Adequate Electric Propulsion System Parameters for Piloted Mars Missions

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Abstract: Continuous electric propulsion proved to be a very interesting alternative as primary propulsion for interplanetary missions. It is also a good alternative for reasonable piloted Mars missions with short total mission durations. Within this paper the required electric propulsion system parameters for piloted Mars missions are assessed with the main focus on the adequate specific impulse. It will be shown that electric propulsion systems with thrust levels of 100 N and specific impulses of 3000 s already enable such missions. Potential electric propulsion concepts are investigated and compared.

Nomenclature

AF	=	Applied-Field
AU	=	Astonomical Unit
BICV	=	Basis-Infrastructure Cargo Vehicle
DLR	=	German Aerospace Center
DRM	=	Design Reference Mission
ECLSS	=	Environmental Control and Life Support System
EPS	=	Electrical Power System
ERV	=	Earth Return Vehicle
ERVCV	=	ERV-Cargo Vehicle
f	=	thrust density
F	=	thrust level
$F_{\max}$	=	maximum thrust level
GESOP	=	Graphical Environment for Simulation and Optimization
HCPA	=	High Current Plasma Accelerator
InTrance	=	INtelligent TRAjectory optimization using NeuroController Evolution
IRS	=	Institut für Raumfahrtsysteme
I_{sp}	=	specific impulse
$I_{\mathrm{sp,ad}}$	=	adequate specific impulse
ISPP	=	In-situ Propellant Production
LFA	=	Lorentz Force Accelerator
m	=	mass
m_0	=	initial spacecraft mass

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MAV	=	Mars Ascent Vehicle
m_{EPS}	=	mass of electrical power system
m_{Pay}	=	payload mass
MPD	=	MagnetoPlasmaDynamic (thruster)
m_{Pro}	=	propellant mass
m_{PS}	=	mass of propulsion system
m_{Tank}	=	propellant tank mass
NLP	=	NonLinear Programming
P_{e}	=	electrical input power
POV	=	Piloted Outward Vehicle
SF	=	Self-Field
v_∞	=	relative velocity at the target
Δv	=	velocity increment
η_{th}	=	thruster efficiency

I. Introduction

Although piloted Mars missions have been studied for years,[11] the existing concepts like NASA's Design Reference Mission[9] or the Caltech Mars Society Mission[8] are scenarios that are most unlikely to perform until 2033, which is currently the discussed time frame for such missions. Both concepts use impulsive propulsion systems (chemical or nuclear-thermal propulsion) to inject spacecraft into ballistic trajectories to Mars or Earth. Although such missions seem to be feasible from the technical point of view, the use of impulsive propulsion systems with their low specific impulses of about I_{sp}=500 s and 800 s for chemical and nuclear-thermal propulsion leads to long total mission durations of about three years, the so called long stay options. Further on, such mission scenarios are very inflexible without nearly any abort options once the crew is launched towards Mars. Reducing the total mission duration (short stay option) would require the use of a high energy return trajectory. For impulsive low I_{sp} propulsion systems this leads to huge and unreasonable spacecraft. The only propulsion system concept that achieves short transit times and moderate masses for short stay options and that allows flexible missions at the same time are continuous electric propulsion systems. Thereby, the term flexibility means that the chosen spacecraft designs guarantee a large number of mission opportunities or may even enable launch possibilities for any planetary constellation and may provide abort- and delay-options.

For piloted Mars mission scenarios that use continuous electric propulsion systems, however, few can be found in the literature concerning the required electric propulsion system parameters like e.g. the maximum thrust level F_{max} or the specific impulse I_{sp}. In general, fast transfer times can be achieved with a high trust acceleration (F/m) and, thus, with high thrust levels or low spacecraft masses. High specific impulses reduce the required amount of propellant, which can reduce the spacecraft masses and increase the thrust acceleration as well. However, for continuous electric propulsion systems better results concerning transfer time and mass cannot be achieved by simply increasing the thrust level or the specific impulse. Since external power sources are required, higher thrust levels and specific impulses come at the expense of power and mass, and the highest attainable specific impulse is not always the optimum choice for a given mission scenario. In fact there exist optimal parameters and especially an optimal specific impulse for each space mission. These parameters depend on the initial and target conditions in a quite complex manner. Within this paper the required electric propulsion system parameters for piloted Mars missions are assessed with the main focus on the adequate specific impulse. Further on the performance of different continuous electric propulsion systems is analyzed and compared.

II. Roundtrip Mission Scenario and Mass Modelling

The mission scenario in this paper is based on the NASA Design Reference Mission (NASA DRM). It is a six-person crew, surface base, split mission concept. However in contrast to the NASA DRM our chosen scenario is a short stay option with low energy trajectories during the outward transfers and a high energy trajectory for the return. On such high energy trajectories the spacecraft crosses Earth's orbit up to a certain solar distance. Thermal and medical aspect that evolve with such types of trajectories and their influence haven't been taken into account within this analysis. However for all trajectories a minimum solar distance was set at 0.7 AU, which seems to be a reasonable value. Like in the NASA DRM three outward transport vehicles are used: BICV, POV and ERVCV. The initial mass of each spacecraft m_0 can be divided in the following parts:

$$m_0 = m_{\text{Pay}} + m_{\text{EPS}} + m_{\text{PS}} + m_{\text{Tank}} + m_{\text{Pro}} \tag{1}$$

