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SYSTEMES PROPULSIFS ELECTRIQUES DE FORTE PUISSANCE POUR PROPULSION NUCLEO-ELECTRIQUE

HIGH POWER ELECTRIC PROPULSION SYSTEM FOR NEP

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ABSTRACT :

Recent US initiatives in Nuclear Propulsion lend themselves naturally to raising the question of the assessment of various options and particularly to propose the High Power Electric Propulsion Subsystem (HPEPS) for the Nuclear Electric Propulsion (NEP). The purpose of this paper is to present the guidelines for the HPEPS with respect to the mission to Mars, for automatic probes as well as for manned missions. Among the various options, the technological options and the trajectory options are pointed out.

The consequences of the increase of the electrical power of a thruster are first an increase of the thrust itself, but also, as a general rule, an increase of the thruster performance due to its higher efficiency, particularly its specific impulse increase. The drawback is as a first parameter, the increase of the thruster's size, hence the so-called "thrust density" shall be high enough or shall be drastically increased for ions thrusters. Due to the large mass of gas needed to perform the foreseen missions, the classical xenon rare gas is no more in competition, the total world production being limited to 20 –40 tons per year. Thus, the right selection of the propellant feeding the thruster is of prime importance. When choosing a propellant with lower molecular mass, the consequences at thruster level are an increase once more of the specific impulse, but at system level the dead mass may increase too, mainly because the increase of the mass of the propellant system tanks. Other alternatives, in disruption with respect to the current technologies, are presented in order to make the whole system more attractive. The paper presents a discussion on the thruster specific impulse increase that is sometime considered an increase of the main system performances parameter, but that induces for all electric propulsion systems drawbacks in the system power and mass design that are proportional to the thruster specific power increase (kW/N). The electric thruster specific impulse shall be optimized w.r.t. the mission.

The trajectories taken into account in the paper are constrained by the allowable duration of the travel and the launcher size. The multi-arcs trajectory to Mars (using an optimized combination of chemical and Electric propulsion) are presented in detail. The compatibility with NEP systems that implies orbiting a sizeable nuclear reactor and a power generation system capable of converting thermal into electric power, with minimum mass and volumes fitting in with Ariane 5 or the Space Shuttle bay, is assessed.

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1 - INTRODUCTION

Recent US initiatives in Nuclear Propulsion lend themselves naturally to raising the question of the assessment of various options and particularly to propose the High Power Electric Propulsion Subsystem (HPEPS) for the Nuclear Electric Propulsion (NEP).

In the paper, the main mission considered is the mission to Mars (and return). Thus, the purpose of this paper is to present the guidelines for the HPEPS for automatic probes as well as for manned missions. The current technologies available cannot achieve this kind of mission unless huge launchers to be develop and with the corresponding huge budget.

Among the various options, the technological options are to use the electric propulsion.

Using the electrical propulsion, the higher performance of the propulsion enables to foresee very high energy mission on the condition that a very large power is available aboard the spacecraft.

The consequences of the increase of the electrical power aboard is to increase the power of the thrusters. This aspect is first discussed. Then, the right selection of the propellant feeding the thruster is analysed. Finally the trajectory options are pointed out.

2 - CONSEQUENCES OF INCREASED ELECTRICAL POWER FOR THE PROPULSION SYSTEM

a. Optimum specific impulse

While the practical limit to high-ΔV space missions using chemical propulsion resides with prohibitive propellant mass due to limited propellant energy density, electric propulsion systems use an external power source, thus allowing superior performance for propulsively demanding missions, provided that sufficient power is available. Such missions, *e.g.*, to outer planets or their satellites, or to trans-Neptunian objects, considerably benefit from electric propulsion because of the great specific impulse values that can be achieved. If the requirement of a short trip time is added—whether to avoid physical or psychological damage for the astronauts or to allow for a reasonable delay before scientific return for a robotic mission to a distant object—very high power levels become mandatory. This can best be visualized through the expression of the electric power-to-thrust ratio, or specific power, given by :

$$\frac{P_e}{F} = \frac{1}{2\eta} g_0 \; I_{sp} \qquad \textbf{(1)}$$

where g_0 is the standard free fall, I_{sp} is the specific impulse, and η is the thrust efficiency, *i.e.*, the efficiency of converting thruster input power into effective jet power. Because thrust efficiencies are typically within the range 50—70 %, the equation can even be reduced to a first approximation to :

$$\frac{P_e}{F} \approx g_0 \; I_{sp} \qquad \textbf{(2)}$$

Inspection of this simple equation shows that a high thrust F requires a high input electric power in order to maintain even a moderate specific impulse. Conversely, maintaining an even moderate thrust – or acceleration – at extremely high values of specific impulse will come at the cost of an extremely high power level. Thus, while a high specific impulse is desirable in order to minimize propellant mass for a given mission ΔV, this must be balanced by the realization that maintaining a given thrust level can only be achieved by increasing power, and therefore power plant mass. In other words, for a given trajectory, *i.e.*, a given ΔV and trip time – or acceleration – there exists an optimal I_{sp} which we may call Isp_{opt} (Figure 1).

The typical architecture of a nuclear electric propulsion system, and its main interaction with the rest of the spacecraft, are conceptually represented in Figure 2 . An important difference of (nuclear) electric propulsion when compared to chemical propulsion is indeed the necessity for an external power source, represented here by the power subsystem.

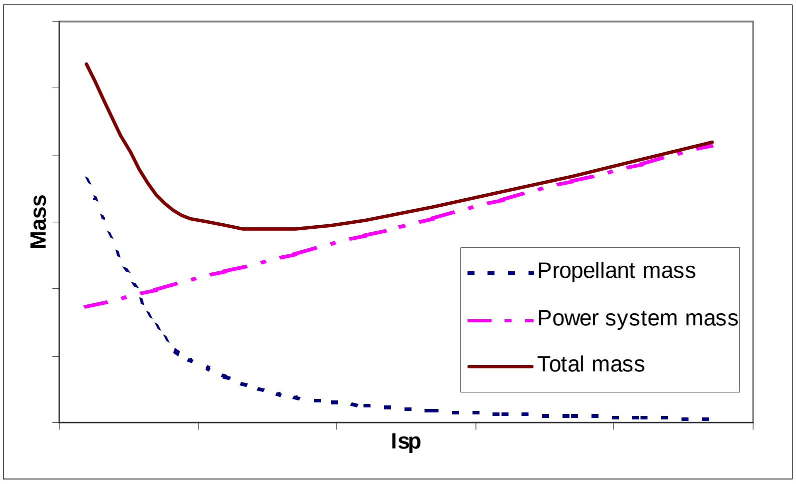


Figure 1 ***Optimal specific impulse for a given constant-thrust mission, after [1].***

The mass of the power sub-system is an increasing function of power, and therefore, for a given thrust level, an increasing function of I_{sp}. The optimal value of I_{sp} depends on the specific power plant mass α, in kg/kW, on the efficiency, and on thrust time Δt:[1]

$$I_{sp_{opt}} = \frac{1}{g_0}\left(\frac{2\eta\Delta t}{\alpha}\right) \tag{3}$$

A specific impulse larger than Isp_{opt} leads to a penalty on the system dry mass, whereas a specific impulse lower than Isp_{opt} leads to a penalty on propellant mass. As a rule of thumb, an optimized electric propulsion system should thus have a comparable balance between propellant mass and power system dry mass.

