JOURNAL OF SPACECRAFT AND ROCKETS
Vol. 42, No. 5, September–October 2005

Trajectory Options for Manned Mars Missions

Brice N. Cassenti*
Pratt and Whitney Aircraft, United Technologies Corporation, East Hartford, Connecticut 06108

Manned missions to Mars have been the subject of studies for more than 50 years. Trajectory options examined include opposition and conjunction class missions, a direct landing on Mars with in situ propellant production, and a dash to Mars by the crew coupled with a hyperbolic rendezvous at Mars, and possibly at the Earth. This study examines the effect of trajectory choice on the initial mass in low Earth orbit and illustrates the important decisions that must be made when balancing risk and cost. Three different propulsion systems are used in the comparison, including chemical, nuclear thermal, and gas core nuclear. Of the three, nuclear thermal propulsion appears to be the good choice because the mass launched into Earth orbit is low, and because of the testing in the late 1960s, the technological risk associated with nuclear thermal propulsion appears to be small. Of the trajectories examined, the lowest cost trajectory appears to be a dash mission, but for this trajectory a hyperbolic rendezvous is required, which greatly increases the mission risk. The choice of the trajectory is crucial to success because decisions to reduce the cost to a manageable level must be carefully balanced against the risk to the crew.

Nomenclature

f_R	=	reserve propellant mass as a fraction of the required propellant mass, m_{prop}
f_T	=	tank mass as a fraction of the total propellant mass, $(1 + f_R)m_{\text{prop}}$
g_0	=	acceleration of gravity at the surface of the Earth, m/s^2
I_{sp}	=	engine specific impulse, s
m_{prop}	=	required propellant mass, tons
MR	=	mass ratio (initial to final) from required speed change
T_D	=	dash transit time from habitat to Mars, years
T_S	=	stay time on Mars, years
T_T	=	total dash mission time, $T_D + T_S$, years
v_L	=	Mars lander heliocentric velocity, km/s
v_M	=	Mars heliocentric velocity, km/s
v_S	=	habitat heliocentric velocity at start of dash mission, km/s
β	=	angle between habitat heliocentric velocity and Mars heliocentric velocity, deg
ΔV	=	change in speed, km/s
Δv_{dash}	=	Mars lander heliocentric velocity relative to habitat heliocentric velocity, km/s
$\Delta\beta$	=	angle between habitat heliocentric velocity and lander heliocentric velocity, deg

Introduction

FROM the wide variety of piloted Mars missions that have been proposed, the choice of trajectory is among one of the most important decisions that must be made in developing a detailed mission scenario. The trajectory will have an enormous influence on the costs and the risk, including both mission and technological risks.

Von Braun,[1] in the early 1950s, examined a piloted Mars mission using minimum-energy Hohmann trajectories. The mission consisted of an Earth to Mars transit phase of 0.71 years (260 days), a stay time on Mars of 1.24 years (449 days), and Mars to Earth transit of 0.71 years. The total propulsive change in velocity, from low Earth orbit to Earth entry, is 9.6 km/s assuming aerodynamic drag is used to land on Mars and to reenter at Earth. About 3.6 km/s is required to reach a low circular orbit about Mars.

Although Von Braun's choice for the trajectory appears to be the most efficient, there are other choices that can further reduce the total mass required to complete the mission. For example, Donahue[2] has recently examined a mission scenario that removes the transfer into and out of Mars orbit. The mission consists of a flyby of Mars with the transit habitation section and a short stay time on Mars. The lander is separated from the transit section before arrival at Mars and then makes a "dash" for the landing on Mars before the flyby of Mars by the transit section. Equivalently, the heliocentric trajectory of the transit section can be changed, leaving the lander on the initial heliocentric trajectory to intercept Mars. The transit section arrives at Mars a few weeks later. The crew in the lander at this time must launch and execute a hyperbolic rendezvous with the transit section as it passes Mars. The total change in speed required from low Earth orbit is about 3 km/s less for the transit stage and about 1.5 km/s more for the lander. The change in speed is actually better than the difference of 1.5 km/s because the speed decrease of 3.0 km/s includes the entire spacecraft, and this has been traded for a speed increase of 1.5 km/s that only includes the lander. The dash mission has the major disadvantage of requiring a hyperbolic rendezvous, greatly increasing the mission risk, although it has a major advantage of decreasing the initial mass in low Earth orbit (IMLEO). For example, if a gravity assist is used at Venus this flyby/dash mission saves about 200 t, or 40% of the IMLEO, when compared to a Hohmann trajectory (which requires a mass of more than 500 t for the IMLEO) if nuclear upper stages are used for both mission scenarios.[2]

