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## ANALYSIS OF A MANNED MARS MISSION WITH NUCLEAR ELECTRIC PROPULSION (NEP) SYSTEM

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## ABSTRACT

The goal of this paper is to analyze the performance of a ion NEP system for a manned mission to Mars. The objectives are to evaluate whether NEP could reduce journey's time and mass of propellant compared to both chemical and Nuclear Thermal Propulsion (NTP). To these authors' knowledge this is the first time that a manned Mars mission using only NEP has been analyzed in detail.

The trajectory assumed consists in an acceleration spiral to leave LEO, followed by acceleration with constant thrust after reaching escape speed. Thereafter the spacecraft decelerates until reaching the  $\Delta V$  needed to capture a Low Martian Orbit (LMO) through a spiral. This trajectory was calculated for thruster power varying from 100 to 300 MW. Mission times are found to depend strongly on power. The gain in shortening times more than compensate for the increase in the mass budget.

Reactor sizing has been done using Los Alamos National Laboratory realistic data for Particle Bed Reactors. The shield uses Be, LiH and W and its architecture has been designed based on a simplified model obtained with the neutronic MCNP code. After optimization, reactor mass, including shielding, is estimated between 4600 kg (333 MWth), to 5370 kg (1000 MWth).

The NEP system consists of a cluster of 20 to 60 ion thruster modules, depending on power, each with thrust 15.55 N, beam diameter 0.4 m and power 5 MWe. Total thruster power assumed after thermal to electric conversion is 100 MWe (150 MWe, 200 MWe and 300 MWe). Specific impulse with Xe propellant is 32,780 s, much higher than that of a NTP system using hydrogen.

The spacecraft mass budget found at convergence includes structure, propulsion system, propellant, environmental control and life support system, water, avionics, crew, crew accommodation (including food), and modules for Mars landing and travel back to Earth. The conclusions indicate that NEP is likely a serious contender for more economical manned Mars missions.

## INTRODUCTION

Although the recently released Augustine Commission report takes a rather negative view of the economic feasibility of a US-driven Manned Mars Mission (M3) in the next decades with current NASA budgets, the interest in Mars exploration remains high.

M3 have been considered for a long time since the Von Braun-inspired "Collier's" article of the '50s. Recent ones are the NASA Mars reference mission [1] and the very detailed Russian study [2] commissioned by ESA, and including nuclear electric propulsion (NEP) options. More recently, the ESA Future EU Space Architecture presented in 2009 investigated chemical as well NP alternatives, but only as a system study, not in any propulsion detail. Among its findings is that NP may become a key requirement for human missions to Mars.

It is now apparent that galactic and cosmic radiation (GCR) is probably the bottleneck of human interplanetary travel. The energy spectrum of GCR is such as to pose excessive risks for extended space travel [3;4]. Countermeasures based on shielding or MHD are commonly thought not to be effective in view of the energy composing the GCR spectrum [5] and reaching  $> 10^{15}$  eV.

Basic physics tells that the radiation dose is (flux) x (time). Not being able to reduce the flux below a safe threshold (of order 1-5 mSv/yr, vs. 200 to 650 mSv/yr for GCR, depending on solar cycle) implies the only alternative is to reduce travel time.

Conventional chemical propulsion (CP) has vacuum specific impulse  $I_{sp} \approx 450-460$  s. A reasonable mass ratio (MR) forces therefore Hohmann trajectories, with round trip times lasting 3-4 years or more, depending on Mars stay time and ephemerides (position relative to Earth). Hence the interest for NP,

since in principle the Isp (equal to exhaust velocity for an ideal expansion) is potentially much higher than that of CP. In fact, to 0-th order, the potential energy E and the exhaust velocity  $V_e$  (=Isp for ideal expansion) of a rocket are linked by