The steeper the dependence of power system mass on power level (or the larger α), the lower the optimum specific impulse, and conversely. Likewise, the lower the ΔV – or the lower the propulsion needs for the mission – the lower Isp_{opt}. Therefore, propulsively demanding missions such as exploration missions to the outer planets and their moons may require I_{sp} values in excess of 4000 s [2], whereas one-way Mars transfers are routinely done with chemical propulsion and may be done with electric propulsion at moderate I_{sp}. The case for electric propulsion becomes stronger as the mission ΔV increases, as, say, in the case of a Mars sample return.

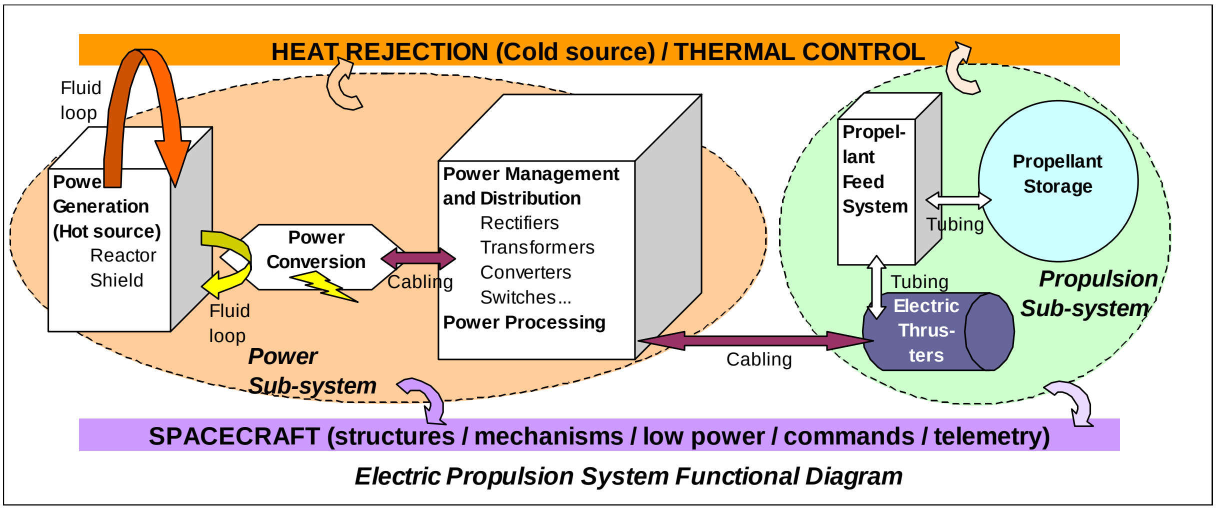


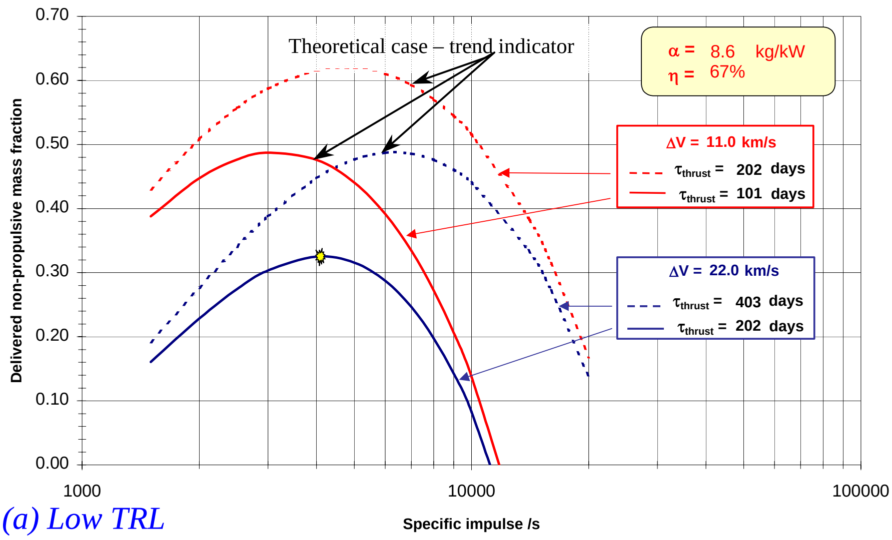
Figure 2 ***Nuclear electric propulsion system schematic diagram***

As an illustration, Figure 3 shows the dependence of delivered non-propulsive mass fraction on specific impulse for several values of mission ΔV and system α. Here, the delivered non-propulsive mass includes the mass of anything that is not propulsion-related, *i.e.*, anything that is *not* included in the nuclear electric propulsion system comprising power source and conversion, shielding, radiator, power management, conditioning and distribution, thrusters, propellant, tanks and fluid systems. The example takes as an assumption a 252-day, one-way transfer to Mars from Low Earth Orbit (LEO) to Low Mars Orbit (LMO). The total accelerated (thrust-on) time on this real trajectory is 202 days, with a total ΔV of 22 km/s (blue solid line). The theoretical effect of either halving total mission ΔV (red lines) or increasing thrust-on time (solid lines) is illustrated. Allowing more thrust-on time for the same total ΔV favors lower-thrust (lower-acceleration) and thus higher-Isp solutions. Conversely, short time-constrained or moderate-ΔV missions require more moderate values of I_{sp}. In addition, and as mentioned earlier, the lower the system specific mass, the higher the optimum specific impulse and the greater the mission performance. This example was calculated using the mission and technology assumptions indicated in Table 1. The efficiency term η given in Figure 3 relates to the total efficiency of converting source electric power to effective jet power, and the system α is the ratio of total electric propulsion system dry mass (Figure 2) to source electrical power.

