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Flexible Piloted Mars Missions Using Continuous Electric Propulsion

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The potential of continuous propulsion systems for future space is outlined and compared to impulsive propulsion (chemical and nuclear–thermal). Although the results are related to piloted Mars missions, some of the stated issues hold true for a broad range of space missions with high-velocity increments, for example, sample return missions. It is demonstrated that for piloted Mars missions the use of impulsive propulsion can lead to very inflexible missions with a long total mission duration, whereas continuous electric propulsion does not only guarantee short total mission durations with moderate masses but also a high degree of flexibility. This can be achieved with a continuous electric propulsion system that has a thrust level of 100 N and a specific impulse of 3000 s, which is not too futuristic for piloted Mars missions in 2030 and beyond.

Nomenclature

F	=	thrust
I_{sp}	=	specific impulse
k	=	tank mass fraction
m	=	mass
P_e	=	electric input power
v_e	=	atmospheric entry velocity
v_∞	=	relative velocity
Δv	=	velocity increment
η_{th}	=	thruster efficiency
χ	=	throttle factor

Subscripts

add	=	additional
max	=	maximum
pay	=	payload
pro	=	propellant
SPP	=	structure, power, and propulsion
0	=	initial value

Introduction

PILOTED Mars missions have been studied for years.[1] From these studies, major objectives and boundary constraints can be derived for the design of such missions, which may differ from common interplanetary probe missions. These objectives are a short transfer time, a moderate mass, a short mission duration, and a high flexibility. These missions also differ from common probe missions in regard to the design philosophy. Piloted Mars missions are not designed as single missions, but within a Human-Mars-Exploration Program that typically consists of three piloted missions at intervals of about two years each (time span of recurring relative constellations of Mars and Earth). All three single missions should not be planned separately, but should have a common design with regard to the mission itself, the spacecraft, and the subsystems. In the following text, this common design approach will be called a program principle. Such a principle could also be applied within each single piloted mission, which typically consists of different vehicles. This can be achieved by using similar subsystems for example, the same propulsion systems for different spacecraft, a strategy that is also used in the NASA Design Reference Mission (NASA DRM).[2] Current concepts such as NASA's DRM and the California Institute of Technology Mars Society Mission[3] use impulsive propulsion systems (chemical or nuclear–thermal propulsion) and ballistic trajectories. Unfortunately, with these propulsion concepts one cannot fulfil all four objectives mentioned earlier. Hence, their primary objectives are only a moderate mass and a short transfer time. Moderate spacecraft masses with impulsive propulsion require the use of low or moderate energy trajectories that recur at intervals of about two years within a rather small launch window. Thus, a less flexible mission with practically no abort or delay options and a long total mission duration (about three years) due to the necessary long-stay time at Mars is the consequence. This is beyond any human spaceflight experience so far. At present, little attention is given to the high risk of mission failure that is involved with such nonflexible mission scenarios. In this paper, piloted Mars missions are analyzed especially with respect to a short total mission duration and flexibility but also with respect to short transfer times and moderate spacecraft masses. Here the term flexibility means that the chosen spacecraft designs guarantee a large number of mission opportunities or even enable launch possibilities for any planetary constellation and provide abort and delay options. The analyses cover impulsive- and continuous-thrust propulsion systems. It will be demonstrated that with impulsive propulsion the only feasible round-trip option is the long-stay option to be discussed, whereas the use of continuous propulsion enables very flexible missions with short total mission durations. The use of continuous electric propulsion systems with moderate to high specific impulses leads to reasonable masses and moderate flight times for long- as well as for short-stay options.

Mission Scenario

Split Mission Concept

The mission scenario in this paper is based on the NASA DRM. It is a six-person crew, surface base scenario. Like in the NASA DRM, a split mission concept with three outward vehicles [basis-infrastructure cargo vehicle (BICV), Earth return vehicle (ERV) cargo vehicle (ERVCV), and piloted outward vehicle (POV)] and one return spacecraft (ERV) was chosen (Fig. 1). The BICV delivers

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Table 1 Payload mass breakdown for BICV and different POVs

BICV		POV 350 days		POV 500 days		POV 900 days	
Components	Mass, t	Components	Mass, t	Components	Mass, t	Components	Mass, t
			Payload				
Mars ascent vehicle	10.3	Habitat/ECLSS	30.6	Habitat/ECLSS	32.4	Habitat/ECLSS	37.0
ISPP plant plus reactor	19.4						
Pressurized rover	16.0						
Two unpressurized rovers	16.0						
Three teleoperated rovers	3.6						
Additional equipment	1.0						
Total	66.3		30.6		32.4		37.0
			Descent stage				
Thrusters	1.9		1.9		1.9		1.9
Structure and additions	2.3		2.3		2.3		2.3
Propellant and tanks	14.5		6.7		7.1		8.1
Heat shield[a]	13.9		6.4		6.8		7.8
Total	32.6		17.3		18.1		20.1

[a]21% of landed payload.