Even the choice of pure Hohmann trajectories can have elements that can reduce the total mass required. Zubrin[3] and Zubrin et al.[4] have proposed an all-chemical piloted mission to Mars (nuclear power would be used on the surface of Mars) involving two heavy-lift launches. The mission yields a considerable savings in mass by using in situ propellant production. Hohmann transfers can be used allowing a long stay time on the surface of Mars, but the authors actually increased the stay times by shortening the transit times to 180 days each way, and they could then add a compensating increase in the Mars stay time. The first launch goes directly to Mars to land the module the crew uses while on the surface. An in situ propellant plant, powered by the nuclear power source, produces oxygen from the carbon-dioxide atmosphere, and with Earth-supplied hydrogen produces methane fuel. The crew is sent on the second launch to rendezvous with the first module on the surface of Mars. On the

Presented as Paper 2002-3787 at the AIAA/ASME/SAE/ASEE 37th Joint Propulsion Conference, Indianapolis, IN, 8–10 July 2002; received 2 April 2004; revision received 26 August 2004; accepted for publication 30 August 2004.

*Fellow, Applied Mechanics, 400 Main Street, Mail Stop 163-07; brice.cassenti@pw.utc.com. Associate Fellow AIAA.

piloted outbound vehicle, no propellant is carried for the return trip to Earth. Hence, this approach is high risk because the crew module has not been sized to have return propellant, and, thus, a modified mission was proposed to include a third launch of a backup lander with a propellant-manufacturing module at the same time the crew is launched.

There are in fact many trajectory variations that must be considered. In Ref. 5 Waldberg has an excellent description of a wide variety of opposition and conjunction missions. These include Venus, Earth, and Mars gravity assists and provides a good summary of the available trajectory options.

One class of missions, first proposed by Crocco[6] and later generalized by Ruppe,[7] includes a launch from Earth that passes Mars and returns back to Earth such that the total time to return is an integral multiple of its heliocentric period and is also an integral multiple of the Earth's heliocentric period. For example, the spacecraft could be placed on a one-year heliocentric trajectory that will also flyby Mars. The one-year mission trajectory would allow a short stay time on Mars, as in the dash/flyby mission of Donahue,[2] and a complete trip in one year. Because, for the one-year Crocco mission, the perihelion must be inside the orbit of the Earth, 1) the radiation risk increases, and 2) the thermal radiation problems are more severe. The velocity change requirement is about 13 km/s larger than Hohmann trajectories for launch to Mars. A more efficient mission would use two-year or 1.5-year heliocentric orbits for the spacecraft. The trajectory in the two-year mission and the 1.5-year heliocentric mission could have a perihelion at the Earth's orbit, so that thermal radiation problems would not be enhanced. These mission scenarios would require considerably less energy than the one-year mission by several kilometers per second for launch to Mars and are competitive with the standard Hohmann transfer.

Cycler orbits between Earth and Mars, proposed by Byrnes et al.,[8] are closely related to Crocco orbits. They proposed using two spacecraft with 15-year return cycles that would require propulsive maneuvers of several kilometers per second. In an extension of this idea, one or more spacecraft could be launched to encounter Earth and Mars at regular intervals. For example, if 17-year repeating orbits are used (because the time between closest perihelion passes with respect to the Earth varies between 15 and 17 years) then a set of 17 craft could be used to provide yearly trips between the Earth and Mars with a slight propulsive maneuver required at the Earth. These trajectories have the same risks as the dash missions proposed by Donahue and are higher cost, but they provide an infrastructure for continuous exploration, settlement and trade.

The choice of trajectory will drive not only the cost but also the mission and the technological risks. The effect of trajectory choice will be illustrated by comparing the mass requirements for Hohmann and Crocco free-return trajectories for three propulsion technologies: chemical rockets, nuclear thermal rockets, and gas core nuclear rockets. The IMLEO will be used as the measure of cost, and both technological and mission risk will be assessed qualitatively. The results will show that the choice of trajectory has a significant effect on the cost and should be chosen only after a thorough risk assessment.

Mission Trajectories

The heliocentric trajectories chosen determine the velocity requirements for the mission. The standard Hohmann mission is the classical minimum for the velocity requirements and was the first to be examined in detail. The Hohmann mission will provide the standard by which other trajectories are compared. Two other classes of trajectories are described. These include 1) free return (or generalized Crocco) and 2) dash/swingby.

Hohmann Mission

For Earth-to-Mars missions, Hohmann trajectories will provide an estimate for the minimum velocity requirements for circular orbits. A spacecraft starting in low Earth orbit will need an additional 3.36 km/s to achieve escape velocity and must add a hyperbolic excess velocity of about 2.96 km/s. Throughout this paper, Mars is assumed to move in a circular orbit of 1.53 astronomical units (AU), but its eccentricity is relatively large and can have a significant effect on the actual velocity requirements. At Mars the spacecraft goes into orbit, and the crew leaves for the surface. A year and a quarter later Earth and Mars are correctly aligned for the return. The crew on the surface performs a rendezvous with the orbiting stage and then must add 1.36 km/s to reach escape velocity and an additional 2.73 km/s excess hyperbolic velocity for the proper Hohmann trajectory to return to Earth.