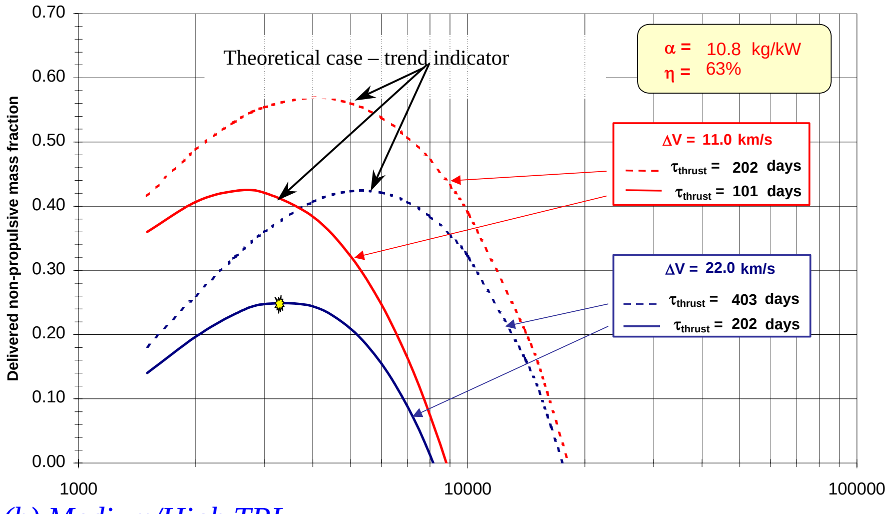
Table 1 Summary of mission and technology assumptions for Figure 3

Case	**Electrical power level**	**Reactor technology**	**Conversion technology** **(Technology Readiness Level TRL)**	**Radiator technology**	**Payload mass (tons)**	**Initial mass* (tons)**
a	1 MW	Liquid metal in core reactor	Rankine (Low TRL)	C-C heat pipe radiating on both sides	6.4	41.0
b	1 MW	Gas-cooled reactor (pin fast)	Brayton (medium/high TRL)	C-C heat pipe radiating on one side	6.4	55.2
c	1 MW	Liquid metal / heat pipe reactor	Thermoelectric (high TRL)	C-C heat pipe radiating on both sides	1.3	62.0

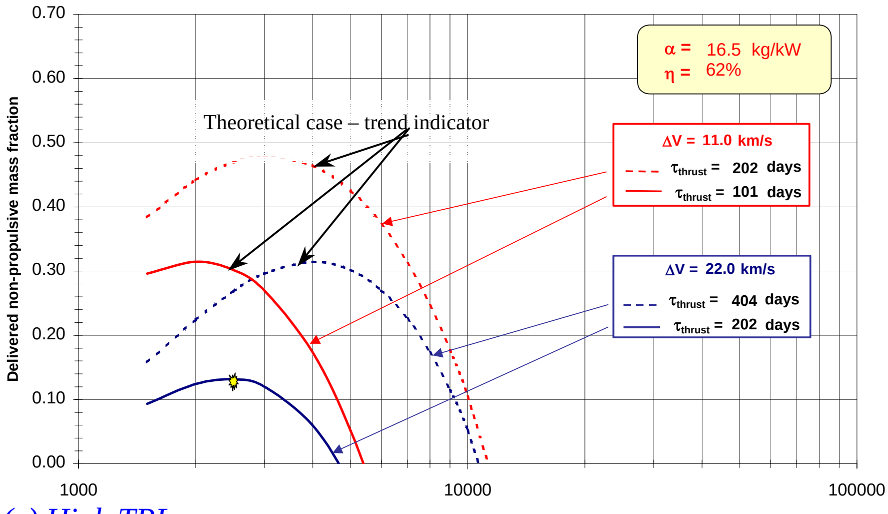
*Mass in LEO, including 30% gross mass margin.



(a) Low TRL



(b) Medium/High TRL



(c) High TRL

Figure 3 ***Delivered mass fraction as a function of I_{sp} for two values of ΔV and acceleration.*** Figures a (top) through c (bottom) illustrate the effect of increasing the TRL indeed, increasing the system α (kg/kW) on mission performance and optimum I_{sp}. For short thrust duration (101 days), the optimum I_{sp} is in the range 1500 to 2500 s for values of α = 16.5 kg/kWe.

b. Thruster technology choices

Case *c* in the example detailed in the Table 1and Figure 3 represents a reasonable estimate of nuclear propulsion system options and performance available for first-generation NEP missions. The optimum specific impulse is roughly comprised within the range 2000—3000 s for the "real" example trajectory (22 km/s of ΔV and 202 days of total thrust-on time). Longer-term technology will further reduce the system α or may call for larger ΔVs, such as in the case of round trips to Mars. Increased values of I_{sp} can thus become desirable in the long term, although the peak in mission performance as a function of I_{sp} is relatively flat (Figure 3).

The main electric thruster technology options for such future NEP Mars missions are Hall-Effect (or stationary plasma) Thrusters (HETs), Gridded-Ion Thrusters (GITs) and Magneto-Plasma Dynamic Thrusters (MPDTs).

Hall thrusters are a technology that has first been flown successfully on Russian spacecraft. A total of over 120 such devices have been flown successfully since 1972, and this technology has recently penetrated the western commercial space market (MBSAT launched in March 2004, Intelsat-10-02 launched in June 2004, and Inmarsat 4-F1 launched in March 2005). All propulsion systems are reported to be operating flawlessly.

In addition to this, the currently-ongoing scientific SMART-1 ESA lunar mission is using a single PPS®1350, which to date has logged over 4600 hours of operation. The disruption in this technology for applications to space exploration such as Mars missions will come from the development of higher-I_{sp}, high power devices. The development effort towards such devices is already underway, with the demonstration of the high versatily of HETs [3][4] and the on-going research work on dual-stage devices [5]. The main limitation to this technology is the unavailability of very high values of I_{sp}, as may be mandatory for propulsively more demanding missions such as, *e.g.*, the exploration of the outer solar system and beyond. On the other hand, its key advantages lie in its simplicity, robustness, reliability, and more importantly for Mars missions its capability to process high power levels at moderate voltages in the I_{sp} range of 1500—3000 s. Indeed, in the lower end of the range, thrust-specific power (or the power cost of creating the thrust) can be lower than 18 kW/N. Conversely, this means more thrust per unit power (over 55 mN/kW), a benefit when the mission has a time (or minimum acceleration) constraint at a given power level.