$$E = \frac{1}{2} m (V_e)^2$$

Thus Isp scales as  $(E/m)^{1/2}$ . Chemical E/m is limited to about  $10^7$  J/kg, while nuclear is about a factor  $10^6$  to  $10^7$  larger. Thus the Isp in NP could be of order  $10^3$  larger, with a corresponding increase in MR. In addition, if thermodynamic acceleration (e.g., as in thermal NP) is replaced by Coulomb or Lorentz acceleration (as in gridded ion engines and MPD thrusters)  $V_e$  (and Isp) limitations due to thermodynamics and materials may be removed [5;6]. Of course, because the product thrust times Isp is proportional to the exhaust (jet) power, at fixed reactor power raising Isp means lowering thrust. Thus the trade-off is between high thrust (controlling mainly travel time) and low Isp, or (vice-versa) high Isp (controlling MR) and low thrust. The particular mission will determine the choice. For a M3 where trip time must be as short as possible cost will be the main driver, as spacecraft and all auxiliaries (including any cargo) must be lifted to LEO, a very costly proposition.

The goal of this paper is, in fact, to investigate whether a M3 is feasible with NP, based on the mass and power budgets resulting from using this technology, in particular NEP. Compared to the ISTS Russian study mentioned (assumed power  $\sim 10$  MW), here power is in the O(100) MW. This analysis was inspired by a 2008 study led by A. Lorenzoni (at the Italian Space Agency ASI), where two student teams at the School of Aerospace Engineering of Rome compared M3 using CP and thermal NP (paper IAC – 09.C4.7. - C3.5.2 reports their findings). Here the same M3 is analyzed using NEP, to compare the mass budget and performance; to these authors' knowledge, this is the first time that a M3 is analyzed using NEP as the *only* propulsion means.

Because of time and space limitations, only the main details of the study are reported here, among them electric thruster sizing in Section 1, nuclear reactor sizing and power in Section 2, the powered trajectory (that is, non-Hohmann) in Section 3, and the mass budget in Section 4. Final considerations attempting a comparison with CP are reported at the end of this paper.

## 1. NEP: POWERED PROPULSION SYSTEM

### 1.1 Advantages of Electric Propulsion

Electric propulsion (EP) thrusters, and ion thrusters in particular, are a propulsion technology already used for sending spacecraft to the Moon (ESA's Smart-1 mission) and to the asteroids (JAXA's Hayabusa mission). The methods for ionizing the propellant and accelerating ions vary, but all take advantage of the Coulomb force. Even relatively small potential differences can create high velocity exhaust gases, resulting in large Isp. Accordingly, mass or propellant requirements drop according to the Tsiolkovski's equation (rocket equation)

$$\Delta V = v_{eff} \log_e \left( \frac{M_o}{M_f} \right) = v_{eff} \log_e \frac{M_o}{M_o - \Delta M}$$

(1.1)

where  $v_{eff}$  is the *effective* exhaust velocity, or Isp,  $M_o$  and  $M_f$  are the masses of the spacecraft at the beginning and end of the maneuver, and  $\Delta M = M_o - M_f$  is the mass of propellant consumed. The rocket equation illustrates the importance of achieving high  $v_{eff}$ , providing adequate power is available. For a given mission  $\Delta V$ , low exhaust velocity must be compensated by large  $|\Delta M|$ , implying a large increase of vehicle size and cost. It is the dependence of  $\Delta V$  on the logarithmic term in the above equation (for constant  $v_{eff}$ ) that has driven EP development. In terms of Isp, comparing EP with the best cryogenic chemical propulsion (CP) shows very clearly the superiority of the former, e.g., see [6]. At the same time, raising Isp increases the amount of power of the exhaust ion jet (scaling with the cube of Isp) that must be supplied by an *external* power source. This escalates rapidly with Isp and with the thrust required by the mission.

### 1.2 Gridded Ion Engines (GIE)

In the vast area of EP technologies, that based on Coulomb acceleration of a ionized propellant is arguably the more mature. Ion engines consist of a discharge chamber closed at one end, and with a set of perforated and accurately aligned grids at the other, hence their Gridded Ion Engine (GIE) name. In their discharge chamber gaseous propellant is injected and ionized, forming a plasma. The positive ions in the plasma diffuse towards the grids and are extracted and accelerated by the potentials relative to the grid(s). The space charge of the emerging ion beam is neutralized by electrons provided by an external cathode (the so-called neutralizer). Various ionization strategies characterize different GIEs (see [6], for a thorough review, used here to report key aspects of GIEs).