Crocco Free-Return Missions

The one-year mission first proposed by Crocco[6] is a special case of Mars free-return trajectories. For the Crocco one-year mission the spacecraft is launched into an orbit with a period of one year (i.e., a semimajor axis of 1 AU), so that the spacecraft returns to the same point the Earth will be one year later. If the aphelion is chosen to be tangent to the orbit of Mars, then the perihelion is inside the orbit of Venus. The velocity of the spacecraft with respect to the sun at the orbit of the Earth is exactly the orbital velocity of the Earth, but the angle between the velocity vectors is considerable and results in an excess hyperbolic velocity of 16.42 km/s, which is well outside the range of currently feasible chemical systems.

A generalized form of the Crocco mission, proposed by Ruppe[7] in 1961, has the spacecraft on a trajectory that returns in an integral number of years to the launch point. For example, the spacecraft can orbit the sun with a two-year orbit or even a 3/2-year orbit, which will rendezvous with the Earth in three years after the spacecraft orbits the sun twice. These missions will be referred to as generalized Crocco missions and will be differentiated using the heliocentric orbital period. The two-year mission requires an excess hyperbolic velocity at the Earth of 5.08 and 9.15 km/s for a launch from Mars, whereas the 1.5-year mission has a 3.34-km/s excess hyperbolic velocity at the Earth and 4.53 km/s at Mars. The launch velocity at the Earth is assumed to be parallel to the heliocentric velocity of the Earth. The angle between the heliocentric velocity vector of the spacecraft at Mars and the heliocentric velocity of Mars drives the hyperbolic excess velocity requirement at Mars.

Dash Segment

The generalized Crocco missions provide a flyby capability with a free return to Earth. A two-year mission flyby mission can be readily performed using nuclear thermal rockets. But a landing will require the crew to leave the habitat spacecraft and use a separate lander to dash to Mars. After a short stay the habitat will pass by Mars, and the ascent portion will, at the time the transfer vehicle approaches its swingby periapsis point at Mars, launch ahead of it on a preplanned hyperbolic departure path. After ascent, the ascent stage is positioned to allow the transfer stage to catch up to it as they both recede away from the planet. The ascent burn provides enough ΔV to reach trans-Earth injection velocity; this would require only an additional 1.0 to the 5.3 km/s ΔV required for a nominal ascent-to-parking orbit ascent (about 20%). The ascent and transfer vehicles, traveling at the same speed in parallel flight, would rendezvous with the small reaction control system of either vehicle used to affect the small closing ΔV. The landing portion can be a direct entry or use an aeroshell to go into orbit and then land using the Martian atmosphere to slow down at the surface.

The velocity increment required for the dash portion of the mission can be estimated by considering the habitat spacecraft and Mars to be moving in a straight line at constant speed as the habitat nears Mars. The motion is illustrated in Fig. 1.

The heliocentric spacecraft speed v_s can be readily found from the conservation of energy, and the angle between the habitat and Mars velocity vectors β can be found form the conservation of angular momentum. The velocity required for injecting the lander into the dash trajectory is, for this approximation, a function of 1) the time the spacecraft spends on the dash trajectory T_D and 2) the total time the crew is separated from the habitat T_T. The difference $T_S = T_T - T_D$ is the time the crew spends on the surface. Using the angle between the heliocentric velocities of the spacecraft and Mars at the time the Mars transfer vehicle arrives at Mars, then

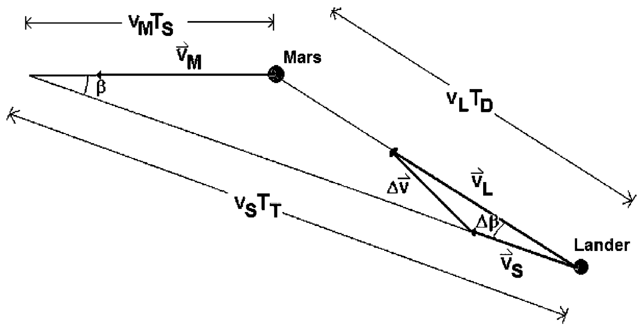

Fig. 1 Approximate heliocentric motion near Mars.

from Fig. 1

$$v_L = \sqrt{\left(v_M \frac{T_S}{T_D}\right)^2 + \left(v_S \frac{T_T}{T_D}\right)^2 - 2\left(v_M \frac{T_S}{T_D}\right)\left(v_S \frac{T_T}{T_D}\right)\cos\beta} \quad (1)$$

Let $\beta + \Delta\beta$ be the angle between the lander and Mars velocity vector. Again using Fig. 1, the angle $\Delta\beta$ is

$$\Delta\beta = \sin^{-1}[(v_M T_S / v_L T_D)\sin\beta] \quad (2)$$

The cosine law can now be used to find the magnitude of the required velocity change Δv_{dash} as

$$\Delta v_{\text{dash}} = \sqrt{v_S^2 + v_L^2 - 2 v_L v_S \cos(\Delta\beta)} \quad (3)$$

For example, for the 3/2-year Crocco mission and T_s/T of 0.25 (e.g., four weeks total time and one week on the surface), the velocity change Δv is 2.75 km/s. The short stay time applies only to the initial missions. Subsequent missions can make use of previously launched habitats.