Gridded-ion thrusters also have been demonstrated in space, with most notably the NASA Deep-Space One demonstration mission where the NSTAR engine was operated for more than 16000 hrs. This technology was also flown on the ESA Artemis demonstration mission, and is also penetrating the commercial space market on board the Boeing HS-601 and HS-702 platforms for both orbit topping and station keeping duties. Finally, the on-going JAXA (Japanese Aerospace Exploration Agency) Hayabusa mission, powered by four ion engines, is scheduled to reach asteroid Itokawa in June 2005 and return samples to Earth in June 2007. GIT technology is well suited for high to very-high I_{sp} missions, such as missions to the outer solar system or missions with little time constraints. Typical gridded-ion engines can readily yield 3500 to 6000 s of I_{sp} [6]. Conversely, this also means that the power cost of generating the thrust is high (over 30 kW/N). This technology also offers a greater thrust efficiency above 3000 s of I_{sp}, *i.e.*, typically higher than 60%. On the other hand, grid expansion and alignment constraints place a practical limit on thruster sizes to about 40 cm in diameter. As a consequence, the power density (in kW/cm²) needs to increase as power level increases. Because the space-charge limitation applies in the non charge-neutral inter-grid region, this can only be done by increasing accelerating voltage, and hence the specific impulse. Thus, missions requiring a power greater than 10 kW per thruster may use GITs only if the required optimum specific impulse is large enough: from about 5000 s at 20 kW to about 8000 s at 50 kW, 9000 s at 500 kW or 15000 s at 1.5 MW, if xenon—a heavy element—is used as propellant [6]. The respective beam acceleration potentials are then 2500 V, 5000 V, 6000 V, and 18000 V.

Unlike the previous two technologies, where acceleration of the working fluid—positive ions—is electrostatic (or quasi-electrostatic in the case of HETs), the MPDT technology uses the electromagnetic force to accelerate elementary volumes of plasma, an effect that can be enhanced in some devices by gasdynamic acceleration. The optimal power range of so-called applied-field MPDTs is 10—100 kW, whereas so-called self-field MPDTs operate best in the 1—10 MW range. Interesting features of MPDTs include the ability to operate on a large variety of propellants, good scaleability and throttleability, with possible values of I_{sp} in the range 1000—4500 s [7]. The high power density achievable by such devices make them a candidate disruptive technology for NEP missions. The maturity of this technology is not, however, as advanced as for the previous two thruster technologies: issues such as cathode life, onset of unstable discharge phenomena, magnetic coil technology (for applied-field MPDTs), propellant management and feed systems for alkali metals, and predictive numerical modeling [7], still need to be addressed before a first flight demonstration mission can be envisioned.

3 - THE RIGHT SELECTION OF THE PROPELLANT

Due to the large mass of gas needed to perform the foreseen missions, the classical xenon rare gas is no more in competition, the total world production being limited to 20 –40 tons per year (L'Air Liquide accounting with nearly 30% of the world's supply, BOC accounting for nearly 20% of the world's supply and about 12 other suppliers)[8].

Thus, the right selection of the propellant feeding the thruster is of prime importance.

When choosing a propellant with lower molecular mass, the consequence at thruster level is an increase of the specific impulse, but at the electric propulsion system level the dead mass can increase due to the increase of the mass of the propellant system tanks.

In the course of previous studies, the so called optimum storage pressure with respect to the minimum of mass (taking into account the real gas characteristics) was found to be around 120 bar for xenon into full metallic tanks, 150 bar with xenon into composite tanks. For the current paper, the composite tanks are considered and the range of Maximum design pressure (MDP) from 15 to 30 MPa is covered. The optimum tank pressure of various gas is presented in Figure 4 . The gas considered are : Xe, Kr, Ar, Ne, He, with the following assumption: Minimum usable pressure= 0.3 Mpa, Burst pressure = 2 x MDP.