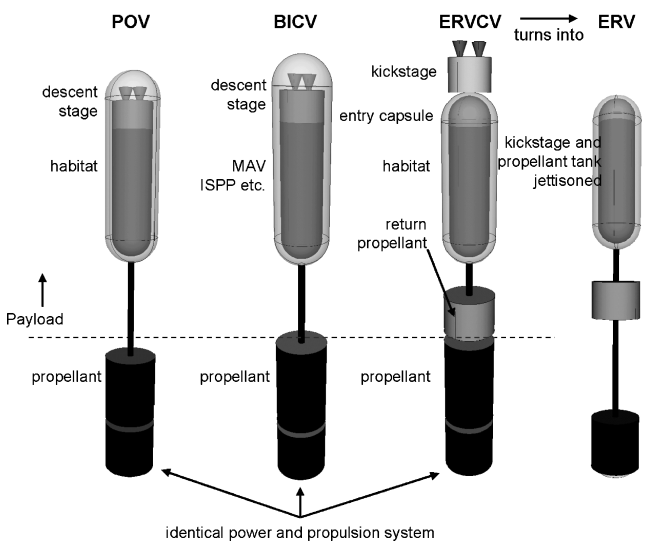
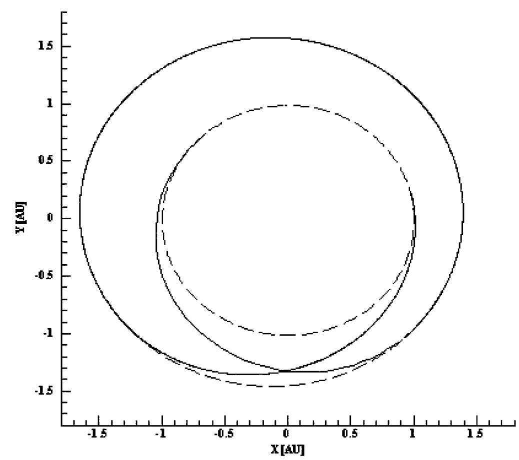
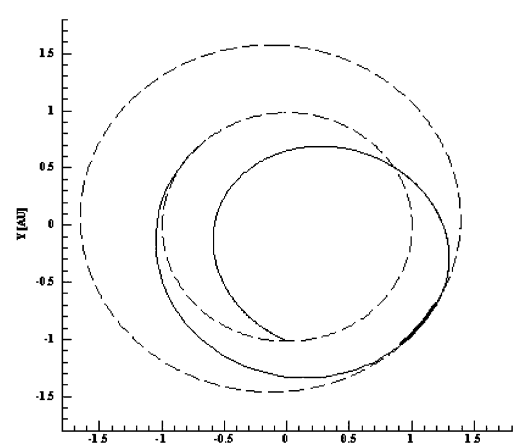


Fig. 1 Different transfer vehicles for chosen split mission concept (exemplary for mission using continuous-propulsion systems).

all components that are required for the surface base to Mars and lands them on the surface. This includes rovers, the science payload, a nonfueled Mars ascent vehicle (MAV) and an in situ propellant production plant (ISPP) including a reactor (Table 1). The ERVCV delivers a fully fueled ERV to Mars and parks it in a 24-h elliptic orbit with a pericenter of 250 km. This orbit was chosen because it results in moderate initial masses of the MAV (results of an internal study[4]) as well as a low escape Δv (221 m/s) for the ERV. Because this paper will focus on the piloted vehicles, results for the BICV and ERVCV will not be presented here. Data and results for these vehicles are given in Refs. 5 and 6. The POV delivers the crew to Mars and lands them within the transit habitat on the surface. As already stated in the NASA DRM, this approach is less risky and the most comfortable way to land because a capsule landing would force the probably deconditioned astronauts to walk several 100 m to reach the surface base. At the same time, an additional spacecraft that transports a surface habitat to Mars can be saved. The payload of the POV consists of the transit habitat including the environmental control and life support system (ECLSS) and a descent stage. At the end of the mission the crew will return within the ERV. This spacecraft is also equipped with a transfer habitat and an Earth entry capsule. A program principle should be used within the Human-Mars-Exploration Program (same scenario and spacecraft design on the subsequent missions) and also within a single piloted mission. Therefore, all transfer vehicles use the same power and propulsion system, as shown in Fig. 1, and the ERV will reuse the ERVCV's power and propulsion system.



a) Long-stay option, 1000 days total mission time

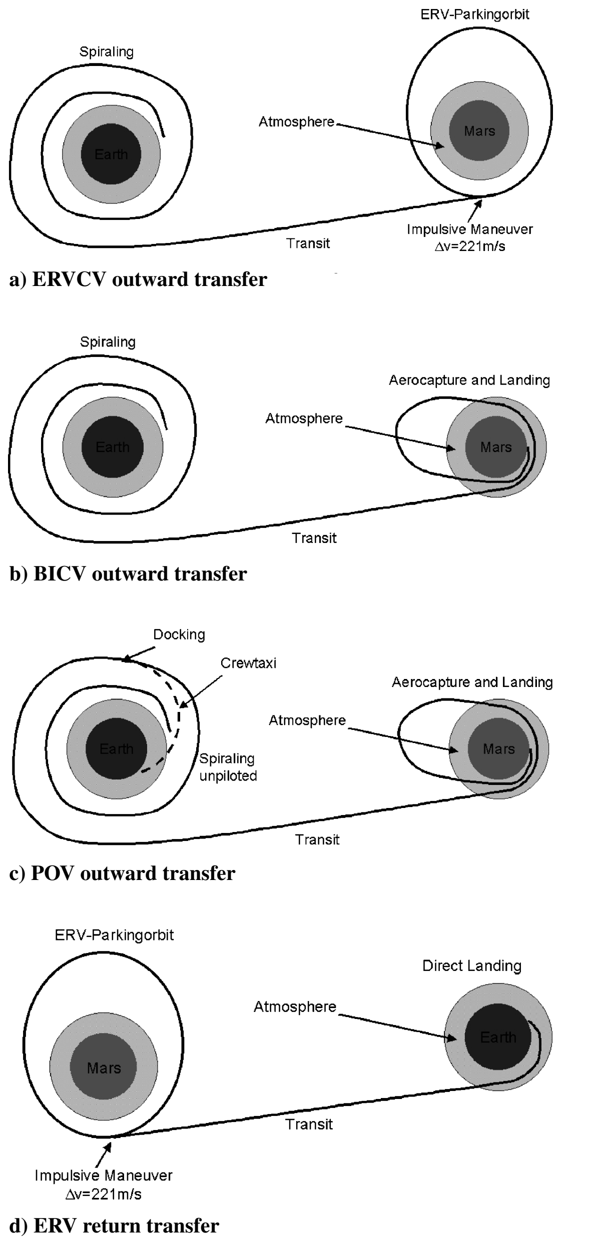


b) Short-stay option, 500 days total mission time

Fig. 2 Classical round-trip options.