with m_{Pay} the payload, m_{EPS} the electrical power system (EPS) mass, m_{PS} the propulsion system, m_{Tank} the main propellant tank mass and m_{Pro} the propellant mass. For the mass dimensioning and calculation of subsystem components system studies have been performed at DLR and IRS. The component masses, volumes and power requirements for several subsystems are based on the results of this study and are technology predictions for 2030 and later. A payload mass breakdown for each spacecraft is given in Table 1. The "Basis-Infrastructure Cargo Vehicle" (BICV) delivers all components that are required for the surface base to Mars and lands them on the surface. This includes rovers, the science payload, a non-fuelled Mars Ascent Vehicle (MAV) and a propellant production plant (ISPP) including a reactor. The "ERV-Cargo Vehicle" (ERVCV) delivers a fully fuelled "Earth Return Vehicle" (ERV) to Mars and parks it in a 24 h elliptic orbit with a pericenter of 250 km. This orbit was chosen, since analyses, done at DLR, showed that this choice results not only in moderate initial masses of the MAV but also in a low escape Δv for the ERV. The "Piloted Outward Vehicle" (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, since a capsule landing would force the probably de-conditioned astronauts to walk several 100 meters 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. For short-stay options the POV and ERV have habitats that are identical in design, and they have been designed for 350 days of operational time. The POV and the BICV require Mars descent stages that consist of a heatshield, retrorockets and reaction-

POV		ERV		BICV		ERVCV	
payload	mass [mt]	payload	mass [mt]	payload	mass [mt]	payload	mass [mt]
ECLSS	5.3	ECLSS	5.3	Mars ascent vehicle	10.3	ECLSS	5.3
habitat	25.3	habitat	25.3	ISPP plant+reactor	19.4	habitat	25.3
		Earth entry capsule	6	pressurized rover	16	Earth entry capsule	6
				2 unpressurized rovers	16	return propellant tank	-[a]
				3 teleoperated rovers	3.6	chemical kickstage	-[a]
				additional equipment	1		
descent stage	17.5			descent stage	32.9		

[a]mass depends on chosen propulsion system

Table 1. Payload mass breakdown of the different transfer vehicles for the chosen split mission scenario

control engines including propellant. The total mass is based on a DLR internal study[12] and is about 17.5 and 32.9 mt respectively. The ERV carries an Earth entry capsule of about 6 mt. Upon arrival at Earth the capsule is jettisoned from the rest, and it performs an aeromaneuver with a direct landing on Earth. Except for the ERVCV all other transfer spacecraft use aeromaneuver upon arrival at Mars or at Earth to reduce the spacecraft mass and heliocentric transit time. An assessment of the heliocentric transit and the atmospheric phase (see also Ref. 4,5,15) 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. Since the ERV-cargo vehicle carries viable systems for the return flight an aeromaneuver was discarded for safety reasons. Instead the vehicle flies on a so called rendez-vous trajectory to Mars (v_∞=0) and uses a chemical kickstage for the final orbit insertion (Δv=221 m/s). In order to reduce the total mass to low Earth orbit per mission the ERV reuses the electrical power and the propulsion system of the ERVCV. The chemical kickstage is also reused by the return vehicle to inject the ERV from the chosen parking orbit into a heliocentric trajectory instead of spiralling out of Mars' gravity field.

III. Trajectory Simulation and Optimization

For continuous propulsion systems the trajectories cannot not be approximated as parts of ellipses but have to be calculated numerically. For such systems the transfer legs are divided into two phases, the planetocentric spiraling and the heliocentric transit phase. To save the spiralling time for the astronauts, the piloted outward vehicles starts spiralling without the crew. The crew is brought aboard with a crew taxi that docks with the POV at the end of the spiralling phase. This crew taxi consists of a chemical kickstage and a capsule identical in design to the ERV's Earth entry capsule and is designed to bring back the crew to Earth in case of emergencies 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 (transit time). Since the heliocentric transfers are more important than the spiraling phases, the analyses concerning the adequate propulsion systems parameters were mainly performed for the heliocentric phase. The heliocentric trajectory simulation and optimization within this investigation was done with the commercially available software GESOP[17] with the NLP-solver SNOPT[7] and InTrance,[6] a software, recently developed at DLR, that fuses evolutionary algorithms and neural networks. For the optimization it is assumed that the thrust vector can be modified at any time in magnitude and direction, until the desired target conditions (position, relative velocity) are achieved. For these analyses the trajectories are optimized with respect to transfer time.

IV. Propulsion System Parameters

The four major propulsion system parameters are the maximum thrust level $F_{\max}$, the specific impulse I_{sp}, the thruster efficiency η_{th} and the electrical input power P_{e}. They depend on each other according to Eq. 2.

$$P_{\mathrm{e}} = \frac{F_{\max} \cdot I_{\mathrm{sp}}}{2 \cdot \eta_{\mathrm{th}}} \tag{2}$$

Higher thrust levels and specific impulses or lower thruster efficiencies lead to higher input power levels. The mass of such an electrical power systems increases with increasing power levels as shown in Table 2 and Fig. 1.