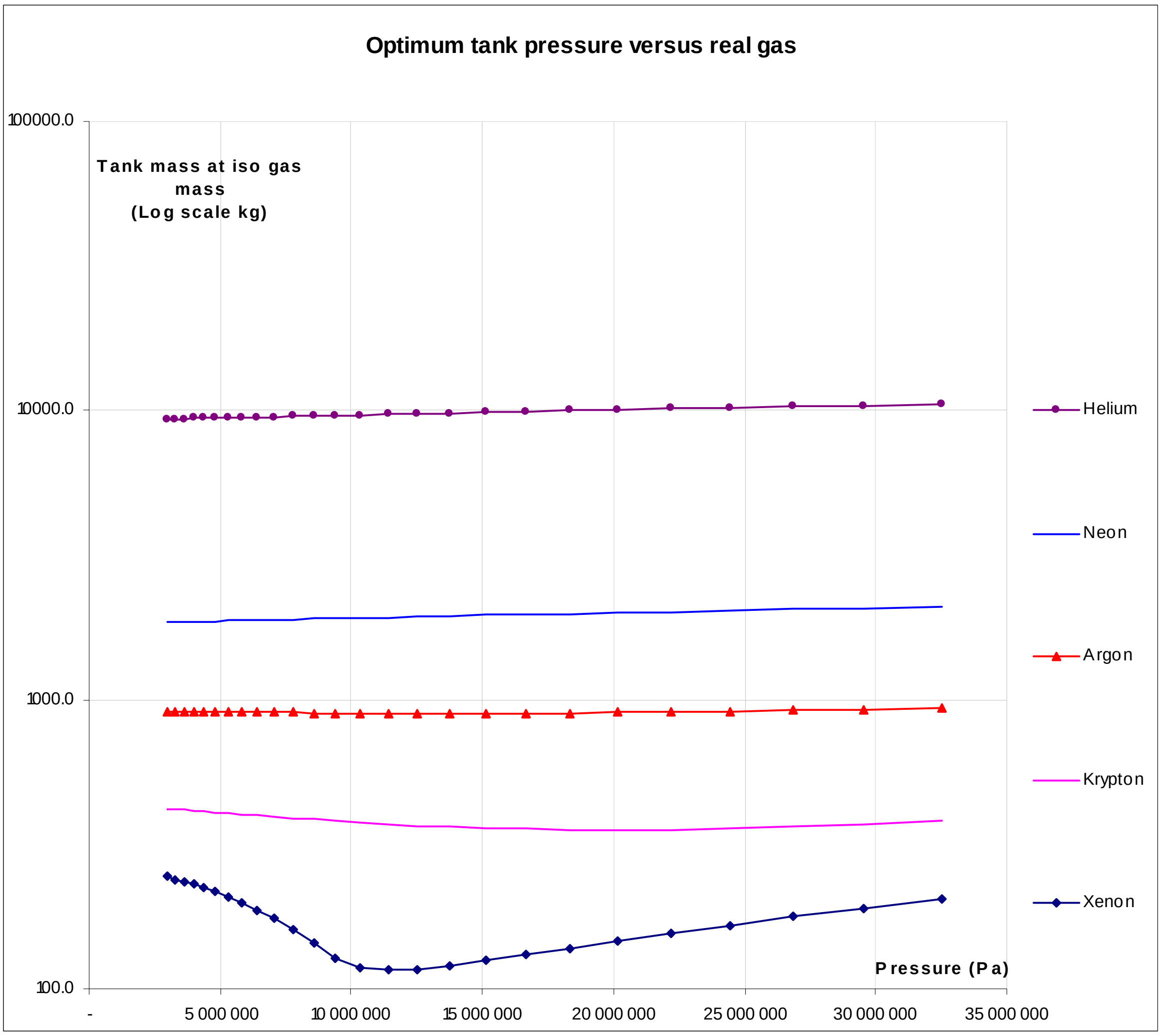


Figure 4 ***Optimum gas pressure for various gases in composite tanks.*** The xenon presents a real optimum around 15MPa, krypton optimum is quite flat, while for the other gas the minimum pressure is recommended when considering only the mass criteria.

Once the optimum pressure wrt the mass is chosen, the performance of an electric propulsion system can be assessed roughly by using the concept of System specific impulse introduced by Erichen [9] and Garisson [10], see equation (4), or a more accurate analysis can be performed using the real equation that drive the design of the EPS.

$$\boxed{Issp = \frac{Isp_{thruster}}{1 + k_{\tan k}}} \quad \textbf{(4)}$$

where k_{tank} is the ratio of the tank mass by the propellant mass inside the tank.

3.1 - System Specific Impulse Approach

The first trade-off between various gases is performed on the basis of the System specific impulse approach. The results are presented in the next Table 2.

The specific impulse of the thruster is taken for a first order approximation, simply proportional to the inverse of the molecular mass square root, according to equation (5).

$$g_0.Isp = \eta_{jet}.\sqrt{\frac{F_a.U_{accd}}{2.M}} \qquad \textbf{(5)}$$

with $F_a = \text{Faraday charge (C)}$; $M = Molecular\ mass$ (kg);

U_{accd} the discharge acceleration potential (V) and η_{jet} the efficiency. The tank mass ration « k » is coming from the previous analysis giving the optimum pressure.

Table 2 « Fast Trade-off between various gas ». Even if the helium thruster specific impulse is very high, at system level, the System specific impulse is decreased to a very low value due to the huge specific mass of the required tank to contain the required gas mass.

	Xenon	Krypton	Argon	Neon	Helium
k tank mass ratio (kg/kg)	0.12	0.35	0.90	1.85	9.35
Mass mol (kg)	0.131	0.084	0.040	0.020	0.004
Isp thruster (s)	2000	2503	3626	5101	11454
Issp System Isp (s)	**1 793**	**1 848**	**1 912**	**1 787**	**1 107**

The Table 2confirms that, if the ionisation efficiency differences between the gases can be neglected, the various gases Xenon, Krypton, Argon and Neon can be considered as very similar on a preliminary system point of view, which is the real propulsive performance of the system.

The advantage of the system specific approach is that the answer to the question regarding the trade-off can be answered very rapidly, but on the other hand, one can only eliminate the very bad solution like here, one can eliminate the gas helium, because a part of the dry mass is considered as a propellant. To select the “best choice“ for a particular mission, a complete analysis has to be performed.

3.2 - COMPLETE ANALYSIS FOR A PARTICULAR MISSION

The mission considered is to provide a delta V =10 km/s to a dry spacecraft mass of 10 tons (excluding the dry mass of the tanks). The tank mass ratio is taken from the table above, as well as the specific impulse of the thruster with the same assumptions.

The results of the classical analysis are shown in the following Table 3.

Table 3 « Classical Trade-off between various gas ». The results are very similar to the table above: the launch mass for the case helium is rather prohibitive. However, the best choice (minimum of launch mass) is obtained for the xenon.

	Xenon	Krypton	Argon	Neon	Helium
M payload, dry w/o tanks (kg)	10 000	10 000	10 000	10 000	10 000
Delta V (m/s)	10 000	10 000	10 000	10 000	10 000
Isp thruster (s)	2000	2503	3626	5101	11454
M gas (kg)	7 204	6 118	4 582	3 752	7 202
M tank /M gas (kg)	0.12	0.35	0.90	1.85	9.35
M tank (kg)	833	2 168	4 108	6 956	67 344
M launch (kg)	**18 037**	**18 286**	**18 691**	**20 708**	**84 546**

Table 3confirms the results presented in the previous paragraph. In addition, the launch mass considered as the parameter of prime importance for the trade-off is here minimum for the xenon gas. But the performance with Krypton, Argon and Neon can be considered as very near to the one of the xenon. Even if the specific impulse for the same thrusters technology (here Plasma HET) increases from 2000 s to 5100 s, the launch mass remain roughly the same. One shall note that the analysis does not consider the increase of electrical power needed by the higher specific impulse thruster to provide the same thrust (see first section above). That means that the so-called parameter “alpha (kg/kWe) “ is here taken as zero.

If alpha were taken at its real value like 16 or 20 kg/kWe, then the table would have shown naturally a very large disadvantage for the high specific impulse solutions using lower molecular mass instead of the classical xenon.

The question is now to propose an attractive way to really take advantage of the higher specific impulse provided by the gases with lower molecular mass than xenon.

Other alternatives, in disruption with respect to the current technologies, are presented in order to make the whole system more attractive.

3.3 - Disruptive low temperature tankage

The mass fraction of the tankage for Krypton and Argon become really very expensive at ambient temperature: 0.35 kg/kg or 0.9 kg/kg.
The low temperature storage of those gases may transform the usual pressure vessel into a simple container dimensioned for a low pressure and high density liquids(2413 kg/m^3 for Kr; 1392.8 kg/m^3 for Ar). However, a special refrigerator shall be designed in order to maintain a low temperature (boiling point at 119.7 K for Kr, 87 K for Ar), and to follow the concept so called “zero boil-off” (that maintains the low temperature of the liquid or solid by an active refrigerator device, without any loss of propellant). The preliminary figures of performance for such systems are very promising.