Round-Trip

For every piloted mission within the Human Mars Exploration Program the cargo vehicles (BICV and ERVCV) are launched to Mars two years before their crewed counterparts (POV). As in the NASA DRM, the cargo vehicles of subsequent missions are supposed to be used as backup systems. For the piloted vehicles, there exist two classical round-trip stategies: long-stay and short-stay options. The long-stay option, as shown in Fig. 2a, is a round-trip that uses moderate energy trajectories on the outward and on the return leg, resulting in a long stay time (about two years) at Mars. The



a) ERVCV outward transfer

b) BICV outward transfer

c) POV outward transfer

d) ERV return transfer

Fig. 3 Schematic of the different transfer scenarios (exemplary for a mission using continuous propulsion systems).

short-stay option uses a moderate- or high-energy trajectory during the outward leg and a high-energy trajectory during the return leg, as shown in Fig. 2b. On such high-energy trajectories, the spacecraft crosses Earth's orbit up to a certain solar distance. Thermal and medical aspects that evolve with such types of trajectories and their influence on the performance have not been taken into account in this analysis. However, for all trajectories, a minimum solar distance was set at 0.7 astronomical units (AU), which seems to be a reasonable value. Whereas the ERV cargo vehicle uses propulsive maneuvers for the orbit injection, all other transfer spacecraft will perform aeromaneuvers before landing for mass savings (Fig. 3). Therefore the BICV and the POV require a descent stage. This stage consists of a heat shield, retrorockets, and reaction-control engines including propellant. On the return leg, the Earth entry capsule is jettisoned on arrival at Earth and will perform an aeromaneuver with direct landing. For the aeromaneuvers at Mars or at Earth, an assessment of the heliocentric transit and the atmospheric phase[7–9] yields that the maximum relative velocity should be $v_{\infty,\max} = 6$ km/s at Mars and $v_{\infty,\max} = 9.5$ km/s at Earth. For safety reasons, the POV performs an aerocapture into the ERV's parking orbit first before landing on the surface. This guarantees a safe haven and a return possibility if the crew cannot land on the Martian surface. In this case, the POV will dock with the ERV.

Table 2 Habitat mass including environmental control and life support system, different scenarios

	Habitat mass/maximum crew survival, t/days			
	Impulsive propulsion		Continuous propulsion	
Stay	Outward	Return	Outward	Return
Short	30.6/350	30.6/350	32.4/500	32.4/500
Long	37.0/900	28.0/250	37.0/900	28.0/250

Spacecraft and Subsystem Design

Analyses have been performed for impulsive- and continuous-propulsion systems and the different round-trip options. For the mass dimensioning and calculation of subsystem components, system studies have been performed at DLR, German Aerospace Center and the Institut für Raumfahrtsysteme (IRS), that led to the development of the program SAFIR.[10] This program calculates component masses, volumes, and power requirements for several subsystems based on existing data or on proposed concepts. The chosen habitat complex has an inner diameter of 6 m and the habitat's length is 17 m. The free volume per person is about 68 m^3. The corresponding mass, however, is different for different investigated mission scenarios. For the mass calculations of a habitat and an ECLSS (components and resupply), one needs to know about the operational times of the systems that are onboard the different piloted spacecraft within a single mission (POV and ERV) and within the Human-Mars-Exploration Program. The operational times and resulting subsystem masses are summarized in Tables 1 and 2. For long-stay options, the outward/surface habitat requires an operational time of about 900 days, and the corresponding mass is 37 t including spares. The operational time of the return habitat depends on the return transit time, which is between 120 and 220 days. The mass of such a habitat/ECLSS complex is about 28 t. For short-stay options and impulsive-propulsion systems, the operational time during the outward leg (plus surface stay) is about 350 days, and the habitat mass is 30.6 t. For safety reasons, the return habitat also was designed for an operational time of 350 days. Hence, both habitat complexes are practically identical in design. To achieve a high mission flexibility that also includes abort and delay options (discussed later), the operational time of the habitat complex was increased to 500 days for continuous-propulsion systems, and the habitat mass is 32.4 t. Masses for the different descent stages are also given in Table 1. Besides a return habitat, the ERV requires an Earth entry capsule, which has a mass of about 6 t.

Impulsive-Propulsion Systems

Impulsive-propulsion systems such as chemical or nuclear–thermal rockets use impulsive high-thrust maneuvers to inject spacecraft into heliocentric ballistic trajectories. On arrival at the target, a second impulsive maneuver is performed, if the relative velocity is higher than the maximum values $v_{\infty,\max}$ for safe aeromaneuvers or if a fully propulsive orbit insertion is used. Further impulsive maneuvers during the flight have not been considered in our analyses, and the scenario is, thus, a ballistic interplanetary mission. For impulsive-propulsion systems, the required velocity increment Δv can be approximated analytically. For systematic analyses, however, this can be a very time-consuming task. Hence, in this paper, the required Δv for injection into a ballistic trajectory has been calculated with the code InTrance,[11] a program developed at DLR that fuses evolutionary algorithms and neural networks to optimize low thrust trajectories. However, it also contains an option to calculate ballistic interplanetary missions and to minimize their hyperbolic excess velocities and, thus, the required Δv for injection. The required velocity increment to adjust the velocity at the target for safe aeromaneuver was calculated using

$$\Delta v_{\mathrm{adjust}} = v_{\infty} - v_{\infty,\max} \qquad (1)$$

if $v_\infty > v_{\infty,\max}$ with $v_{\infty,\max} = 6$ km/s at Mars and $v_{\infty,\max} = 9.5$ km/s at Earth; also see the "Mission Scenario" section.