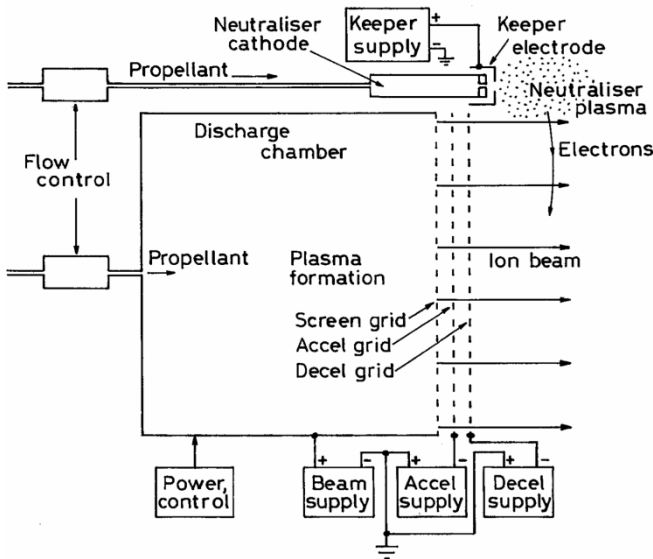


Figure 1. Schematic diagram of a GIE [6]

A triple-grid system is typical of current GIEs, see Figure 1. It operates in an accelerate/decelerate mode to enhance ions throughput at the voltages employed. In Figure 1 the thruster body is at a potential  $V_B$  appropriate to the ion beam velocity ( $I_{sp}$ ) desired. This is thus the potential of the inner, or screen, grid. The next grid, the acceleration, is at a negative potential to focus each single ion 'beamlet' and to enable its extraction. Note that each beamlet is in fact an electric ionic current. Ions are slightly decelerated to the space potential, via a low negative voltage applied to the outer deceleration grid that couples the ion beam to space plasma [6].

The external neutralization of the ion beam is done by a weak external plasma created by means of a DC discharge between the neutralizer cathode and an adjacent keeper electrode closely coupled to the space plasma. Electrons are extracted from this plasma by the ion beam as necessary to neutralize its space charge. This is a natural process involving no active control [6]. An important function of the outer grid is to minimize the erosion of the accel grid due to the bombardment by low energy charge-exchange ions. These ions are generated by interactions between beam ions and the neutral gas atoms, both escaping from the discharge chamber and also created by the neutralizer [6].

A major advantage of GIE is that plasma production, ion extraction and acceleration, and ion beam neutralization are separate processes, simplifying GIE design. As shown later, further separation of functions is possible in a 4-grid configuration for ultra-high  $I_{sp}$  operation, where ion extraction and acceleration can be examined independently [6].

### 1.3 Engine Parameters

GIE do have limitations dictated by physics and technology. Some [6] are reported in Table 1.1:

Parameter	Lower limit	Upper Limit	Comments
Beam diameter (cm)			
Beam diameter (cm)	~ 3	~ 40	Upper limit due to grid distortion
Beam accelerating potential (V)	10	70,000	Values > 5 kV need 4 grids
Beam current density (mA/cm <sup>2</sup> )	-	150	Determined by perveance
Grid perveance factor, $P_g$ (*)	-	4500	Limit determined by Qinetiq's T5 thruster
Thrust density (mN/cm <sup>2</sup> )	-	30	Determined by perveance
Power density (W/cm <sup>2</sup> )	-	4500	
Specific impulse (s)	2000	150,000	Determined by max voltage, assumed to be 70 kV, and ion mass
Mean ion mass (AMU)	~ 4.5	~ 200	Minimum value is hydrogen compound, maximum is mercury
Thruster temperature for Kaufman-type, RIT-type and ECR thrusters (deg C)	-	~ 800	Determined by solenoid and RF coils, also wiring
Thruster temperature for MESC thrusters (deg C)		~ 300	Determined by permanent magnets

Table 1.1 Some current parameter limits. Adapted from [6]