However, the real choice of the right gas to feed the electrical high power thrusters will depend also on the availability of ground tests facilities and the capability to maintain the required vacuum level without increasing hugely the size of the vacuum chamber. Condensable materials like caesium are attractive for such reasons.

4 - TRAJECTORY OPTIONS

The trajectories taken into account in the paper are constrained by the allowable duration of the travel and the launcher size.

4.1 - Simple orbit transfer to Mars, using electric propulsion

The first simple orbit transfer to Mars, using electric propulsion almost continuously, can be described as

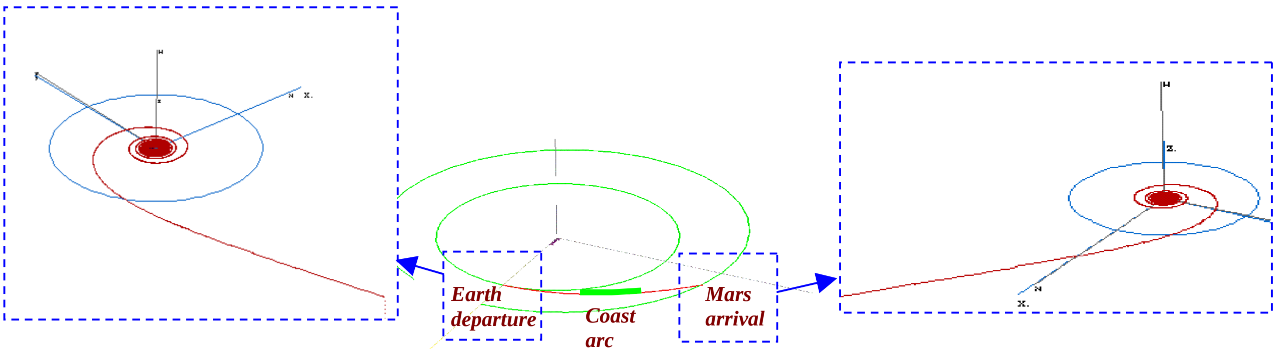


Figure 5 ***Simple mission from LEO to LMO with Electric propulsion.*** The electric propulsion is used continuously at departure as well at arrival.

A. starting from LEO,

B. perform a continuous thrust acceleration for a spiraling orbit around the Earth until the escape velocity is reached with the right declination. *Note:The optimum spiral may led to a slightly different shape for the last orbit before escape* [12] *in order to have more efficiency.*

C. continue the thrust acceleration on the heliocentric orbit until the transfer orbit to Mars is reached

D. perform a ballistic phase (coast arc)

E. perform a continuous thrust acceleration on the heliocentric transfer orbit for reaching the Mars heliocentric velocity (ie, reaching just the Mars escape velocity with the right declination)

F. perform a continuous thrust deceleration for a descending low eccentricity spiraling orbit around Mars until the LMO altitude is reached.

The deltaV to achieve from LEO (1000 km) to LMO (450 km) are the following:

Theoretically: (B) 7.35* +(C) 2.94** + (E) 2.64** + (F) 3.4* = $\boxed{16.4\ \text{km/s}}$

Note: *(Edelbaum formula),; ** Mean Earth and Mean Mars orbits, heliocentric mean duration = 8.4 months.

However, the integration process of the equations of the spiraling trajectory from the Earth to the heliocentric transfer orbit shows a substantial decrease of 1.7 km/s of the deltaV with respect to the corresponding theoretical values. This is due to the fact that the integration takes into account the real distance from the focus and not an infinite distance as in the theoretical approach, while the spacecraft pass the parabolic speed. When considering in addition a similar behaviour at Mars arrival, the mean value of the total delta V needed for the electric propulsion used continuously except for the coast arc is :

Integration : (B) 6.6 +(C) 2.0** + (E) 1.8** + (F) 3.1= $\boxed{13.5\ \text{km/s}}$

Note ** Mean Earth and Mean Mars orbits, heliocentric mean duration = 8.4 months..

The high value considered here shows that the first simple orbit transfer to Mars is probably not optimal, even if it is considered as the fastest mission.

- The cost of the phase (B) can be considered as prohibitive taking into account the fact that for impulsive manoeuvres, the required deltaV is only 3.1 km/s instead of 6.6 km/s .
- Moreover, the impulsive manoeuvre to a direct injection into the heliocentric transfer orbit (i.e. a manoeuvre that combines the phase (B) and the phase (C)) is only 3.45 km/s instead of 8.6 km/s --due to the total specific energy “vis viva” equation (kinetic and gravitational), it easier, in terms of delta Velocity, to increase the total specific energy (i.e. the $V_{infinity}^2/2$) by increasing the kinetic energy as near the planet as possible because the Velocity is at its highest value, and it is raised at the power 2 for the kinetic energy--).
- Thus the idea has been to propose a manoeuvre that needs a deltaV as close as possible to the one needed for the impulsive case, but suited for electric propulsion characterized mainly by its low thrust.

4.2 - OPTIMISED DELTAV ORBIT TRANSFER TO MARS, USING THE ELECTRIC PROPULSION

The solution of the problem is the multi-arcs trajectories to Mars.

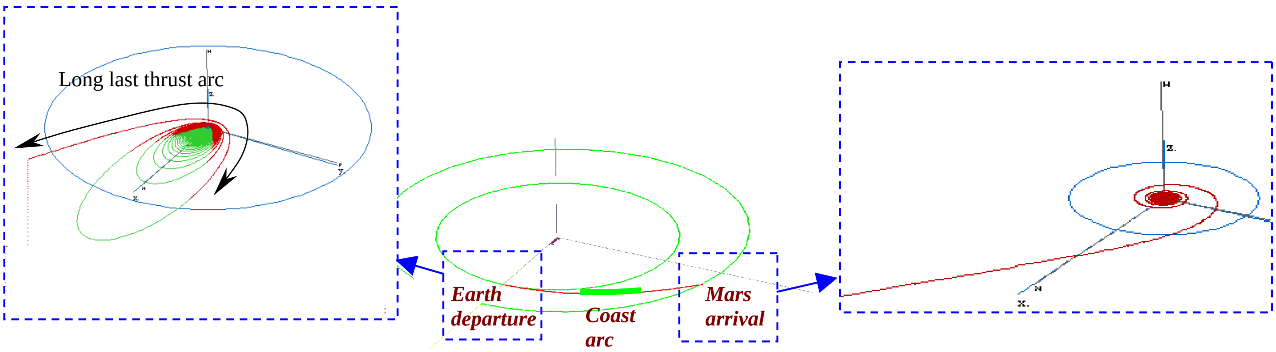


Figure 6 ***Mission from LEO to LMO with Electric propulsion.*** The electric propulsion is used along arcs at departure and continuously at arrival.