Dash/Swingby Mission

The dash mission can also include a gravity assist at Venus[2] to decrease the total trip time, which decreases the time the crew spends weightless but adds a closer approach to the sun. This dash mission with a Venus assist gravity is essentially a sprint mission,[5] and, hence, the dash/swingby can provide IMLEO savings over the Hohmann and generalized Crocco missions.

Cycler Missions

Byrnes et al.[8] have shown that after the Mars transfer vehicle (MTV) returns to Earth, a gravity assist and a propulsive maneuver can return the Mars transfer vehicle to Mars after one (or more) heliocentric orbits. As little as two of these vehicles could provide ready access to Mars. These missions are an extension of the Crocco free-return missions and will require not only a hyperbolic rendezvous at Mars but also require one at the Earth.

Mission Sizing

IMLEO is the measure that will be used to compare the requirements (and ultimately the cost) for the mission scenarios discussed. The mass estimates presented are to be used only for comparison and were completed using the mass estimates in Ref. 9 and summarized in the Appendix.

Vehicle Modules

The vehicle consists of several modules. The mass estimates are adopted from Donahue.[2] The Mars ascent module (16 tons with propellant and tanks) will lift the crew from the surface of Mars to a low orbit about Mars. It will leave on the surface a descent module of 46 tons with initial terminal propellant and tanks. The descent module will leave in orbit an escape (or kick) module, with a dry mass (i.e., module mass without propellant and tanks) of 3 tons. The propellant and tank mass will be a function of the velocity change ΔV required for rendezvous with the MTV. The MTV will be used by the crew during the transit from the Earth to Mars and back. An aeroshell, used to decelerate the escape module to orbital speeds and to slow the descent and ascent modules, is included, and its mass is increased proportionally to include the escape modules tankage and propellant. A summary of the masses used for the nuclear thermal rocket (NTR) missions can be found in the Appendix.

Bimodal nuclear thermal propulsion will be used for the dash and ascent to the Mars transfer vehicle. The engines will also provide power for the stay on the surface. Each NTR engine is assumed to have a total thrust of 15,000 lb and a mass of 3.4 tons. For an all-NTR mission five NTR engines are originally on the MTV. Three of the engines remain on the MTV during the dash. One engine is on the ascent (from Mars) portion of the dash module. The other engine on the dash module is used to provide a total of two engines during the dash propulsive maneuver and the descent to Mars. It is assumed that all five engines can be used leaving low Earth orbit. Note that the location of the engines, including radiation shields, has not been considered in detail, but a satisfactory layout should be feasible so that they can be distributed throughout the MTV, the lander dash (or descent) module, and the lander ascent module. The radiation shield locations must also accommodate requirements for crew exposure while they exit and enter the lander during surface operations.

Propellant Mass

Although the propellant mass can be readily calculated from the ΔV and the engine specific impulse, we must also include the mass of the tanks (with thermal insulation) and a reserve propellant mass. If m_{prop} is the mass of propellant required without reserve, we will take $(1 + f_R)\, m_{\text{prop}}$ to include the reserve propellant mass. For the present analysis we took f_R to be 0.02, which neglects boil-off.[10] The mass of the tanks was taken to be proportional to the total propellant mass, as $f_T\,(1 + f_R)\, m_{\text{prop}}$. The fraction f_T was taken to be 0.03 and includes only the structural mass.[11] Other mass contributions such as a thermal insulation or a cryogenic system are not included. Using these additional masses associated with the propellants, we can find the propellant mass from

$$m_{\text{prop}} = \frac{(MR - 1) m_{\text{dry}}}{[1 - (f_T + f_R + f_T f_R)(MR - 1)]} \quad (4)$$

where m_{prop} is the propellant used, m_{dry} does not include the tanks or reserve propellant, and

$$MR = \exp(\Delta V / g_0 I_{\text{sp}}) \quad (5)$$

Trans-Mars Injection Propulsion Systems

Three propulsion systems will be considered, including chemical, nuclear thermal, and gas core nuclear. The main task will be to inject the spacecraft onto the Hohmann trajectory or the Crocco free-return trajectory.