	$\eta_{\mathrm{th}} = 30\%$		$\eta_{\mathrm{th}} = 40\%$		$\eta_{\mathrm{th}} = 50\%$	
Isp [s]	P_{e} [kW]	Mass [mt]	P_{e} [kW]	Mass [mt]	P_{e} [kW]	Mass [mt]
1500	2453	32.7	1839	24.8	1472	20.1
2000	3270	43.1	2453	32.7	1962	26.4
2500	4088	53.6	3066	40.5	2453	32.7
3000	4905	64.0	3679	48.3	2943	38.9
4000	6540	84.9	4905	64.0	3924	51.5
5000	8175	105.8	6131	79.7	4905	64.0

Table 2. Power levels and electric power system masses for different I_{sp} and thruster efficiencies at a thrust level of 100 N

Higher I_{sp} reduce the amount of propellant. However, looking at the total spacecraft mass, there is a point when for increasing I_{sp} the increase in the EPS mass exceeds the gain in propellant consumption, as shown in Fig. 1. Hence, a further increase in I_{sp} beyond that point is not recommendable. The specific impulse that leads to the lowest spacecraft mass is further on called the adequate specific impulse $I_{\mathrm{sp,ad}}$[a]. It depends on the mission scenario (Δv) and, thus, on the initial and target conditions e.g. the chosen celestial bodies, spacecraft mass and maximum relative velocity at the target. Each space mission that uses continuous electric propulsion has its own adequate propulsion system parameters. To investigate the performance of continuous propulsion for piloted Mars missions a systematic trajectory and mission analysis was performed at DLR with rather simple spacecraft models in the first step. The aim was not to find the absolute optimal propulsion system parameters but to narrow them down to a certain range. The results[13,14] indicated that for fast heliocentric transit times (4-6 months) a thruster acceleration of about 0.3 to 0.7 mm/s^2 is required.

[a]The term adequate instead of optimal is used, since the optimality has not been proven within this analysis.

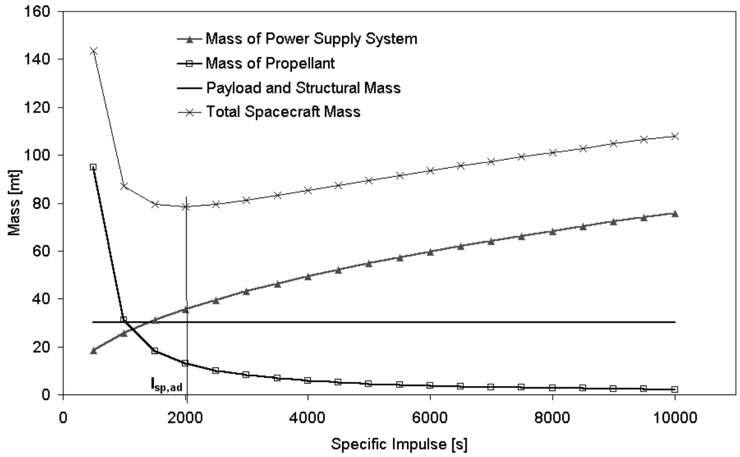

Figure 1. Mass versus specific impulse, qualitative diagram

Hence, for the chosen mission scenario and typical[b] spacecraft masses in the order of 200 mt the maximum thrust should be about $F_{\max}$=100 N. It was found that the promising range for the specific impulse $I_{\rm sp}$ is between 2000 and 6000 s.[13]

In a second step analyses have been performed using detailed subsystem models. The results are shown in Tables 3, 4 and 5 and in Figs. 2, 3 and 4 exemplary for a propulsion system using hydrogen as propellant with a propulsion system mass of $m_{\rm PS}$=630 kg. Due to different boundary conditions of the different spacecraft within the chosen split mission concept, different results concerning the adequate specific impulse were found.

For moderate energy outward trajectories using aerocapture at Mars (piloted outward vehicle and the basis-

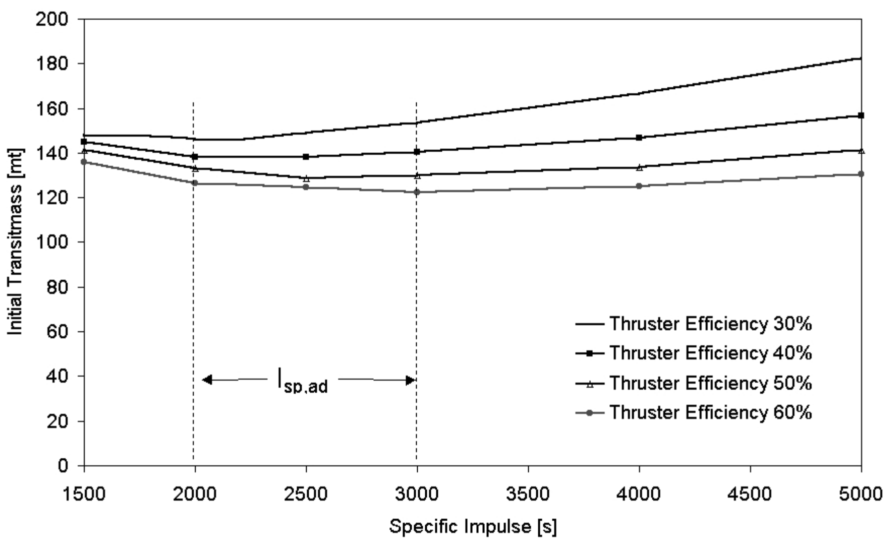

Figure 2. Mass versus specific impulse for moderate energy outward trajectories using aerocapture at Mars, POV

infrastructure cargo vehicle) the adequate specific impulse is about 2200 s for a thruster efficiency of 30 and 40% and increases to 2500 s or 3000 s respectively, for 50 and 60%. For high energy return trajectories the adequate $I_{\rm sp}$ increases and ranges between 2500 and 3000 s. For the heavy ERV-cargo vehicle that uses a rendez-vous trajectory the $I_{\rm sp,ad}$ is highest and ranges between 4000 and 5000 s. Since almost every spacecraft in the chosen Mars mission scenario has a different adequate specific impulse, all vehicles would require a different type of continuous electric propulsion system and electric power system. The development of three