This orbit transfer to Mars can be described as

I. starting from LEO,

II. perform thrust arcs acceleration with electric propulsion around the perigees in order to reach a very high elliptic orbit around the Earth,

III. After rendez-vous with the possible shuttle passenger, perform a last long thrust arc around the perigee until the escape velocity is reached,

IV. continue the thrust acceleration on the heliocentric orbit until the transfer orbit to Mars is reached

V. perform a ballistic phase (coast arc)

VI. perform a continuous thrust acceleration on the heliocentric transfer orbit for reaching the Mars heliocentric velocity (ie, reaching just the Mars escape velocity)

VII. perform a continuous thrust deceleration for a descending low eccentricity spiraling orbit around Mars until the LMO altitude is reached.

The main difference with the previously presented simple strategy resides in the first phase of the transfer. The particular phase (III) has been studied in details see Figure 7 .

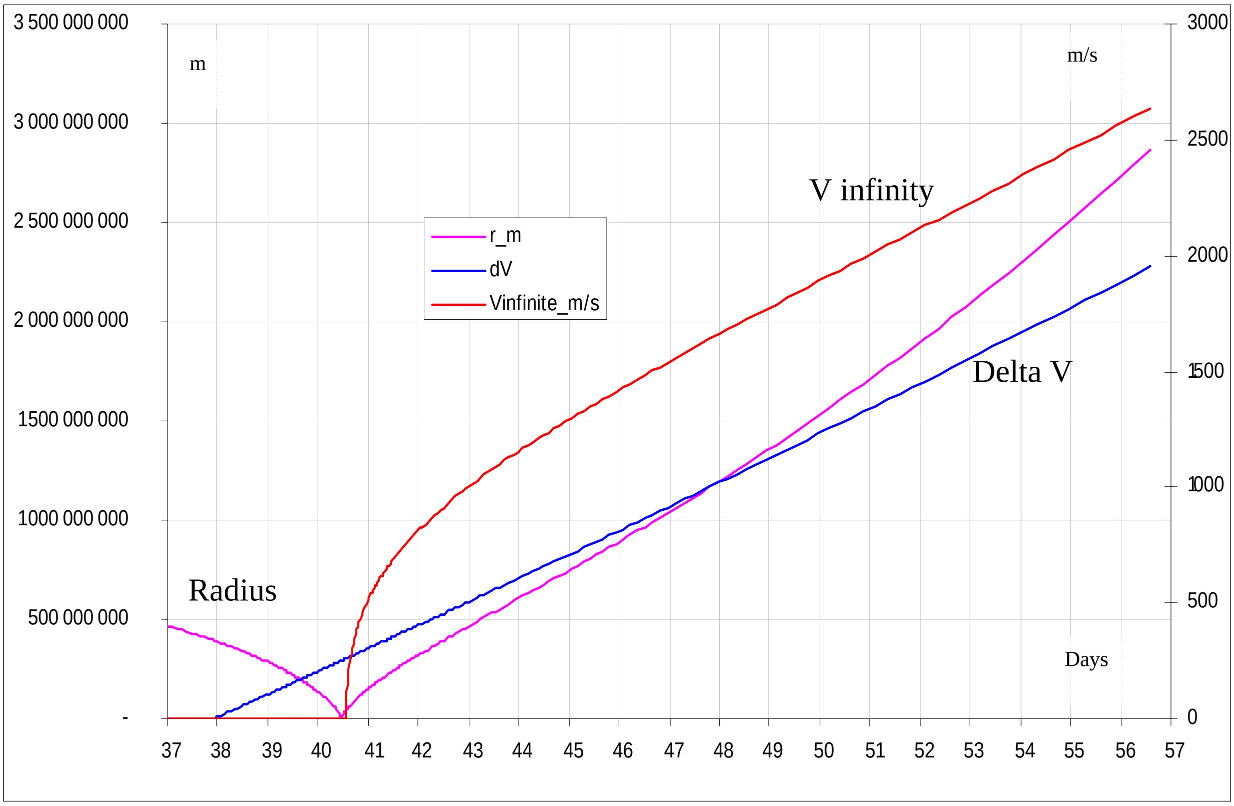


Figure 7 ***Delta V to be provided by the low thrust propulsion to reach the Mars orbit transfer from a high elliptic Earth orbit.*** The velocity infinity (escape velocity) of the hyperbolic branch starts to be positive just at the lowest radius (perigee), the location of highest efficiency to increase the V infinity.

The deltaV to be achieved from LEO (1000 km) to LMO (450 km) are the following:

<u>Integration</u> : (II) 3.54 +(III+IV) 2.4** + (VI) 1.8** + (VII) 3.1 = $\boxed{\text{10.84 km/s}}$

Note ** Mean Earth and Mean Mars orbits.

In order to increase the performance of the strategy in terms of delta V, an optimized combination of chemical and Electric propulsion can be proposed for performing the particular phase (III).

The compatibility with NEP systems that implies orbiting a sizeable nuclear reactor and a power generation system capable of converting thermal into electric power, with minimum mass and volumes fitting in with Ariane 5 or the Space Shuttle bay, is assessed.

4.3 - Optimised deltaV orbit transfer to Mars, using the combination of electric propulsion and chemical higher thrust propulsion

The last arc of the thrust around the Earth is provided with a higher thrust (chemical) propulsion. This is a very interesting solution of the multi-arcs trajectories to Mars.

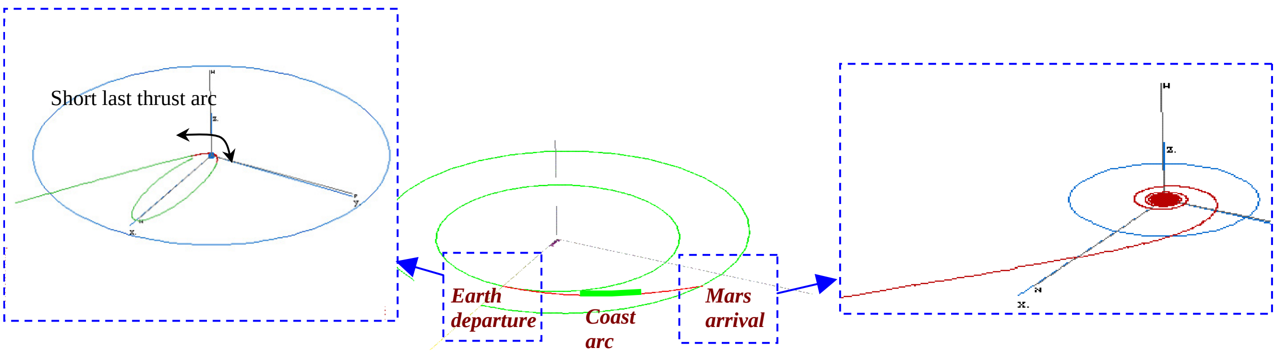


Figure 8 ***Mission from LEO to LMO with Electric propulsion and higher thrust chemical propulsion.*** The electric propulsion is used along arcs at departure followed by a short arc of higher chemical thrust, and electric propulsion is used continuously at arrival.