The chemical propulsion system will be characterized by an advanced liquid-oxygen/liquid hydrogen engine. The vacuum specific impulse will be taken to be an optimistic 475 s, and the engine mass will be set to 3.2 tons (Ref. 12). Nuclear thermal will be assumed to have a vacuum specific impulse of 950 s (Ref. 13). Each engine will have a mass of 3.4 tons, and five engines will be used. Two will be used for the dash portion, and all five will be used for the injection to Mars. The gas core rocket will be taken to have a vacuum specific impulse of 1900 s and a mass of 34 t (Ref. 14). The values for the specific impulse were chosen to be exactly a factor of two apart and close to realistic values after some technological improvements.

A summary of the method used to calculate the masses is presented in the Appendix, and a complete mass breakdown for the NTR missions is included in Table A1 of the Appendix.

IMLEO Comparison

From the ΔV calculations for the various trajectories and engines, the IMLEO estimates can now be determined and are shown in Fig. 2 and Table 1. The one-year chemical rocket mission was not feasible with one stage at Trans-Mars Injection (TMI) and is not

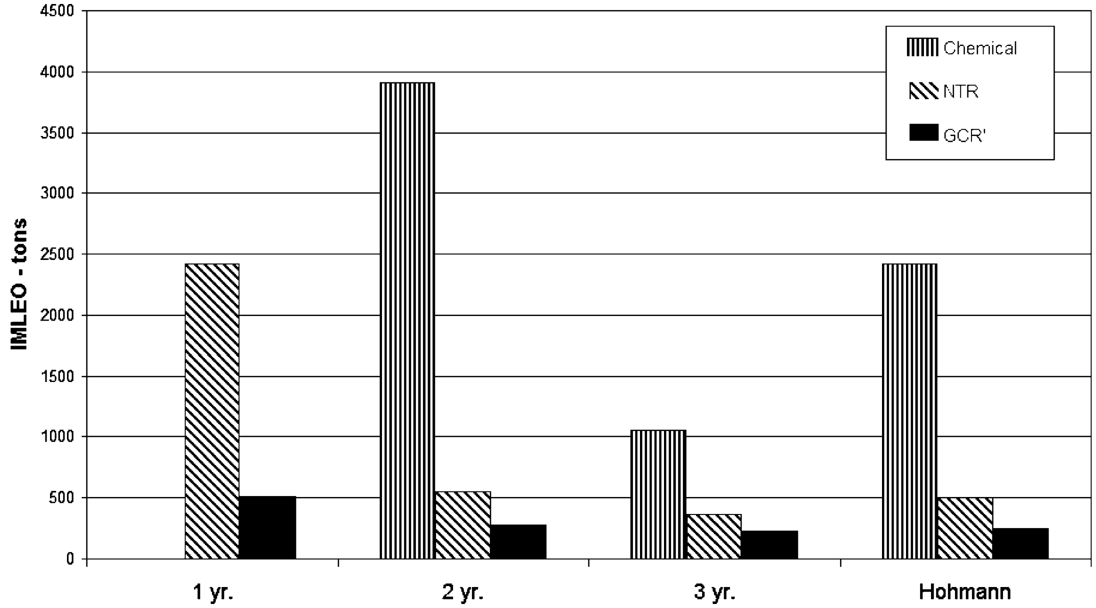

Fig. 2 Mass comparisons for Crocco missions.

Table 1 Mission IMLEO for propulsion systems and mission trajectories

Propulsion system	Mission trajectory	IMLEO, tons
Chemical	2-yr Crocco	3912
Chemical	3-yr Crocco	1056
Chemical	Hohmann	2426
NTR	1-yr Crocco	2419
NTR	2-yr Crocco	552
NTR	3-yr Crocco	364
NTR	Hohmann	504
Gas core rocket	1-yr Crocco	513
Gas core rocket	2-yr Crocco	271
Gas core rocket	3-yr Crocco	226
Gas core rocket	Hohmann	243

included in the comparison presented in Fig. 2 or summarized in Table 1.

Figure 2 illustrates that the choice of trajectory has a dramatic effect on the IMLEO required for chemical missions. For a nuclear thermal rocket, the higher specific impulse reduces the difference in IMLEO for different choices in trajectory, whereas the gas core rocket shows even less dependence on the choice of trajectory.