[b]Initial masses that are similar to the spacecraft masses of other proposed concepts

	I_{sp} [s]	Time [s]	m_{EPS} [mt]	m_{Pro} [mt]	m_{Tank} [mt]	m_0 [mt]
$\eta_{th} = 30\%$	1500	143	32.7	52.8	13.8	147.9
	2000	143	43.1	43.1	11.5	146.3
	2500	144	53.6	36.6	9.9	148.8
	3000	146	64.0	31.9	8.8	153.4
	4000	152	84.9	25.6	7.3	166.4
	5000	162	105.8	21.8	6.4	182.6
$\eta_{th} = 40\%$	1500	140	24.8	56.6	14.7	144.7
	2000	139	32.7	44.8	11.9	138.00
	2500	139	40.5	38.7	10.4	138.3
	3000	140	48.3	34.2	9.4	140.5
	4000	143	64.0	26.7	7.6	146.9
	5000	147	79.7	22.0	6.4	156.8
$\eta_{th} = 50\%$	1500	138	20.1	57.6	14.9	141.3
	2000	136	26.4	45.9	12.1	133.0
	2500	136	32.7	37.4	10.1	128.9
	3000	136	38.9	33.4	9.2	130.1
	4000	138	51.5	26.3	7.5	133.8
	5000	141	64.0	22.0	6.4	141.1
$\eta_{th} = 60\%$	1500	137	17.0	55.7	14.5	135.8
	2000	135	22.2	44.0	11.7	126.5
	2500	134	27.4	38.2	10.3	124.6
	3000	134	32.7	32.1	8.9	122.3
	4000	135	43.1	25.8	7.4	124.9
	5000	137	53.6	21.7	6.4	130.3

Table 3. Flight times and subsystem masses for the piloted outward vehicle for different specific impulses and thrust efficiencies at 100 N, payload mass 48 mt and propulsion system mass of 630 kg

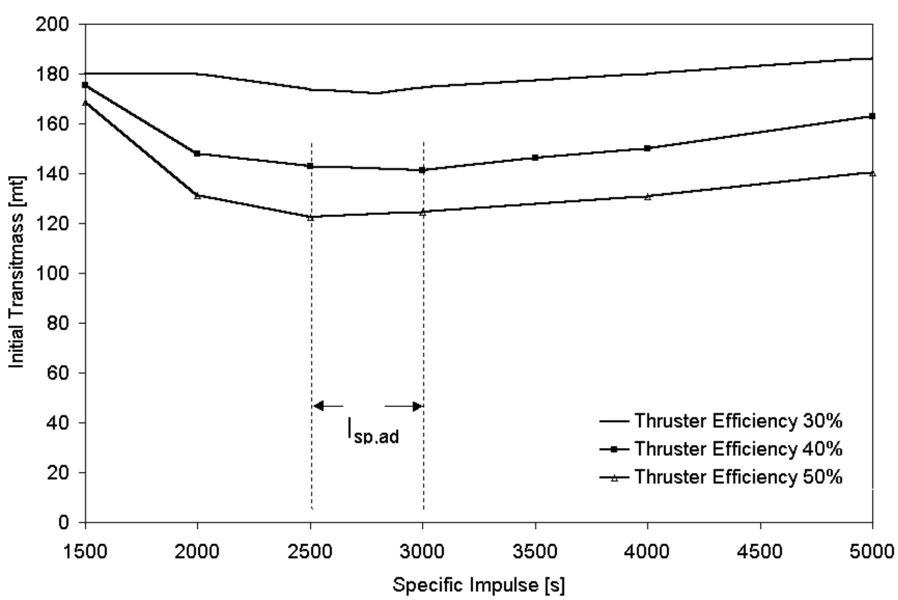

Figure 3. Mass versus specific impulse for high energy return trajectories using aerocapture at Earth, ERV

	I_{sp} [s]	Time [s]	m_{EPS} [mt]	m_{Pro} [mt]	m_{Tank} [mt]	m_0 [mt]
etw=30%	1500	343	32.7	88.2	22.0	180.0
	2000	318	43.1	79.7	20.1	180.0
	2500	312	53.6	66.3	16.9	173.9
	2800	313	59.8	59.9	15.4	172.3
	3000	314	64.0	58.2	15.1	174.5
	4000	328	84.9	45.8	12.1	180.0
	5000	367	105.8	34.0	9.3	186.3
etw=40%	1500	318	24.82	90.78	22.62	175.35
	2000	291	32.7	62.1	15.9	147.9
	2500	286	40.5	51.6	13.5	142.7
	3000	287	48.3	44.1	11.7	141.3
	3500	291	56.2	41.9	11.2	146.4
	4000	296	64.0	38.5	10.4	150.0
	5000	306	79.7	36.1	9.8	162.7
etw=50%	1500	304	20.12	89.28	22.27	168.79
	2000	275	26.4	53.6	13.9	131.1
	2500	271	32.7	41.6	11.1	122.5
	3000	271	38.9	38.1	10.3	124.5
	4000	277	51.5	32.9	9.1	130.6
	5000	286	64.0	30.6	8.5	140.3

