/a	15.9	27.5	n/a	4.1	1.0	27.0	$\boxed{75.5}$	$\boxed{52.0}$	247.3
	NTP	Aero capture	Aero capture	O2/ H2-Be	79.6	32.3	$\boxed{112}$	n/a	n/a	n/a	n/a	13.7	17.6	n/a	5.0	1.9	27.0	$\boxed{65.2}$	$\boxed{52.0}$	229.4

^ TMI propellant in drop tank attached to Core stage; Core stage engines used for TEI.

* Lander carries NTP ascent stage (eng t/w=6) and surface payload

Major Vehicle Element for Launch Vehicle

FIG. 15

Mars Design Reference Mission Comparisons
DRM Vehicle Set vs Non-In-Situ All NTP Transfer Vehicle set
2011 Cargo, 2014 Piloted Opportunities

Design Reference Mission Mode	Payload						TMI		MOC			Core Mars Transfer Vehicle								NOTES
	Asc Stage	In-Situ Plant, Seed H2	Surf P/L	Surf Hab	Desc Stage	Lander Aero Shell	Propel	Inerts	MOC Prop	Portion of TMI Propel	Sep-arate Tank-age	MTV Aero Shell	TEI Prop	Core Inerts Tankage Engines Structure	Transfer Habitat	Total Core Stage	Total IMLEO	ETO Flights Req'ed	ETO Max Cap	
Mission 1 2011	n/a	n/a	n/a	n/a	n/a	n/a	46.5	23.2	1.2	n/a	n/a	11.6	33.4	4.6	21.6	72.3	142.0	2	72	
Mission 2 2011	9.7	9.6	22.8	n/a	26.3	18.4	54.5	25.5									166.9	2	87	
Mission 3 2014	n/a	n/a	10.3	19.3	24.1	15.3	52.3	27.9									149.3	2	80	
																Total	458.2	6	87	
Type 2 **Vehicle, non-DRM Mission Mode Vehicle Set**																				
Mission 1 2011	n/a	n/a	32.5	19.3	23.1	14.4	63.5	15.4	10.8	n/a	n/a	n/a		15.6			195.3	3	89	ISPP Eliminated
Mission 2 2014	27.0*	n/a	n/a	n/a	14.1	11.0	62.1	15.3^	20.9	18.5	7.1	n/a	14.0	29.1	27.0	73.9	249.8	4	77	
																Total	445.1	7	89	
Type 3 **Vehicle, non-DRM Mission Mode Vehicle Set**																				
Mission 1 2011	n/a	n/a	32.5	19.3	23.1	14.4	63.5	15.4	10.8	n/a	n/a	n/a		15.6			195.3	3	89	ISPP & Aerocapture Eliminated
Mission 2 2014	27.0*	n/a	n/a	n/a	14.1	6.7	44.0	28.8	23.0	40.0	10.7	n/a	20.5	4.4	27.0	53.1	247.7	4	74	
																Total	443.0	7	89	

^ TMI propellant in drop tank attached to Core MTV

* NTP Ascent stage

FIG. 16

Reusability Evaluation: Earth Capture Penalty for
Single Mars Transfer Vehicle Recapture in to Low Earth, and *High Elliptical Earth Orbits
NTP for TMI, MOC, TEI and EOC maneuvers, 2014 Piloted Opportunity
Type 2 Vehicle Configuration

Vehicle Earth Return	Payload	Trans Mars Injection			Mars Transfer Vehicle								
					MOC		TEI	**EOC	Core Stg Inerts	Boil-off	Transfer Habitat	Total MTV	Total IMLEO
	Lander Total	Propel	Tankage	Total TMI	MTV Aero Shell	Propel	Propel	Propel	Truss, Engs, Tankage, RCS				
Expendable	52.0	87.1	20.0	107.2	n/a	18.8	13.0	n/a	25.7	3.4	30.0	90.9	250.1
High Elliptical Earth Orbit													
1st Mission 2014 Departs LEO Returns HEEO	52.0	110.0	24.7	134.7	n/a	26.1	18.1	13.5	33.7^	7.8	30.0	129.2	315.9
2nd Mission 2016 Same MTV Departs HEEO returns HEEO	52.0	33.5	9.9	43.3	n/a	25.7	17.8	13.3	33.7^	7.7	30.0	128.2	223.5
or 2nd Mission 2016 Same MTV Departs HEEO Expendable EOC	52.0	28.4	8.7	37.1	n/a	20.6	14.3	n/a	33.7^	3.7	30.0	102.3	191.4
Low Earth Orbit	52.0	154.1	33.5	187.6	n/a	40.1	26.4	40.1	50.3	16.0	30.0	202.9	442.5

All masses are in metric tons.