Although a comparison of the results of other studies cannot be used to determine the accuracy of this study, a comparison will give some idea how much the mass could be reduced if the missions were optimized. For example, the dash/swingby mission Donahue[2,15] estimated the IMLEO to be 275 tons using nuclear thermal propulsion. The IMLEO is 350 tons for the two-launch Mars direct chemical mission of Baker and Zubrin,[10] while Rauwolf and Pelaccio[11] for a one-year chemical mission got 3550 tons. Finally Howe et al.[14] have estimated 186 tons for a mission using a gas core nuclear rocket. The predictions in this analysis are higher than Refs. 2 and 12–15, but no attempt has been made here to optimize the results in this study. The comparison with previous studies indicates that the relative masses in Fig. 2 are accurate. The comparison also shows that there could be significant reductions in the IMLEO, and these would primarily benefit a chemical propulsion mission. An examination of Fig. 2 and Table 1 indicates that the total mission speed change must not substantially exceed the exhaust velocity (i.e., the product of the specific impulse and the acceleration of gravity) and that the greatest improvement in IMLEO results when the exhaust velocity reaches values around the total mission change in speed.

Mission Scenario Comparison

Many options must be considered before a mission scenario can be chosen. Table 1 and Fig. 2 show that a nuclear thermal rocket employing a 3/2-year Crocco free-return trajectory with the Donahue dash segment is an excellent choice. The hyperbolic rendezvous required for the return to Earth is obviously a high-risk operation, which must be completed if the crew is to return safely. Yet it is not only important to consider cost (IMLEO) and risk, but also science return from the missions should be included. A measure of the science return is the time spent on Mars, and here the dash mission appears to have a serious disadvantage because the time on Mars is relatively short in a dash mission. But the disadvantage applies only to the initial missions. Subsequent missions can make use of previously launched habitats. Stay times can then range up to one year, and the return trips will be shorter than the first mission. Previously launched habitats can have the consumables replenished on flybys past Earth, and their orbits can be adjusted. Hence the infrastructure for continuous exploration missions to Mars can be incrementally constructed.

Nuclear thermal rockets provide a second major advantage over other propulsion systems as they can provide power in the form of bimodal nuclear propulsion. Chemical rockets, or even gas core nuclear rockets, must transport a power source to the surface of Mars, and this will add to the mass, which is precisely the reason the comparisons included a nuclear thermal rocket for the dash portion of the mission for all three types of propulsion systems considered. The comparison would have been confused by the added mass for surface power and adds to the suitability of nuclear thermal rockets for the initial piloted missions. Their exhaust velocity not only matches the mission ΔV but also provides power for the mission.

Other important mission considerations include the total mission time. On long missions traversing interplanetary space, the crew will be subject to cosmic rays and occasional solar flares. The shielding required for the cosmic rays consisting of heavy nuclei can result in a substantial mass for the radiation shields, but the shield mass can be reduced if the transit times are decreased, which makes the one-year mission studies of Rauwolf and Pelaccio[11] of considerable importance.

There is yet another point that needs to be considered. The final design decision on mission scenario must include a risk vs cost analysis along with a technology readiness examination of the major subsystems. A risk analysis can then be performed, and a risk mitigation program, to retire all major risk elements, can be completed before moving to the preliminary design. Our experience from the Space Exploration Initiative clearly demonstrates that high-cost, low-risk missions are not acceptable. Contrast this with the example of the Apollo program. Three scenarios were examined: 1) direct launch from the Earth to the surface of the moon, 2) rendezvous in Earth orbit and then a direct flight to the surface of the moon, and 3) rendezvous in lunar orbit. Of the three scenarios, lunar orbit rendezvous had the lowest cost, but the greatest risk, yet it was chosen.[16] Of the 12 manned Apollo missions (including Apollo 1 and Apollo 7) two failed to complete their objectives, and one resulted in the loss of the crew, but there were no failures as a result of the added risk of lunar orbit rendezvous. The fact that lunar orbit rendezvous resulted in no failures can be attributed to the fact that the risk of lunar orbit rendezvous was recognized and therefore minimized before the missions were flown.

Hence, for the 12 Apollo missions the mission success rate was 83% with a 92% chance of the crew returning safely. Yet the popular conception is that the Apollo program was a success. Clearly the choice of lunar orbit rendezvous was the key to the success even at a significantly higher risk to the crew. The same lesson applies to piloted Mars missions: if we do not decrease the cost by accepting an increased risk, then we might not go at all.

Conclusions

Several trajectories were examined in the form of Crocco free-return trajectories. These trajectories provide a method for comparing the effect of propulsion system choice. Lower-energy trajectories will have less effect on the initial mass in low Earth orbit than higher-energy trajectories. Rough estimates on the effect of trajectory choice on propulsion requirements indicate that the total mission speed change must not substantially exceed the exhaust velocity (i.e., the product of the specific impulse and the acceleration of gravity). The results clearly demonstrate that a nuclear thermal rocket most closely matches this requirement for Mars missions with trip times of two or more years. Nuclear thermal rockets also provide a ready source for power during the transit to and from Mars and during the stay on Mars, and since test articles were developed in the late 1960s the technological development risk for nuclear thermal rockets is small when compared to the gas core rocket. Hence, for the propulsion systems considered in this study nuclear thermal rockets appear to be the most suitable propulsion system for the first piloted Mars missions.