Table 4. Flight times and subsystem masses for the piloted return vehicle for different specific impulses and thrust efficiencies at 100 N, payload mass 36.6 mt and propulsion system mass of 630 kg

	I_{sp} [s]	Time [s]	m_{EPS} [mt]	m_{Pro} [mt]	m_{Tank} [mt]	m_0 [mt]
η_{th}=40%	1500	520	24.82	120.6	29.5	315.5
	2000	484	32.7	114.2	28.0	315.5
	2500	442	40.5	106.1	26.1	313.4
	3000	483	48.3	94.3	23.4	306.6
	4000	413	64.0	71.4	18.1	294.1
	5000	416	79.7	60.7	15.6	296.7
η_{th}=50%	1500	409	20.1	103.8	25.6	290.1
	2000	502	26.4	120.0	29.3	316.4
	2500	418	32.7	98.9	24.5	296.7
	3000	386	38.9	83.2	20.9	283.6
	4000	372	51.5	59.3	15.3	266.7
	5000	372	64.0	49.1	12.9	266.7
η_{th}=60%	1500	541	17.0	136.7	33.1	327.4
	2000	479	22.2	112.9	27.7	303.5
	3000	384	32.7	88.2	22.0	283.5
	4000	374	43.1	68.5	17.5	269.7
	5000	371	53.6	57.6	14.9	266.7

Table 5. Flight times and subsystem masses for the ERV-Cargo vehicle for different specific impulses and thrust efficiencies at 100 N, payload mass 140 mt and propulsion system mass of 630 kg

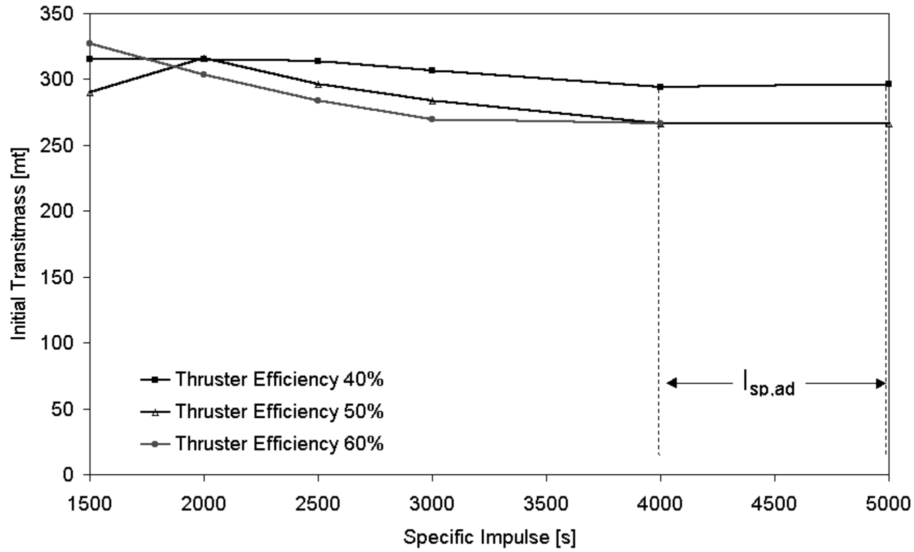

Figure 4. Mass versus specific impulse for an outward rendezvous trajectory,ERVCV

different electric propulsion and power systems for one space mission seems very unlikely. Further on the return vehicle should reuse the ERVCV's power system to save mass. It, thus, requires a propulsion system that is similar to the ERCV's. The only solution for a reasonable mission would be that all spacecraft use the same propulsion and power system. Hence, for the best mission performance a good compromise concerning the adequate specific impulse has to be found. The results indicate that a specific impulse of about 3000 s and a thruster efficiency as high as possible results in the overall best performance.

The power level and, thus, the EPS mass does not only increase with increasing I_{sp} but also with increasing thrust level as shown in Table 6. For return trajectories calculations have been performed for different I_{sp} and F_{max} at constant power levels. As long as the specific impulse is in the range of the "adequate" value

	Propulsion System Parameters				
	200 N 1500 s	150 N2000 s	100 N 3000 s	100 N 1500 s	50 N 3000 s
P_e [kW]	4905	4905	4905	2453	2453
Time [days]	399	395	371	428	569
m_{EPS} [mt]	64.0	64.0	64.0	32.7	32.7
m_{Pro} [mt]	96.4	77.0	59.9	93.4	53.1
m_{Tank} [mt]	23.9	19.4	15.5	23.2	13.8
m_0 [mt]	198.5	188.4	176.6	186.4	152.2

Table 6. Return flight times and ERV masses for different thrust levels and I_{sp} at $\eta_{th} = 30\%$ with 36.6 mt payload and a propulsion system mass of 630 kg

it is always better to "invest" power into I_{sp} rather than into thrust, since lower initial masses and shorter transit times can be achieved. Although the initial mass for the 50 N, 3000 s propulsion parameter set at P_e=2453 kW is lower than for the 100 N, 1500 s set, it can be seen in Table 6 that thrust levels below 100 N lead to a significant increase in transit time.