Mass calculations have been performed using a mass model according to

$$m_0 = m_{\text{pay}} + m_{\text{SPP}} + (1 + k)\, m_{\text{pro}} \tag{2}$$

with m_{pay} for the payload, m_{SPP} for the structure, power, and propulsion system mass, m_{pro} for the propellant mass, and m_0 for the spacecraft's initial mass. The factor k represents the tank mass fraction and depends on the propellant and tank material. The payload mass contains the habitat/ECLSS complex and is given Table 2 for the different scenarios and piloted transfer vehicles. Three impulsive-propulsion systems have been investigated: 1) an liquid oxygen (LOX)/LH_2 system, where $m_{\text{SPP}} = 7.31$t, $I_{\text{sp}} = 460$ s, and $k = 0.04$ (which is a typical value for chosen propellant type), based on the space shuttle main engine; 2) an LOX/CH_4 system, where $m_{\text{SPP}} = 7.31$t, $I_{\text{sp}} = 387$ s, and $k = 0.03$ (which is a typical value for chosen propellant type), based on results from Refs. 2 and 4; and 3) a nuclear–thermal system with H_2, where $m_{\text{SPP}} = 11.31$t, $I_{\text{sp}} = 800$ s, and $k = 0.25$ (which is a typical value for chosen propellant type), based on the NERVA engine.[2]

For piloted Mars missions, the return leg is the major design driver. Especially for short-stay options, a good return trajectory must be found and the corresponding outward trajectory must be adjusted to it. Figure 4 shows the velocity increments (total Δv) for return trajectories (departing from the chosen ERV orbit) within a time span of 20 years, and Figs. 5 and 6 show the corresponding masses and transit times of the ERV. It can be seen that the velocity increment for long-stay options is between 1.0 and 1.5 km/s for each opportunity, resulting in masses for the ERV of about 60–70 t.

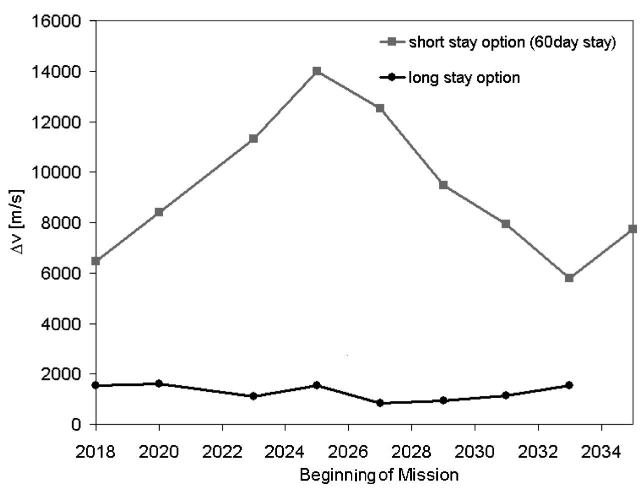


Fig. 4 Velocity increments for impulsive return trajectories launching from ERV orbit (24 h orbit, 250-km pericenter).

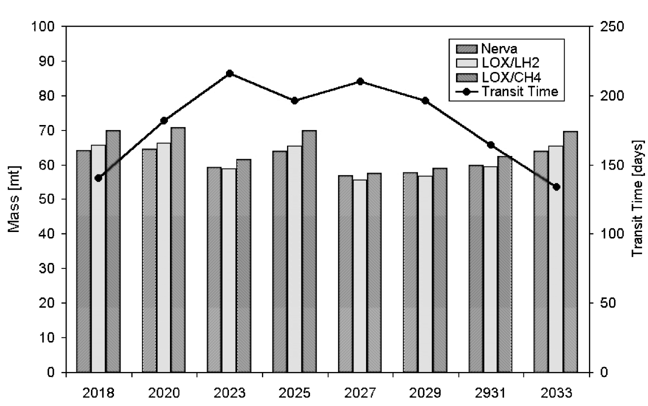


Fig. 5 Return transit times and initial masses of ERV for long-stay options with impulsive propulsion.

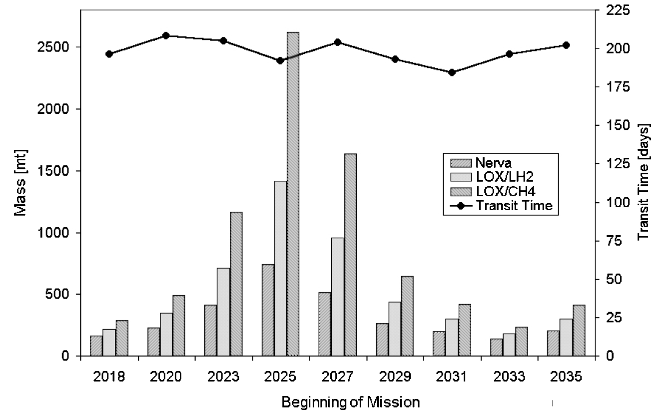


Fig. 6 Return transit times and initial masses of ERV for short-stay options with impulsive propulsion.

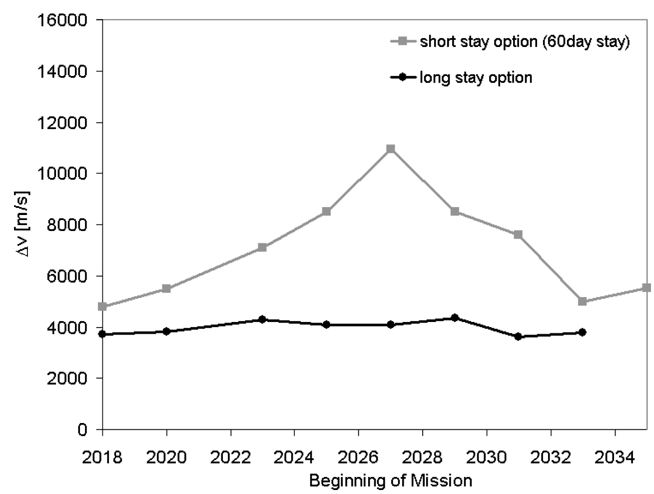


Fig. 7 Velocity increments for impulsive outward trajectories starting from 400-km circular Earth orbit.

The velocity increment for short-stay options is significantly higher (6–14 km/s) and, thus, so are the spacecraft masses. The bigger problem, however, is that the Δv for short-stay options is not only significantly higher but that it also varies significantly for the different mission opportunities. Hence, the initial masses of the ERVs vehicles in the Human-Mars-Exploration Program would be completely different and so would their designs for each single mission. To apply a program principle (same spacecraft design for every single piloted mission) at any rate, the spacecraft have to be designed for the worst case (highest Δv). For the ERV, this would lead to masses of 714 t with NERVA, 1360 t with LOX/LH_2, and 2519 t with LOX/CH_4 engines. Building such spacecraft and placing them in a Martian orbit seems unrealistic for a mission around 2030.