* HEEO defined herein to be 800 km by 45000 km, 14 hr Elliptical Orbit.

** Earth Arrival V infinity = 5.243 km/s. For return to HEEO, EOC dV = 1900 m/s. For return to LEO EOC dV = 4376 m/s

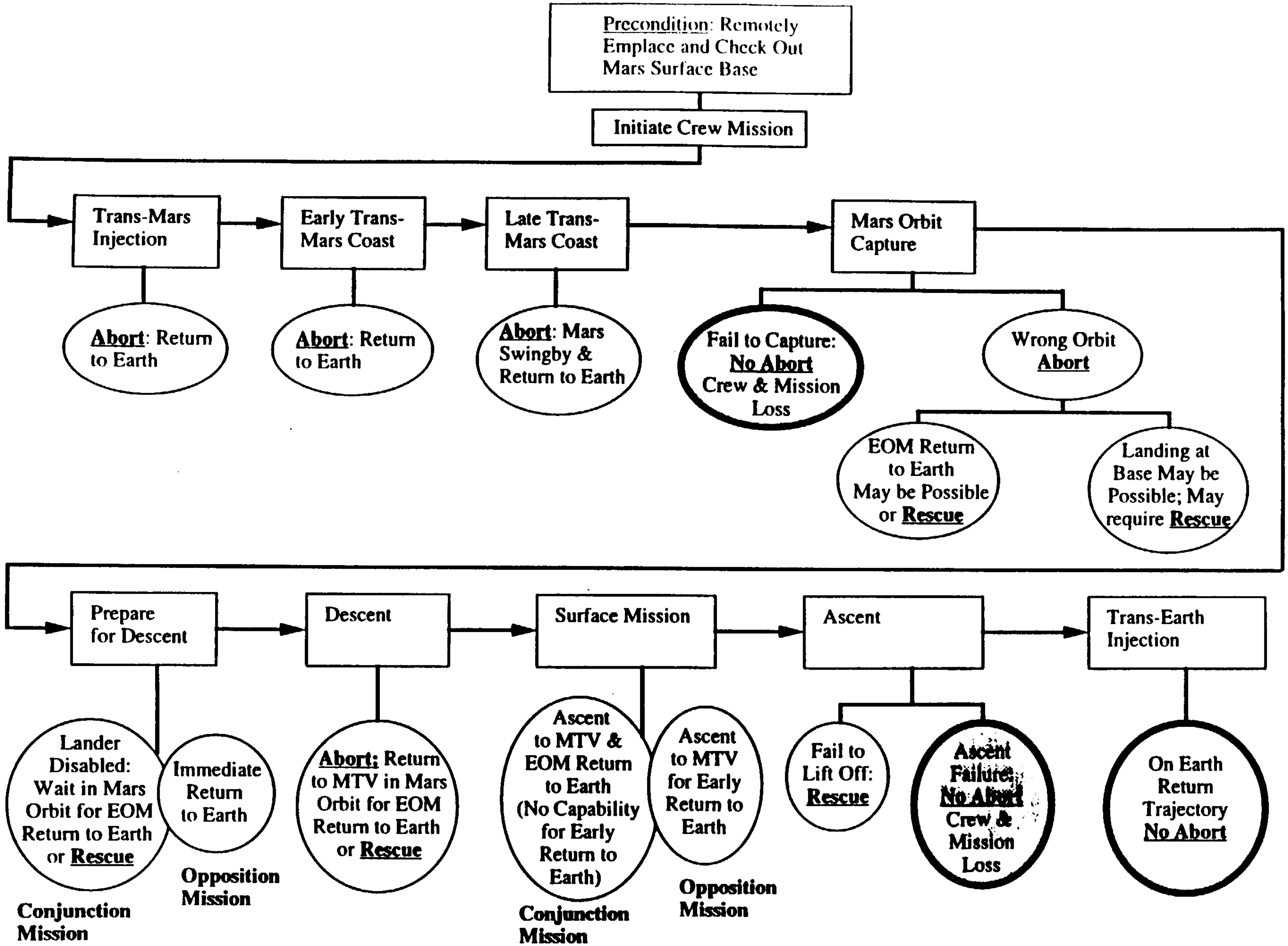
^ Same Core Stage - MOI / TEI / EOC Tankage, Engine Cluster, Structure