(\*) The perveance parameter,  $P_g$ , perveance per unit

$$area, \text{ is defined as } P_g = \frac{I B m_i^{1/2} d^2}{A T_g} = \frac{4 \epsilon_0 \sqrt{2} e}{9} V_T^{3/2}$$

(1.2)

Perveance is the ability of the grids to handle (pass through) current:

$$Perveance = \frac{I_B}{V_T^{3/2}} = \frac{4\epsilon_0}{9} \left( \frac{2e}{m_i} \right)^{1/2} \frac{AT_g}{d^2} \quad (1.3)$$

where  $V_T = (V_B + |V_{ac}|)$ ,  $\epsilon_0$  is the dielectric constant of free space,  $d$  is the ion acceleration distance,  $A$  is the nominal grid area, and  $T_g$  is its effective ‘transparency’ [6].

### 1.3.1 Beam diameter

The beam diameter of the engine may go from a few cm to a maximum (tested) 40 cm, see Table 1. Up to 50 cm is feasible, but the development of grid systems of this magnitude is difficult, due to problems associated with maintaining grid integrity during launch, when the launcher is subject to vibrations, and during thruster operation, typically at high and variable temperature. Assuming a specific impulse of 32,780th s (the highest tested with Xenon propellant) and assuming high perveance, GIE thrust varies with diameter as in Figure 5.

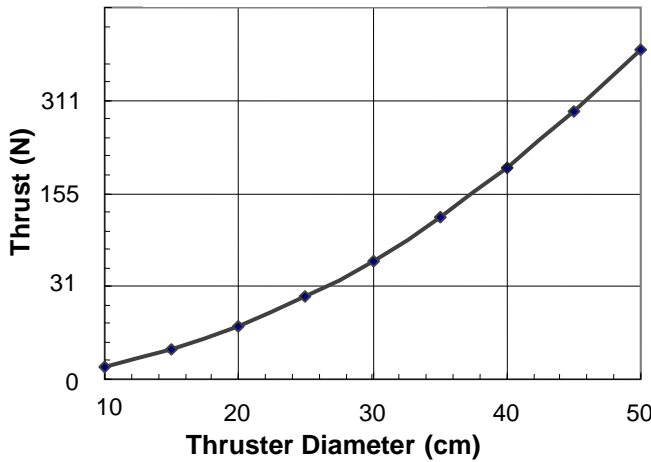


Figure 5 Thrust as a function of thruster diameter [6]

The M3 considered here and the effect of GCR dictate minimum trans-Mars time for astronauts; thus thrust must be maximized, with the [main] constraint that propellant mass to LEO orbit (ultimately, cost) must be capped. Preliminary estimates indicate that thrust should be  $O(10^2)$  N, to be subdivided in a number of GIE modules, each with a [conservative] diameter 40 cm. Note that actual thruster cross section shape may change to compact the entire GIE cluster.

Thrust power (scaling with the product specific impulse times thrust) varies also with the thruster diameter, see Figure 6. A range 100-300  $MW_e$  was therefore assumed and its effect on the present M3 investigated. 300  $MW$

may seem today inordinately large, but note that in the late ‘60s the Phoebus 2A reactor built by LASL was tested at 4.2 GW for 12 minutes.

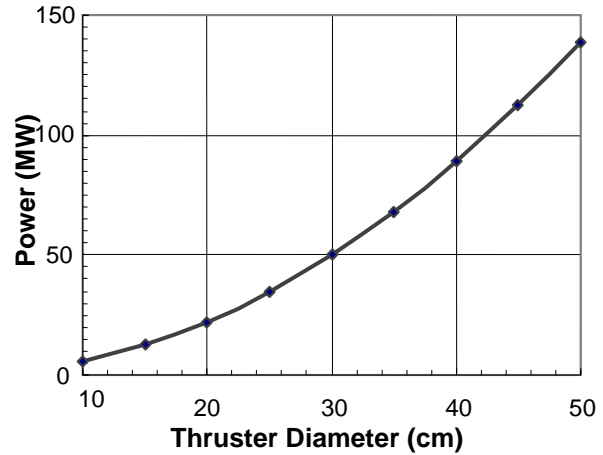


Figure 6 Power consumption as a function of thruster diameter [6]

### 1.3.2 Specific Impulse

The specific impulse, or  $v_e$ , of a GIE depends on the beam acceleration potential and the atomic mass  $m_i$  of the propellant according to

$$v_e = \sqrt{\frac{2eV_B}{m_i}} \quad (1.4)$$

Here the range of  $V_B$  is of order 1-70 kV. Propellants tested, see below, have atomic weight between 1 AMU of hydrogen to 200 AMU of mercury. Most GIE use 131 AMU xenon; and its  $I_{sp}$  is in Figure 7.