A similar tendency was found for the outward transfers, as can be seen in Figs. 7–9 for a departure from a 400-km circular Earth orbit. The outward Δv for long-stay options is moderate ($\approx$4 km/s) and similar for the different mission opportunities. The necessity of adjusting the outward to the return leg for short-stay options typically leads to higher velocity increments on the outward leg for short-stay options compared to long stay, and the Δv also varies between 5 and 11 km/s for the different mission opportunities. Hence, for short-stay options, the design of the POV would also be different for each mission opportunity. Even short-stay piloted Mars missions independent of a Human-Mars-Exploration Program or a program principle are problematic. If a mission is designed for the very good opportunities in 2018 or 2033 (comparatively low Δv), it has to be performed at that particular date. In case of mission delays, the chosen spacecraft designs might not be applicable for the next launch opportunity (two years later) due to the significantly different velocity increments. Hence, short-stay options with impulsive

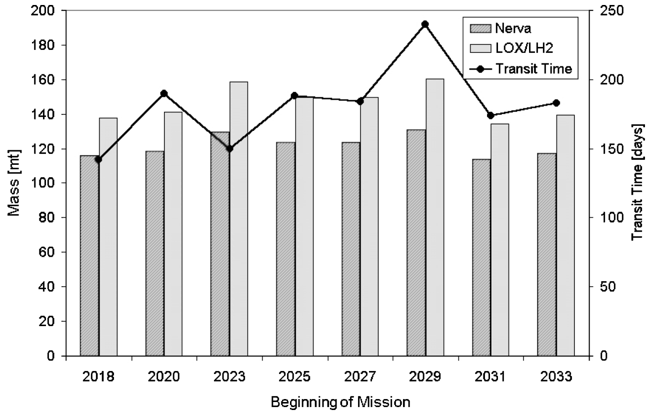


Fig. 8 Transit times and initial masses of the POV, long-stay options with impulsive propulsion.

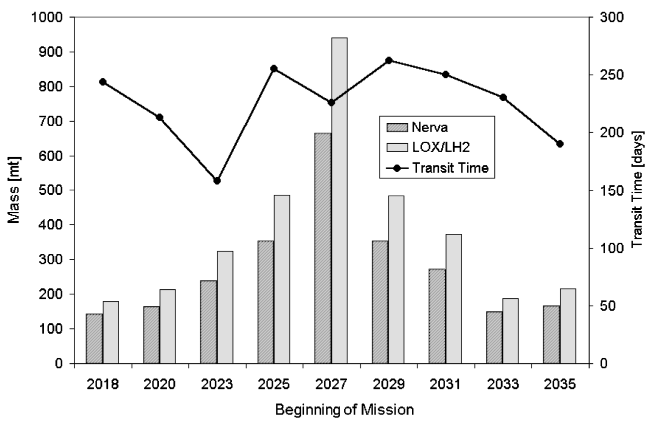


Fig. 9 Transit times and initial masses of POV, short-stay options (60-day stay) with impulsive propulsion.

propulsion do not seem to be feasible or at least are very problematic, a fact that was already mentioned in Ref. 12 and is one of the reasons why NASA chose a long-stay option. The results show that with impulsive propulsion the classic long-stay option is the only reasonable round-trip option. A program principle can be applied without great loss in performance. However, because long-stay options with their low-energy transfers are only feasible for special planetary constellations that guarantee low spacecraft masses, such missions are very inflexible. Hence, at least from the present situation, such 1000-day long-stay missions with all of the risks caused by the use of impulsive propulsion, seem to be not very favorable for the first piloted Mars missions.

Continuous Electric Propulsion Systems

The last section demonstrated that with impulsive propulsion only the objectives of minimum mass and minimum transfer time can be met, whereas the objectives of minimum mission duration and maximum flexibility would result in unreasonable mission scenarios. A propulsion concept that could meet all objectives is continuous electric propulsion.

Transport Scenario

For continuous electric propulsion systems, the transfer legs are divided into two phases, the planetocentric spiraling and the heliocentric transit phase, as shown in Fig. 3. To save the spiraling time for the astronauts, the POV starts spiraling without the crew. At the end of the spiraling phase, the crew is brought aboard with a crew taxi that docks with the POV. This crew taxi consists of a chemical kickstage and a capsule identical in design to the ERV's Earth entry capsule, as shown in Fig. 10. It is designed to bring back the crew to Earth in case of emergency during the transfer to the POV. After the crew has left the taxi, it is jettisoned. The flight time for the crew is, thus, only the time that is required for the heliocentric transit

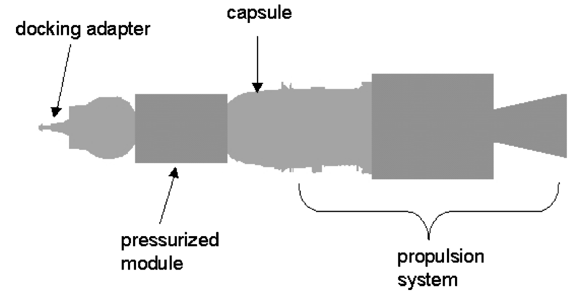


Fig. 10 Schematic of crew taxi.

(transit time). Hence, the required spiraling times and the resulting spiraling trajectory starting from a 400-km circular Earth orbit will not be presented here because these aspects are not important for the investigation in this paper. Because the Δv for the escape kick from Mars is only 221 m/s from the chosen ERV parking orbit, the ERV uses a chemical kickstage (Fig. 1) instead of spiraling out of Mars' gravity field, and it uses its electric propulsion system only during the heliocentric transit phase.