The dash mission presents an excellent example of risk and cost considerations and the influence of trajectory on mission cost. Here the choice of trajectory is not as clear as the choice was for the propulsion system. The dash mission needs to be expanded to include methods to reduce the risk to the crew. For example, the use of redundant systems or the use of propellant reserve in the habitat to complete the rendezvous can reduce the risk. Redundant systems could include the ability to wait on Mars for a second habitat to arrive. Clearly, though, the cost must be reduced from the piloted Mars mission proposals of the early 1990s, and, at this time, increasing the technological or mission risk is the only approach that will reduce the cost to manageable levels.

Appendix: Mass Breakdown for NTR Missions

A detailed mass breakdown is presented in Table A1 using nuclear thermal propulsion. At launch from low Earth orbit, there are five NTR engines each 3.4 tons for a total of 17 tons. There are three basic mission portions: 1) escape from Mars, 2) additional mass for the dash, and 3) the trans-Mars injection. In the dash missions 3 tons are left in a 600-km orbit about Mars to complete the hyperbolic rendezvous. The Crocco missions are all dash missions. The Hohmann mission is not and, of course, must include habitat maneuvers into and out of orbit at Mars.

Table A1 Mass breakdown for NTR Hohmann and Crocco free-return trajectories[9]

	Mass, tons			
Component	Standard Hohmann	3-yr Crocco	2-yr Crocco	1-yr Crocco
Engines	17	17	17	17
Mars escape				
Orbital escape (kick)	0	3	3	3
Tanks (additional)	3	0	0	0
Propellant	89	6	15	11
Ascent	16	16	16	16
Descent	46	46	46	46
Aeroshell	0	19	21	20
Dash				
Dry	0	4	4	4
Tanks	0	1	1	1
Propellant	0	18	44	33
Mars injection				
Dry	246	170	208	192
Tanks	7	6	10	64
Propellant	258	195	343	2226
Total IMLEO	504	365	551	2418

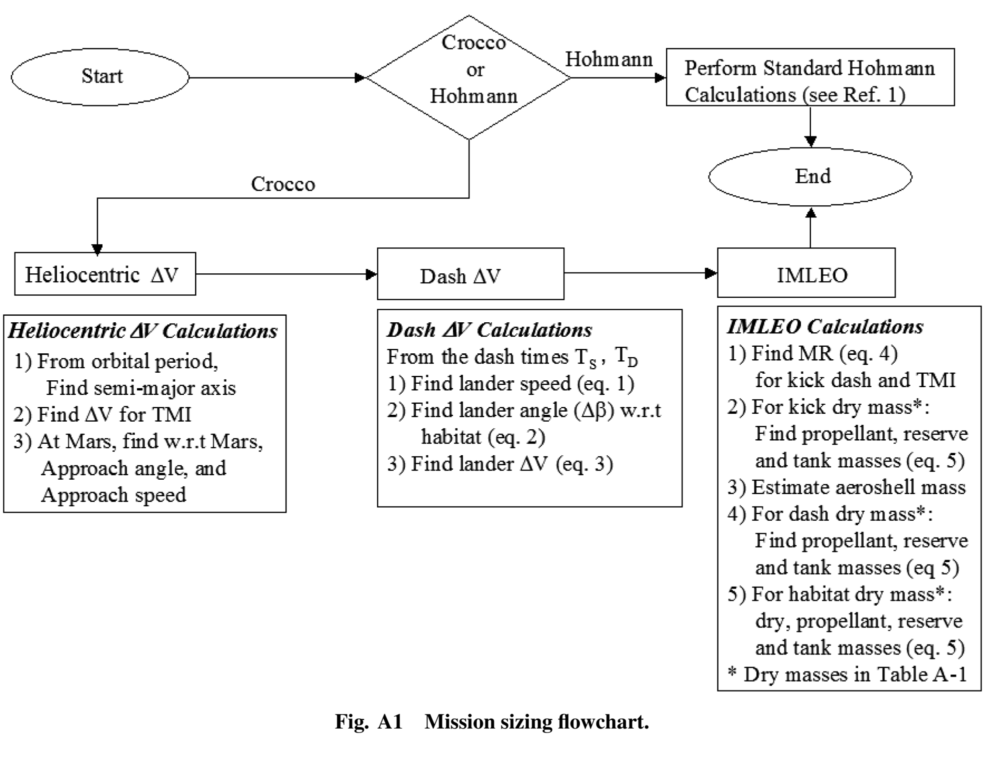

Fig. A1 Mission sizing flowchart.