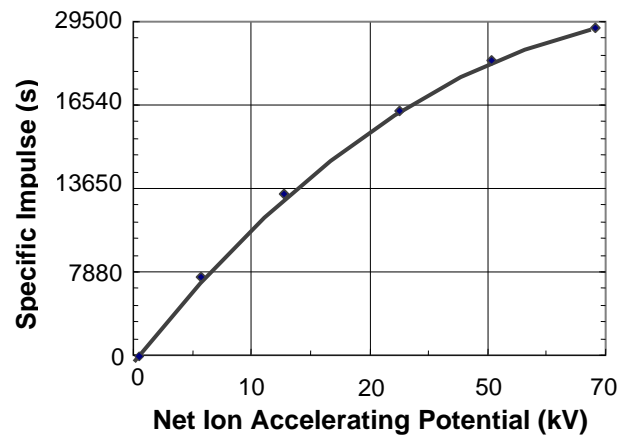


Figure 7 Specific impulse of Xenon as a function of net ion accelerating potential [6]

At fixed atomic mass, the  $I_{sp}$  increases with increasing applied potential, since the potential  $V_B$

between acceleration grids increases. To constrain mass and cost, the engine must have  $I_{sp}$  as high as feasible, so the ion beam acceleration potential assumed here is 70 kV.

### 1.3.3 Propellants

$I_{sp}$  depends on the atomic mass of the propellant. Any inert material easily vaporized and ionized can be used, but currently Xe is the preferred propellant. As a general rule, high  $I_{sp}$  requires low atomic mass. For  $V_B = 70$  kV  $I_{sp}$  varies with propellant as in Figure 8.

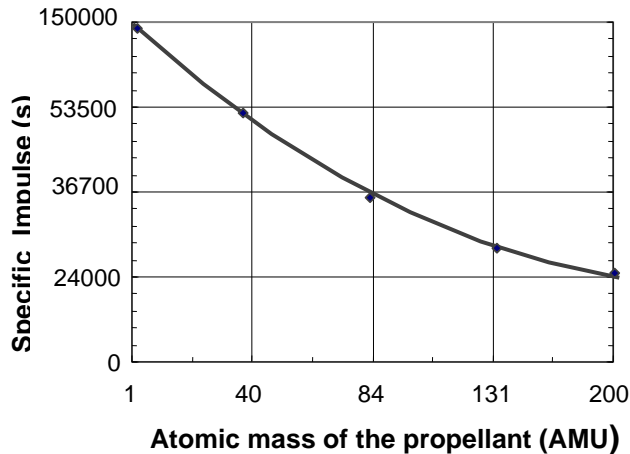


Figure 8 Specific impulse as a function of atomic mass of the propellant [6]

However, thrust increases with increasing atomic mass (*decreasing*  $I_{sp}$ ), see Figure 9:

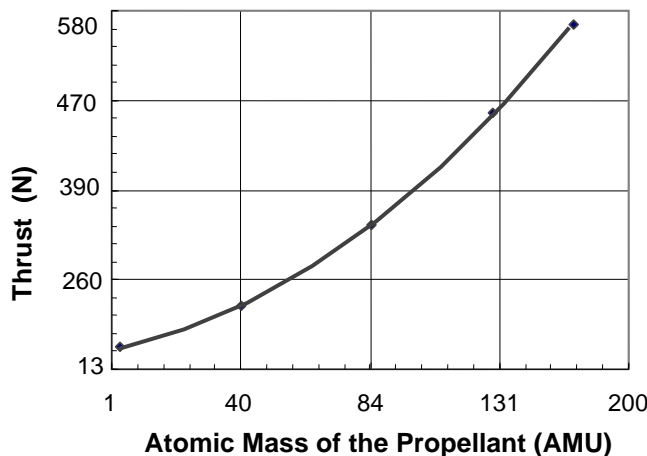


Figure 9 Thrust as a function of atomic mass of the propellant [6]