Modeling and Required Propulsion System Parameters

Because few data exist concerning the use of electric propulsion for piloted Mars missions, a systematic trajectory and mission analysis was performed. First results[13,14] indicated that for fast transfer times (4–6 months) a thruster acceleration (F/m) of about 0.3–0.7 mm/s^2 is required. Hence, for the chosen mission scenario and typical spacecraft masses in the order of 200 t, the maximum thrust should be about $F_{\max} = 100$ N. With continuous electric propulsion, optimal propulsion parameters exist, especially an optimal I_{sp}, for different planetary missions and spacecraft designs that lead to minimum masses and minimum transit times.[15] In Refs. 13 and 14 the most promising range for the specific impulse I_{sp} was between 2000 and 6000 s. In Ref. 6, it is shown that the adequate I_{sp} for piloted Mars missions and the chosen mission concept is about 3000 s. A further important propulsion system parameter is the thruster efficiency η_{th}. In general, the efficiency should be as high as possible ($\eta_{th} \gtrsim 30\%$) to achieve moderate initial spacecraft masses. Promising candidates that could reach or that have already reached these required propulsion system parameters are Arcjets or two-staged electrothermal thrusters such as the ATTILA engine,[16] and Hall-ion thrusters,[17,18] as well as self-field or applied-field magnetoplasmadynamic thrusters. Electrothermal resistojets have to be excluded because they only achieve a specific impulse of about 800 s. Ion thrusters with their high I_{sp} also have not been considered because, assuming present and near-term technology, the thrust density (thrust per diameter) and the resulting thrust per mass appear to be too low for piloted Mars missions.

The spacecraft mass calculation is based on Eq. (2). The payload masses for the different scenarios can be taken from Table 2. A $k = 0.2$ was chosen, which is a typical tank mass fraction factor for such preliminary analyses. The major part of m_{SPP} is related to the electric power systems. Multimegawatt power systems that are discussed in the literature have power specific masses that typically range from 5 to 15 kg/kWe (Refs. 19–21). The most promising concepts that could reach such power specific masses are nuclear dynamic systems with Brayton conversion or lightweight photovoltaic systems such as the stretched lens array/square rigger concept.[22] When a thruster efficiency of about $\eta_{th} = 37\%$ is assumed, the required power level for a 100-N, 3000-s thruster cluster is 4 MWe. With a power specific mass of 10 kg/kWe the resulting power supply mass is about 40 t. For the analyses, a $m_{SPP} = 41.6$ t, which includes the main power supply system and power storage devices in case of system malfunctions. The maximum amount of propellant mass was set to $m_{pro,\max} = 50$ t.

Optimization

For continuous electric propulsion, the heliocentric trajectories cannot be approximated as ellipses but have to be calculated numerically. The thrust of a continuous-propulsion system was modeled according to

$$\boldsymbol{F} = F \cdot \boldsymbol{f} = \chi \cdot F_{\max} \cdot \boldsymbol{f} \tag{3}$$

with $F_{\max}$ being the maximum available thrust, χ the throttle factor with $0 \leq \chi \leq 1$, and $\boldsymbol{f}$ the direction of the thrust vector, which is defined by two thrust angles α and β. Together with the throttle factor, these angles are the control values of the optimization process. A definition of the thrust angles may be found in Refs. 11 and 23. The final conditions are met as soon as the spacecraft reaches the desired target conditions: 1) spacecraft within the target planet's sphere of influence and 2) relative velocity $\leq v_{\infty,\max}$.

The analyses showed one general difference between impulsive- and continuous-propulsion systems: With impulsive propulsion, the departure constellation and chosen boundary constraints dictate a Δv for the transfer. With continuous propulsion systems, it is possible to reduce the required velocity increment (propellant mass) for any departure constellation by allowing longer flight times. Hence, for the chosen $v_{\infty,\max}$ and a minimum solar distance, a continuum of solutions exist, and one has to solve an optimization problem. This problem was solved by limiting the available amount of propellant and minimizing the corresponding transfer time. The obtained results are so-called Pareto optima. This is a common and typically the preferred approach for such optimization problem because it results in a better convergence behavior as compared to the method of using a cost function with weighting factors for transfer time and propellant mass. For verification reasons, the heliocentric trajectory simulation and optimization within this paper was done with two completely different working optimizers: the commercially available software GESOP[24] with the nonlinear programming solver SNOPT[25] and the software InTrance.[11] With both alternatives, similar results have been achieved. For the spiraling phase at Earth, a simple integration of the equations of motion was used with a constant thrust in velocity direction until escape velocity was reached.

Results

Figure 11 shows the return flight times of the Earth return vehicle over 20 years at a step width of 30 days, exemplary for a thruster cluster with 100 N of thrust and a specific impulse of 3000 s. The departure dates are given in modified Julian dates (MJD[26]) as well as in years (A.D.). It can be seen that the transit times can vary significantly (but quasi periodically with the 26-month cycle) depending on the planetary constellation. However, it also clearly demonstrates that fast transfer times can be achieved with continuous electric propulsion. (Also see Ref. 13 for a more detailed discussion of these results.) Figure 12 shows possible round-trip trajectories, and Table 3 summarizes the corresponding transit times and masses. For long-stay options, the transit times are 125 days on both legs, which is even better than in the impulsive case. The ERV masses are 120 t and are, thus, higher compared to impulsive propulsion. This is mainly due to the significantly higher m_{SPP}. For short-stay options, the transit times during the outward legs are 145 days, and on the return leg they increase with increasing stay time from 270 to 510 days. Hence, the objectives minimum transit time (at least during the outward leg), minimum mass, and minimum mission duration can be met with continuous electric propulsion. The analyses furthermore yield a very important aspect: It was found that the variation of the corresponding initial masses of the ERV is only between 98 and 140 t over the investigated 20-year scenario, which is much less of variance than for the impulsive case (LOX/LH_2, 50–1360 t). Hence, with a 140-t ERV, it would be possible to return from Mars to Earth at any time. The same tendency was found for the piloted outward vehicle (maximum transit mass 156 t). A program principle is, thus, always applicable within the Human-Mars-Exploration Program, and launch delays are also not as problematic as with impulsive-propulsion systems. Also, during a piloted mission, it is possible to return from Mars to Earth with the 140-t ERV design in case of mission aborts at Mars or delays, as Table 3 demonstrates. Hence, the objective high flexibility can be met as well. With the possibility of returning to Earth at any time with the same spacecraft design, the use of continuous electric propulsion is a very interesting alternative for long-stay options as well because these would be much more flexible compared to the impulsive case and, thus, probably be less risky. The results also show that, no matter what type of trajectory (low-, moderate-, or high-energy trajectory) is used, it is possible to go to Mars (POV) or back to Earth (ERV) with one type of propulsion system. This is also true for the cargo vehicles, as shown in Refs. 5 and 6. Hence, all spacecraft within a single mission can use the same propulsion and, thus, also the same power system. Such subsystems would have to be developed only once but could be used for several different applications within one mission.