FIG. 17

Architectures & Abort Options

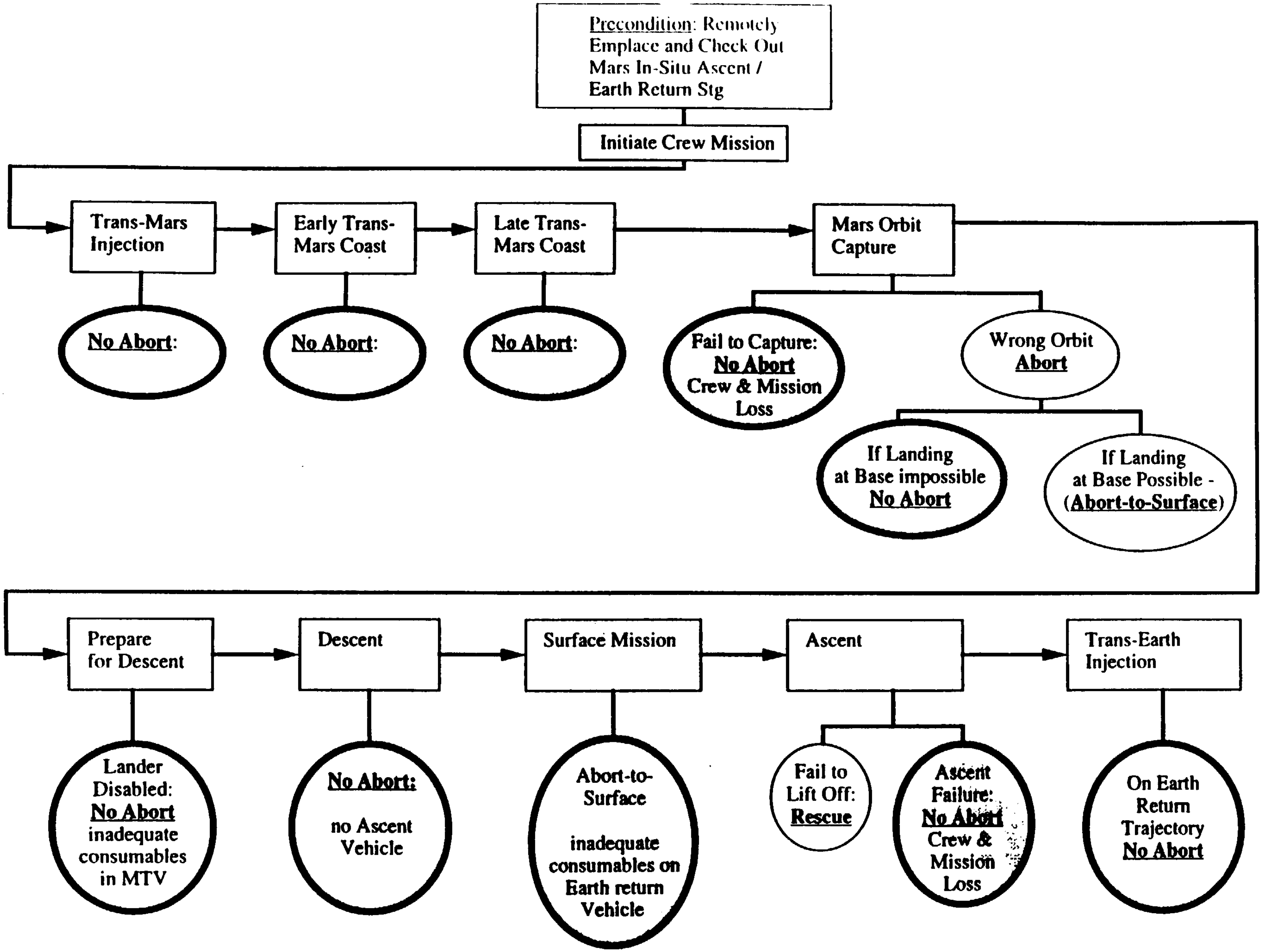
Architecture Type	Mission Trajectory	Primary Abort Approach	Pre Emplacement Missions
Full Round Trip Capable Crew Transfer Vehicles	Conjunction	Return to Earth whenever possible or wait for next return opportunity	1 Surface Habitat System
	Opposition	Return to Earth whenever possible	1 Surface Habitat System
Mars Direct (ala Zubrin)	Conjunction	Abort-to-Surface	1 In-Situ Lander with Return Hab
Design Reference Mission	Conjunction	Abort-to-Surface	1 In-Situ Lander 2 TEI Stage
Variation to Design Ref Mission (Section 3.1.3)	Conjunction	Return to Earth whenever possible or wait for next return opportunity	1 In-Situ Lander
Nominal Electric (STCAEM)	Conjunction or opposition	Abort-to-Surface or return on reduced power	1 Surface Habitat System