Summarizing the results it was found, that the most adequate continuous propulsion system parameter set for piloted Mars missions seems to be a thrust level of about 100 N and specific impulse of about 3000 s. The thruster efficiency should be as high as possible to reduce the required electrical input power and to achieve moderate initial spacecraft masses and should be η_{th}
$geq 40\%$.

V. Continuous Propulsion System Concepts

The previous section revealed the required continuous propulsion system parameters for piloted Mars mission. Next, a continuous propulsion concept must be chosen and developed. Besides a total thrust level of about 100 N and a specific impulse of about 3000 s and high thruster efficiencies, such systems further need a high thrust density (thrust per diameter) in order to avoid large thruster clusters and an availability within the next 30 years.In general, solar-thermal, electric, solar- or lasersails as well as fusion, antimatter propulsion and nuclear pulsed concepts are continuous propulsion systems. However only few concepts fulfill all the requirements that are stated above. Solar thermal propulsion systems offer high thrust densities but are weak in their specific impulse, which is clearly below 1000 s. Solar- or lasersails as well as fusion, antimatter propulsion and nuclear pulsed concepts seem very unlikely to be developed for piloted Mars missions until 2030. Hence, the only remaining continuous propulsion systems that could enable reasonable piloted Mars missions are electric propulsion systems. For the different existing electric propulsion concepts the achievable specific impulses and thrust levels per thruster are shown in Fig. 5.

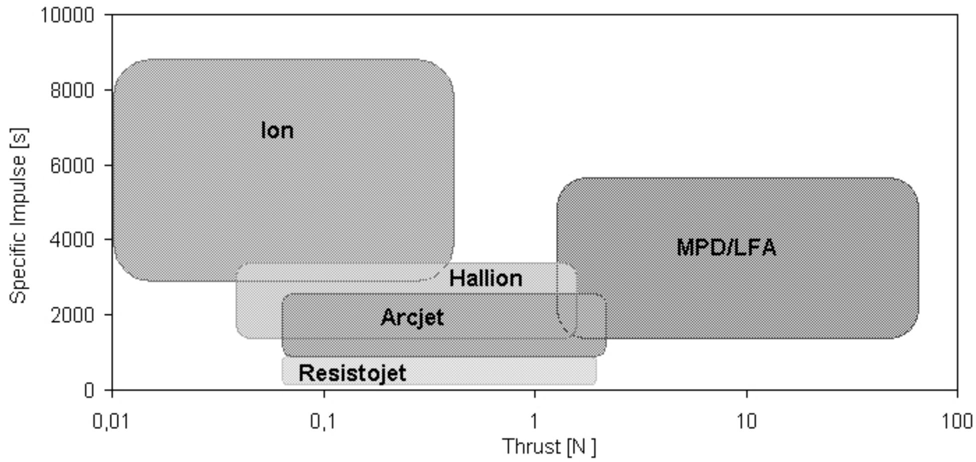

Figure 5. Thrust levels and specific impulses for different electric propulsion system concepts

Based on the propulsion system parameter requirements for piloted Mars missions it can be seen that Resistojets have to be excluded, since their I_{sp} is limited to below 1000 s. All electrostatic (ion) engines have to be excluded as well. Due to their low thrust density the use of such concepts would result in large and heavy clusterplates with diameters of more than 15 m, which is beyond any fairing capability of discussed future heavy lift launch vehicles (8 m maximum diameter). High power thermal Arcjets, magnetoplasmadynamic self-field (SF-MPD) and applied-field (AF-MPD) thrusters, also known as Lorentz Force Accelerators (LFA), as well as Hallion thrusters seem adequate and some have already exceeded the required propulsion system parameters.

To compare the different concepts among each other, propulsion system models have been implemented for each of them. The models for high-power thermal Arcjets and self-field MPD thrusters use technology predictions that are based on experimental data with the Hiparc, DT2 and DT6 engines from IRS.[1,3,18] The applied-field MPD thruster models are based on data from Ref.2 and 16. The Hallion thruster model is based on the NASA457M thruster.[10] The propulsion systems parameters for the different thruster models as well as the thrust density f are summarized in Table 7. Table 8 shows the total amount of thrusters including spares as well as the resulting propulsion and power system masses for a thruster cluster with 100 N.

The results of the heliocentric trajectory simulation and optimization for the different transfer vehicles is shown in Tables 9, 10, 11 and 12.

For the piloted outward vehicle the different propulsion systems have a rather similar performance concerning flight time and spacecraft mass. Fast flight times of 130 to 180 days can be achieved at masses of about 130 to 148 mt. However the best performance concerning flight time and mass can be achieved with the Xe Hallion thruster model due to its high thruster efficiency and therefor lowest EPS mass. For the return flight the flight times and masses are higher and here, the differences between the different thruster models become more obvious. The masses vary between 122 mt for the Hallion thruster model and reach up to

Thruster	P_e [kW]	$\dot{m}$ [mg/s]	F [N]	I_{sp} [s]	η_{th} [%]	f [N/m^2]
H_2 Arcjet	167	150	3.75	2500	28	1326.29
H_2 SF-MPD	1761.90	1850	37	2039	21	4711
Ar SF-MPD	1560.74	4300	60.2	1427	27	3801.28
Li LFA (AF MPD)	214	90	4.55	5153	54	226.3
Li HCPA (AF MPD)	500	300	12	4077	48	346.46
Xe Hallion Thruster	73.2	102.7	2.95	2929	58	17.98