Table 3 Round-trips for different short-stay options and long-stay option using 100-N, 3000 s, $\eta_{\mathrm{th}} = 37\%$ continuous electric propulsion system

Stay time days	Transit time, days		Total mission duration, days	Mass, t	
	Outward	Return		Outward	Return
30	145	270	445	156	140
60	145	320	525	156	140
90	145	410	645	156	140
120	145	510	775	156	140
Long stay	125	125	933	150	120

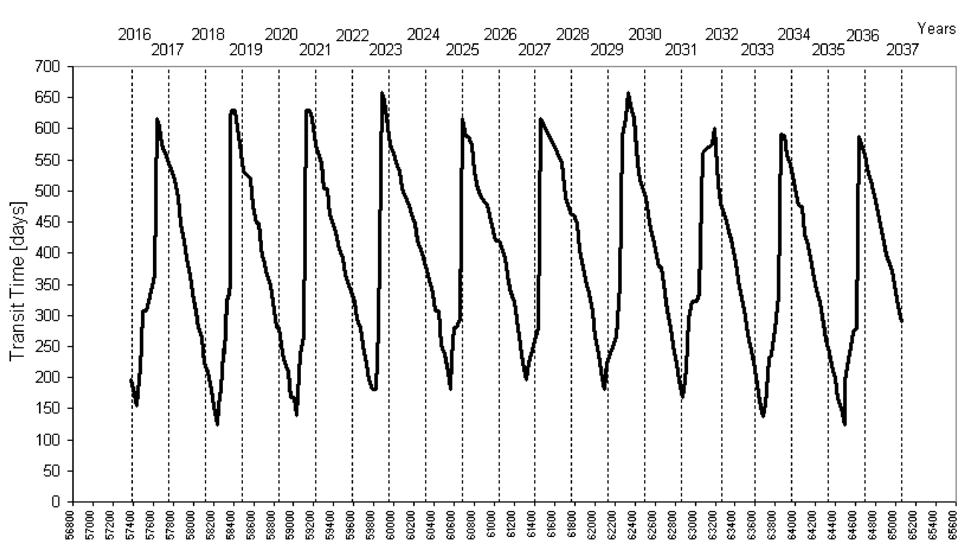
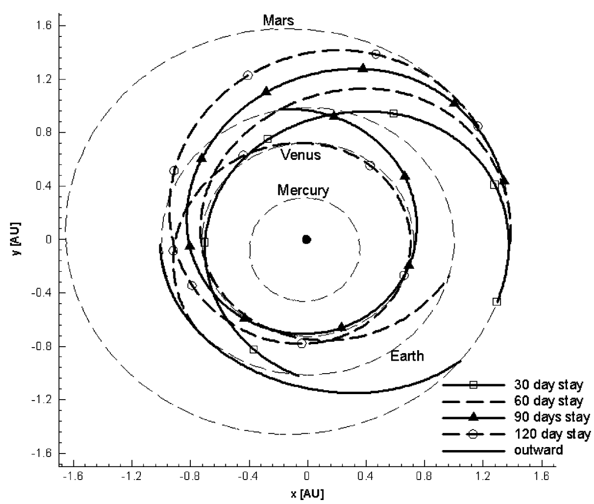


Fig. 11 Return transit times for different departures at Mars for 100-N, 3000-s, $\eta_{\mathrm{th}} = 37\%$ continuous electric propulsion system.

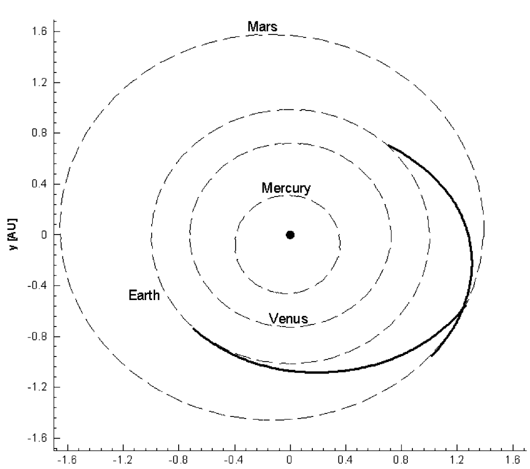
Table 4 Aborts with piloted outward vehicle, different times for 100-N, 2929-s, $\eta_{th} = 58\%$ continuous electric propulsion system and short-stay mission scenario

	Original scenario		Abort scenario				
					Abort at		
Performance	Earth spiraling	Transit	Earth spiraling	Transit	25%	50%	100%
Time, days	113	132	137	145	122[a]	286[a]	283[a]
m_{pay}, t	84.5	55.9	107.8[b]	78.4[b]	55.9[c]	55.9[c]	55.9[c]
m_{pro}, t	34.1	33.6	42.8	35.4	38.0	36.1	22.5
m_0, t	158.4	118.8	188.5	148.1	137.3[d]	126.4[d]	112.8[d]

[a]Return flight time after abort.
[b]Includes abort propellant.
[c]Includes habitat, descent stage, Earth entry capsule.
[d]Spacecraft mass at moment of abort.



a) Various short-stay options



b) Long-stay option

Fig. 12 Round-trip for different stay times at Mars and 100-N, 3000-s, $\eta_{th} = 37\%$ continuous electric propulsion system.