This orbit transfer to Mars can be described as

I. starting from LEO,
II. perform thrust arcs acceleration with electric propulsion around the perigee passes in order to reach an elliptic orbit around the Earth (less elliptical as before),
III. After rendez-vous with the possible shuttle passenger, start with the higher thrust, a last thrust "short" arc before the perigee until the escape velocity is reached,
IV. continue the thrust acceleration around the perigee until the transfer orbit to Mars is reached
V. perform a ballistic phase (coast arc)
VI. perform a continuous thrust acceleration on the heliocentric transfer orbit for reaching the Mars heliocentric velocity (ie, reaching just the Mars escape velocity)
VII. perform a continuous variable thrust deceleration for a descending low eccentricity spiraling orbit around Mars until the LMO altitude is reached.

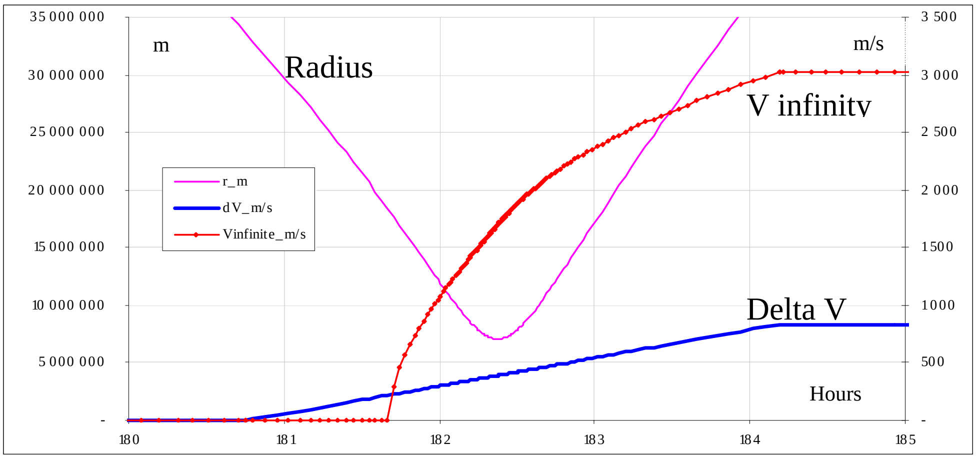


Figure 9 ***Delta V to be provided by the low thrust propulsion to reach the Mars orbit transfer from a high elliptic Earth orbit.*** The velocity infinity (escape velocity) of the hyperbolic branch starts to be positive around the lowest radius (perigee), the location of highest efficiency to increase the V infinity.

The main difference with the previously presented arcs strategy resides in the last orbit of the transfer, performed with a higher thrust (chemical) propulsion. The last arc (III) has been studied in details, the timescale being in hours now instead of days as before with low thrust propulsion.

The deltaV to be performed from LEO (1000 km) to LMO (450 km) are the following:

Integration : (II) 3.44 +(III+IV) 0.83** + (VI) 1.8** + (VII) 3.1

= $\boxed{8.34\ \text{km/s}}$ for EP and $\boxed{0.83\ \text{km/s}}$ for chemical propulsion

Note ** Mean Earth and Mean Mars orbits.

The figures presented above show a reduction by a factor 2 of the overall deltaV to be imparted to the spacecraft to go to Mars. From the time of the rendez-vous in elliptical orbit with a "manned shuttle" to the time of the arrival to Mars, the overall mission appears shorther than with an all electric propulsion system.

5 - CONCLUSION

This paper presented the main features of a high-power electric propulsion sub-system for space exploration missions, with emphasis on Mars missions. The key disruptions in technology will come from high-power, high thrust-specific power engines; novel propellants and storage devices; and proper trajectory analysis to optimize the efficiency of the thrust arcs.

Among the key parameters to be considered in the trade-off, the optimum specific impulse (proportional to thrust-specific power, in kW/N) is strongly dependent on the power sub-system technology, expressed in terms of power-specific mass α (kg/kWe). The lower the specific mass, the higher the optimum specific impulse. For current and mid-term values of *a*, the optimum I_{sp} , which is comparable to the ΔV to be provided by the propulsion system, is not very high: moderate values are sufficient, *i.e.*, 1500 sec for a ΔV of 11 km/s or 2000 sec for a ΔV of 22 km/s. Such values of specific impulse can be readily achieved by Hall-effect thrusters, although higher-power devices than the current state of the art will need to be developed for future heavy cargo missions.

In order to enhance the overall performance of future high-power electric propulsion sub-systems and to enable higher-energy missions such as (fast) human missions to Mars, a disruption in terms of propellant must be brought about. The discussion included in this paper showed that krypton and argon may be considered as good candidates to be substituted to xenon: there are no severe constraints due to limited world production, unlike for xenon. They may be stored at cryo-temperatures, provided appropriate refrigerators are designed. Otherwise, using condensable propellants is another option to mitigate the mass of the propellant storage components. For Hall thrusters, the increase in specific impulse brought by the use of a lower-atomic mass gas is in line with the values of optimum specific impulse applicable for missions with ΔVs of 11 km/s to 22 km/s. Gridded-ion thrusters, on the other hand, already typically operate at values of I_{sp} which are above the optimum for the missions considered here, so that a further increase in I_{sp} due to a lower-atomic mass working fluid is not beneficial.

Finally, the maneuvers to be performed for Earth-Mars transfers and Mars orbit insertion should be optimized for the use of electric propulsion. Promising disruptive strategies, involving a combination of electric propulsion with higher-thrust chemical propulsion, led to significantly reduce the penalty in ΔV with values to be provided as low as 8 km/s; *i.e.*, less than half the total value when compared to an all-low-thrust trajectory. This approach is fully consistent with the use of high-power electric propulsion based on Hall-effect thrusters.

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