Table 7. Performance data of different electric propulsion systems

	H_2-Arcjet	H_2-SF-MPD	Ar-SF-MPD	LiLFA	HCPA	Xe-NASA457M
amount[a]	36	5	4	30	15	46
m_{PS} [kg]	598	880	628	688	1542	1287
P_e [MW]	4.42	5.26	3.12	4.69	4.50	2.48
m_{EPS} [mt]	57.9	68.5	41.1	61.1	58.7	32.9

[a]including spares

Table 8. Amount of thrusters, propellant and power system masses for the different thruster models at a maximum thrust of about 100 N

	H_2-Arcjet	H_2-SF-MPD	Ar-SF-MPD	LiLFA	HCPA	Xe-NASA457M
time [days]	151	182	142	138	136	132
m_{Pro} [mt]	22.5	23.5	40.4	21.0	28.1	33.6
m_{Tank} [mt]	6.5	6.8	1.4	1.2	1.4	1.1
m_0 [mt]	135.6	147.7	131.7	132.1	137.9	117.0

Table 9. Transit-times and subsystem masses of the piloted outward vehicle with m_{Pay}=48.1 mt for the different thruster models

	H_2-Arcjet	H_2-SF-MPD	Ar-SF-MPD	LiLFA	HCPA	Xe-NASA457M
time [days]	394	427	378	311	308	290
m_{Pro} [mt]	64.9	74.8	75.1	37.1	47.6	49.6
m_{Tank} [mt]	16.6	18.9	1.9	1.7	2.0	1.2
m_0 [mt]	176.6	199.7	155.4	137.2	146.4	121.7

Table 10. Transit-times and subsystem masses of the return vehicle with m_{Pay}=36.6 mt for the different thruster models

	H_2-Arcjet	H_2-SF-MPD	Ar-SF-MPD	LiLFA	HCPA	Xe-NASA457M
time [days]	347	336	333	293	304	302
m_{Pro} [mt]	23.3	35.0	37.7	17.5	18.9	22.8
m_{Tank} [mt]	6.78	9.6	1.4	1.1	1.1	1.0
m_0 [mt]	187.8	213.1	180.1	179.6	179.6	157.3

Table 11. Transit-times and subsystem masses of the basis-infrastructure cargo vehicle with m_{Pay}=99.2 mt for the different thruster models

	H_2-Arcjet	H_2-SF-MPD	Ar-SF-MPD	LiLFA	HCPA	Xe-NASA457M
time [days]	479	673	529	381	424	439
m_{Pay} [mt]	152.2	168.2	142.6	97.2	109.2	107.3
m_{Pro} [mt]	95.3	106.4	124.4	28.5	38.1	42.8
m_{Tank} [mt]	23.7	26.2	2.5	1.4	1.7	1.2
m_0 [mt]	329.6	370.2	311.2	188.9	209.3	185.6

Table 12. Transit-times and subsystem masses of the ERV-Cargo vehicle for the different thruster models, reuse of the ERCV's power and propulsion system for the return vehicle

nearly 200 mt for the H_2-SF-MPD model. The Hallion thruster model performs best as expected, since it almost has the adequate propulsion parameters for such types of trajectories. With the AF-MPD models a good performance can be achieved as well, but they have higher masses compared to the Hallion thruster model due to the larger EPS masses (I_{sp} is too high). For the H_2-SF-MPD model and the Arcjet models the thruster efficiency and the specific impulse are too low and the tank masses to high to achieve moderate and comparable flight times with moderate initial masses for the ERV. The Ar SF-MPD model is a special case. Although it has the lowest I_{sp}, its performance is not too bad compared to the others. If it is possible to improve its thrust efficiency and I_{sp} it could be also a good alternative, since this propulsion system cluster would contain only few thrusters and would, thus, be less complex compared to a cluster containing 46 Hall thrusters or 30 Li LFAs. However, the for the Arcjet model and the SF-MPD models the flight exceeds 350 days. Hence, using these propulsion systems would lead to a further increase in spacecraft mass of about 3 to 7 mt, since the mass of the life support system has to be adjusted to the longer flight time. For the basis-infrastructure cargo vehicle the trend is similar as for the piloted outward vehicle. The Hallion thruster model performs best in terms of total mass, followed by the two AF-MPD models, the Ar SF-MPD model and the hydrogen thruster models. However, the flight times are twice as long compared to the piloted spacecraft, due to the larger payload mass. Finally the ERV-cargo vehicle reveals the biggest problems of when dealing with short stay options. While with the AF-MPD and the Hallion thruster model the ERVCV's payload masses are around 100 mt, they already reach up to more than 140 mt for the Arcjet and the SF-MPD models. The masses but especially the flight times are significantly higher compared to the other outward vehicles due to the chosen rendez-vous trajectory. However, for the AF-MPD and the Hallion thruster models the achieved values still seem to be reasonable. For the SF-MPD and the Arcject model, however, the total masses are significantly higher and reach values of more than 300 mt.
The results show that in general all the analyzed propulsion systems seem to be feasible for most of the transfer vehicles. However, the results for the ERV-cargo vehicle show that AF-MPDs or Hallion thrusters or in general propulsion system with similar propulsion system parameters seem to be the best choice for piloted Mars missions. However, one should not forget that these concepts lack in thrust density. Further assessments concerning the performance of large thruster clusters containing many thrusters have to be done. A criteria against the very good performance of the Hallion thruster could be the large amount of required Xenon (140 mt per mission), since the availability of that type of propellant in these amounts still seems to be uncertain from the current production output.[19,20] Even Arcjets and SF-MPD should not be neglected. Further research and development concerning thruster efficiency and I_{sp} could lead to very promising alternatives, since their large thrust density would require only a few thrusters per cluster and the whole propulsion system would, thus, be less complex. Although the continuous electric propulsion systems parameters that are required for piloted Mars missions have been narrowed down, the choice for a special electric propulsion system concept is still open to discuss.