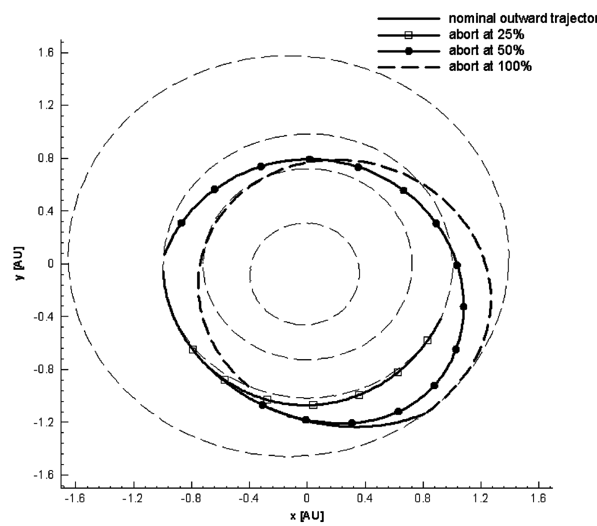


Fig. 13 Abort trajectories during piloted outward transit using 100-N, 2929-s, $\eta_{th} = 58\%$ continuous electric propulsion system.

Another interesting aspect was found while analyzing the performance of continuous electric propulsion systems: the possibility of aborting a mission during the outward transfer. Such an option is not intended in any common mission scenario that uses impulsive propulsion because the required amount of abort propellant would significantly increase the POV's initial mass. In this paper, aborts during the outward transit have been analyzed for different times during the outward transfer: 25, 50, and 100% (in percent of the total transfer time toward Mars). For such a scenario, the POV requires an Earth entry capsule, similar to the one of the ERV. This can be solved quite easily because the crew taxi already contains such a capsule that should be used in case of a mission abort before docking with the POV. In the original design (without abort possibility during the outward leg), this capsule is jettisoned after the crew's transfer to the POV. In the abort design, the capsule is kept and can, thus, be used for the Earth entry in case of mission abort during the outward leg. For the abort scenario, it is assumed that the complete spacecraft is going back to Earth, but one could also consider to jettison parts of the POV, for example, the descent stage, before the abort to reduce the spacecraft mass. In the calculations, the maximum amount of additional propellant was set to $m_{pro,add} \leq 22$ t. Examples for abort trajectories are shown in Fig. 13, and the performance is summarized in Table 4, exemplary for a 100-N, 2929-s, $\eta_{th} = 58\%$ continuous electric propulsion system and a short-stay mission scenario. The results show that the abort design of the POV leads to an increase in initial mass of 30 t ($158.4 \rightarrow 188.5$ t) and an increase in outward transit time of 13 days compared to the original design. However, this increase in transit time and in initial mass is quite moderate, and no additional effort, for example, an extra assembly launch compared to original design, is required as shown in Ref. 5. The results further show that aborts with the POV are theoretically possible at any time during the outward transit without great loss in performance. The return flight times are 122 days for an abort at 25%, 286 days at 50% and 283 days at 100%. In the worst case (abort at 100%), the total flight time for the crew is 428 days, which is still shorter than the chosen operational time of the systems aboard (500 days). Hence, no redesign of the habitat complex is required.

Summary

Major design objectives for piloted Mars missions are a short transfer time and mission duration, a moderate mass, and a high flexibility. In general, piloted Mars missions are planned within a Human-Mars-Exploration Program that typically consists of three single piloted missions at an interval of two years. Within that program, a common design approach, here called a program principle, is used in a way that all single piloted missions within that program should have a similar mission scenario and a similar spacecraft and subsystem design. It was found that with impulsive-propulsion systems (chemical and nuclear–thermal) only the objectives minimum mass and minimum transfer time can be met. The only reasonable round-trip option is the classical but very inflexible long-stay option. Short-stay options that use high-energy return trajectories to reduce the total mission duration are problematic because the large Δv requirements lead to huge initial masses. They also vary significantly for the different mission opportunities making the use of different spacecraft designs in each single mission mandatory. A program principle can only be applied by using worst-case (for the highest Δv) spacecraft designs that lead to very high and, thus, unreasonable masses. Short-stay options independent of a Human-Mars-Exploration Program or a program principle are problematic as well. Because of the varying velocity increments, a mission might become impossible in case of launch delays (two or four years later) inasmuch as the developed spacecraft design could be inadequate for those constellations. The only alternative that guarantees short total mission durations for piloted Mars missions with reasonable initial masses and fast transfer times is the use of continuous electric propulsion systems. The analyses showed that with electric propulsion systems with a thrust of about 100 N and a specific impulse of about 3000 s and high thruster efficiencies ($\gtrsim$30%) short transfer times and moderate initial masses can be achieved for low-energy trajectories. For long-stay options, similar results concerning flight time (but with higher return vehicle masses) have been achieved compared to impulsive propulsion systems. Short-stay options with total mission durations in the range of 500–600 days and acceptable initial spacecraft masses are feasible as well with electric propulsion systems. A great advantage is that it is possible to limit the required velocity increment by allowing longer flight times. Thus, it is possible to go to Mars and/or return to Earth with the same spacecraft design at any time. This is unique and cannot be achieved with any impulsive propulsion system. It was also shown that during the flight toward Mars the mission can be aborted at any time. All single missions within a Human-Mars-Exploration Program can use the same design for the mission scenario, for the spacecraft, and for the subsystems. Hence, a program principle can be applied for any mission and, thus, also for short-stay options. This also demonstrates the great gain in flexibility when using continuous electric propulsion systems. With continuous electric propulsion, all four design objectives that are mentioned can be met. The propulsion system parameters used here are not too futuristic, but rather conservative. In summary, the use of continuous electric propulsion systems is, in contrast to impulsive propulsion, a feasible option for short stays at Mars, and also, due to the high degree in flexibility, a very interesting one for long stays.

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