FIG. 18



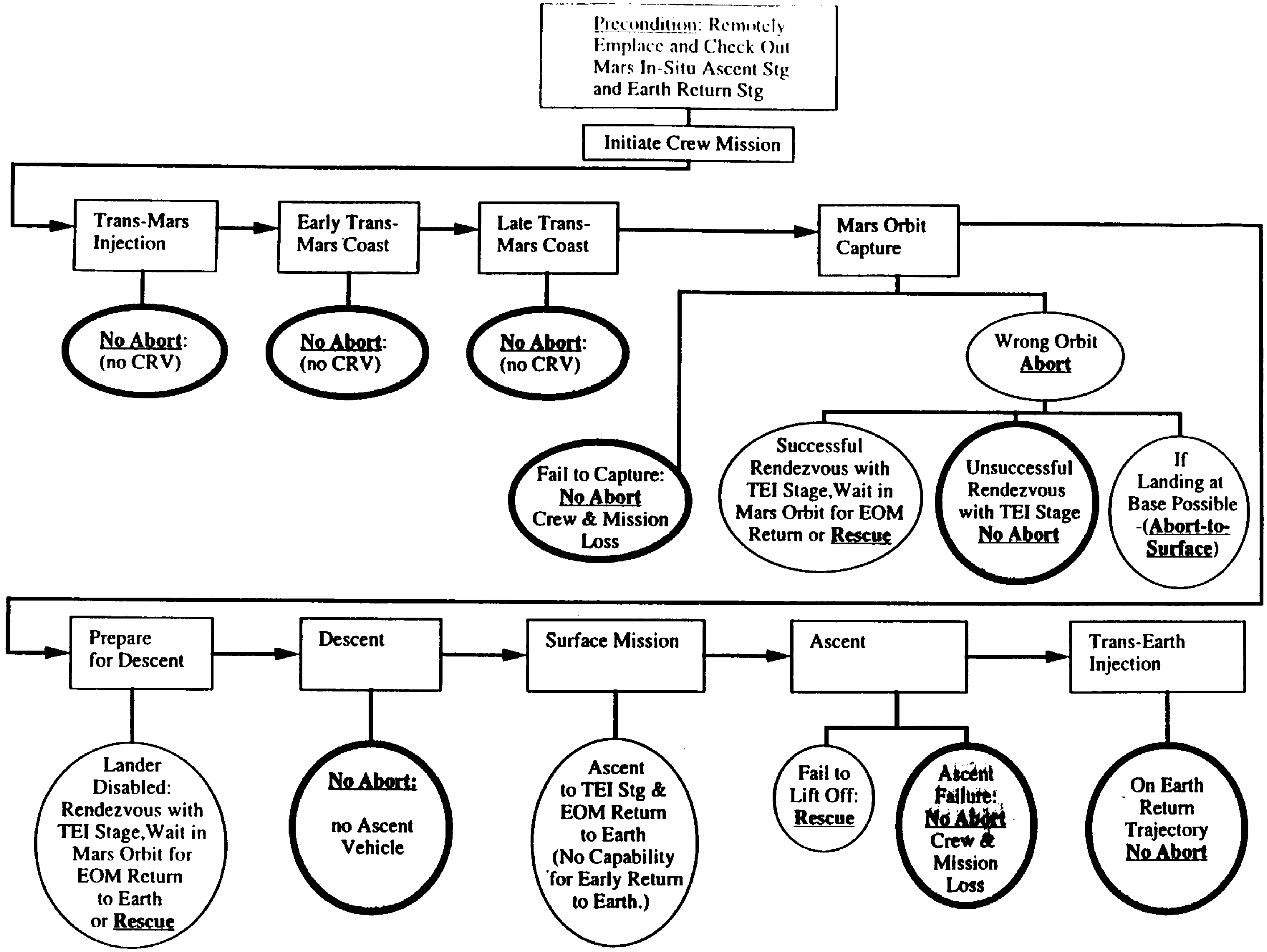
Abort Flow Diagram for Full Round Trip Capable MTV, Non In-Situ (Conjunction or Opposition)

Fig 19



Abort Flow Diagram for Mars Direct Mission

Fig 20



Abort Flow Diagram for Mars Design Reference Mission

Fig 21