VI. Summary and Conclusions

The use of continuous electric propulsion prove to be the only choice for piloted Mars mission with short total mission durations, moderate flight times and reasonable spacecraft masses. For such missions it was found that the thrust level should be around 100 N. The adequate specific impulse for such a mission is about 3000 s. The thruster efficiency should be as high as possible ($\eta_{\text{th}} \geq 40\%$) to reduce the electrical input power and power system mass. Further requirements are high thrust densities (thrust per diameter) in order to avoid large thruster clusters and an availability within the next 30 years according to the current development time frame of such missions. Due to these requirements Resistojets as well as all ion propulsion concepts have to be excluded. Highpower electrothermal Arcjets, magnetoplasmadynamic self- and applied-field thrusters as well as Hallion thrusters are promising concepts. However, all these concepts still require further development to improve upon thruster efficiency (Arcjets, SF-MPD) as well as reliability (Hallion thrusters).

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References

[1]http://www.irs.uni-stuttgart.de/RESEARCH, 2004.

[2]V.P. Ageyev, V.G. Ostrovsky, and V.A. Petrosov. High Current Stationary Plasma Thruster of High Power. IEPC-93-117, 1993.

[3]M. Auweter-Kurtz. High Power Hydrogen Arcjet Thrusters. *Journal of Propulsion and Power*, Volume 14(No.5):pp 764, 1998.

[4]R. D. Braun and R. W. Powell. Earth Aerobraking Strategies for Manned Return from Mars. *Journal of Spacecraft and Rockets*, 29(3):297–304, 1992.

[5]R. D. Braun, R. W. Powell, and L. C. Hartung. Effect of Interplanetary Options on a Manned Mars Aerobraking Configuration. NASA Technical Paper 3019, 1990.

[6]B. Dachwald. Low Thrust trajectory Optimization and Interplanetary Mission Analysis Using Evolutionary Neurocontrol. Ph.D. Thesis, Institut für Raumfahrttechnik, Universität der Bundeswehr, München, www.spacesailing.net, 2004.

[7]P. Gill, W. Murray, and M Saunders. User's Guide for SNOPT 5.3, 1998.

[8]C. Hirata, J. Greenham, N. Brown, D. Shannon, and J. Burke. A New Plan for Sending Humans to Mars. The Caltech Mars Society Mission 2.0, California Institute of Technology, 1999. http://mars.caltech.edu/chris_its/mars/cmsm2r.html.

[9]S. Hoffman and D. Kaplan. The Reference Mission of the NASA Mars Exploration Study Team. http://www-sn.jsc.nasa.gov/marsref/contents.html, 1997.

[10]D.H. Manzella, R.S. Jankovsky, and R.R. Hofer. Laboratory Model 50kW Hall Thruster. AIAA-2002-3676, 2002.

[11]D. Portree. Humans to Mars: Fifty Years of Mission Planning, 1950–2000. Monographs in Aerospace History, Number 21, NASA SP-2001-4521, 2001.

[12]A. Reinacher. Systemanalytische Vorauslegung einer Aufstiegsstufe am Mars für zukünftige bemannte Marsmissionen. Diploma Thesis, IRS-04-S08, Institut für Raumfahrtsysteme, Universität Stuttgart, Stuttgart, 2004, in German.

[13]T. D. Schmidt, B. Dachwald, W. Seboldt, and M. Auweter-Kurtz. Flight Opportunities from Mars to Earth for Piloted Missions Using Continuous Thrust Propulsion. AIAA-2003-4573, 2003.

[14]T. D. Schmidt, W. Seboldt, and M. Auweter-Kurtz. Propulsion Options for Manned Mars Missions. Proceedings of the 6th International Symposium, Propulsion for Space Transportation of the XXIth Century, Versailles, France, 2002.

[15]M. E. Tauber, G. E. Palmer, and L. Yang. Earth Atmospheric Entry Studies for Manned Mars Missions. AIAA-90-1699, 1990.

[16]V.B. Tikhonov, S.A. Semenikhin, J.R. Brophy, and J.E. Polk. Performance of 130 kW MPD Thruster with an External Magnetic Field and Li as a Propellant. IEPC-97-117, 1997.

[17]TTI GmbH. GESOP 4.5.3 Software User Manual. TTI GmbH, Dept. of Optimization, Guidance and Control, Stuttgart, 2003. http://www.gesop.de.

[18]T. Wegmann. Experimentelle Untersuchung kontinuierlich betriebener magnetoplasmadynamischer Eigenfeldantriebe. Doktorarbeit, Fakultät für Luft- und Raumfahrttechnik, Universität Stuttgart, Stuttgart, 1994, in German.

[19]R.P. Welle. Availability Considerations in the Selection of Inert Propellants for Ion Engines. AIAA 90-2589, 1990.

[20]R.P. Welle. Xenon and Krypton Availability for Electric Propulsion: An Updated Assessment. AIAA 93-2401, 1993.