A flowchart for performing the calculations is presented in Fig. A1 with sufficient details to complete the calculations. Inputs to the calculations include the Crocco free-return orbital period and the dry masses. Table A1 lists the dry masses. The aeroshell mass was estimated using[2,9]

$$M_{\text{aeroshell}} = \tfrac{17}{66}(66 + M_{\text{kp}} + M_{\text{kt}}) \qquad \text{(A1)}$$

where $M_{\text{aeroshell}}$ is the aeroshell mass in tons, M_{kp} is the total mass of propellant (including reserve) for the kick stage in tons, and M_{kt} is the tank mass for the kick stage in tons. The 66 tons in Eq. (A1) use the dry masses for the dash descent, dash ascent, and kick stage dry masses, including the engines. Hohmann missions are estimated using Ref. 1 and the dry masses in Table A1. The calculations for the Crocco free-return missions are based on the orbital period only. For the one-year mission, the aphelion is the orbit of Mars. For all of the other missions, the perihelion is at Earth. The speed changes required for TMI and the dash and hyperbolic rendezvous at Mars can now be found using well-known principles from celestial mechanics. Equations (1–3) are used to find the dash segment speed changes. Next the initial masses, using the engine specific impulse, can be calculated from Eqs. (4) and (5), and finally the IMLEO can found by working back from the hyperbolic rendezvous at Mars to the TMI burn.

Acknowledgment

The help and advice of Russell Joyner at Pratt and Whitney Space Propulsion are greatly appreciated. His suggestions were invaluable in increasing both the accuracy and clarity of the work just described.

References

[1]Von Braun, W., *The Mars Project*, Univ. of Illinois Press, Urbana, IL, 1953 (reprinted in 1991), pp. 38–64.

[2]Donahue, B., "Nuclear and Thermal Propulsion Vehicle Design for the Mars Flyby with Surface Exploration," AIAA Paper 91-2561, June 1991.

[3]Zubrin, R. M., *The Case for Mars*, Simon and Schuster, New York, 1996, pp. 75–84.

[4]Zubrin, R. M., Baker, D., and Gwynne, O., "Mars Direct: A Simple, Robust, and Cost-Effective Architecture for the Space Exploration Initiative," AIAA Paper 91-0326, Jan. 1991.

[5]Waldberg, G., "How Shall We Go to Mars? A Review of Mission Scenarios," *Journal of Spacecraft and Rockets*, Vol. 30, No. 2, 1993, pp. 129–139.

[6]Crocco, G. A., "One Year Exploration Trip Earth-Mars-Venus-Earth," *Rendiconti del VII Congresso Internanzionale Astronautico, Associazione Italiana Razzi (Proceedings of the 7th International Astronautical Federation)*, 1956, pp. 227–252.

[7]Ruppe, H. O., "Interplanetary Flight," *Handbook of Astronautical Engineering*, edited by H. H. Koelle, McGraw–Hill, New York, 1961, pp. 9.32–9.44.

[8]Byrnes, D. V., Longuski, J. M., and Aldrin, B., "Cycler Orbits Between Earth and Mars," *Journal of Spacecraft and Rockets*, Vol. 30, No. 3, 1993, pp. 334–336.

[9]Cassenti, B. N., and Bass, R. W., "High Risk Low Cost Mars Mission Scenarios," *AIP Conference Proceedings 420, Space Technology and Applications International Forum*, edited by M. El-Genk, Springer-Verlag, New York, 1998, pp. 1125–1130.

[10]Baker, D. A., and Zubrin, R. M., "Combining Near-Term Technologies to Achieve a Two-Launch Manned Mars Mission," AIAA Paper 90-1896, July 1990.

[11]Rauwolf, G. A., and Pelaccio, D. G., "Mission Performance of Emerging In-Space Propulsion Concepts for One-Year Crewed Mars Missions," AIAA Paper 2001-3374, July 2001.

[12]Isakowitz, S. J. (ed.), *International Reference Guide to Space launch Systems*, AIAA, Washington, DC, 1991, pp. 252, 253.

[13]Borowski, S. K., Dudzinski, L. A., and McGuire, M. L., "Artificial Gravity Vehicle Design Option for NASA's Human Mars Mission Using Bimodal NTR Propulsion," AIAA Paper 99-2545, June 1999.

[14]Howe, S. D., DeVolder, B., Thode, L., and Zerkle, D., "Reducing the Risk to Mars: The Gas Core Nuclear Rocket," *AIP Conference Proceedings 420, Space Technology and Applications International Forum—1998*, edited by M. El-Genk, Springer-Verlag, New York, 1998, pp. 1138–1143.

[15]Donahue, B., "Mars Ascent-Stage Design Utilizing Nuclear Propulsion," *Journal of Spacecraft and Rockets*, Vol. 32, No. 3, 1995, pp. 552–558.

[16]Murray, C., and Cox, C. B., *Apollo: The Race to the Moon*, Simon and Schuster, New York, 1989, pp. 115–120.

C. Kluever
Associate Editor