This trend can be canceled by increasing  $I_B$ , that is, by increasing perveance. At constant  $V_B$ , this can be achieved only by picking a ‘more negative’ grid acceleration, that is increasing  $V_{ac}$  (at the expense of grid life), or by eliminating the inter-grid gap.

$$\text{In fact: } V_T = (V_B + |V_{ac}|) \quad (1.5)$$

$$\text{and } \text{Perveance} = \frac{I_B}{V_T^{3/2}} \quad (1.6)$$

Because thrust increases also with  $V_B$ , the best performance is presumably achieved by combining high values of  $V_B$  and  $m_i$  with high perveance, and using xenon. This choice is a reasonable compromise between thrust,  $I_{sp}$ , power and thus mission time.

### 1.3.4 Effects Due To Perveance

Perveance depends on  $I_{sp}$ : increasing  $I_{sp}$  means also to increase perveance, since both are controlled mainly by  $V_B$ . This said, perveance depends also on the accel grid potential,  $V_{ac}$ , therefore it can also be raised by more negative  $V_{ac}$ , e.g.,  $V_{ac} = -5$  kV. Of course, more negative  $V_{ac}$  will worsen the erosion caused by charge-exchange ions. Allegedly, carbon grids are capable to reduce or solve this problem [6].

The perveance of a GIE grid system is a very important parameter influencing maximum performance for a given ion beam diameter and specific impulse, see Figures 10 and 11:

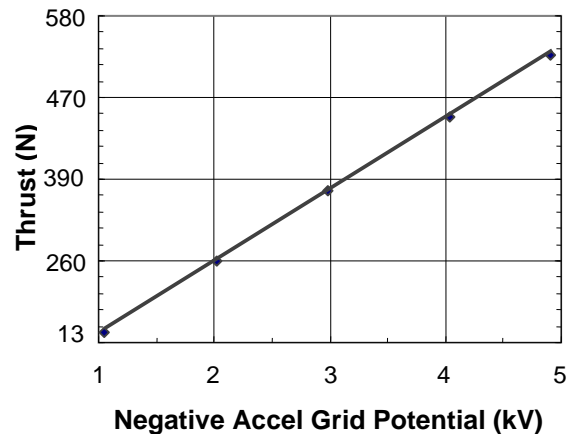


Figure 10 Thrust as a function of  $V_{ac}$  for a 30 cm beam diameter thruster [6]

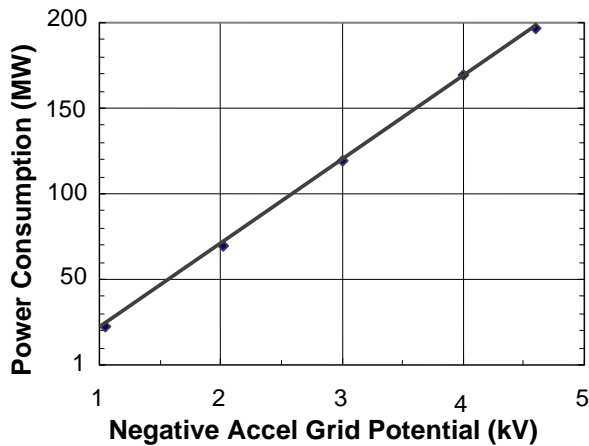


Figure 11 Power Consumption as a function of Vac for a 30 cm beam diameter thruster [6]

### 1.3.5 Power Consumption

With Isp and thrust T appropriate to a M3 the resulting power P (scaling as the product of Isp times T) may be expected to be very large. Electric power scales of course as total voltage times total beam current  $I_B$ . With the numbers estimated or chosen here for the M3, electric power may be in the 100 to 300 MW range. Roughly speaking, the nuclear reactor thermal power may be a factor 3 higher.

For what said about grid size, handling such power in a single engine is unfeasible. Individual GIE modules will have to be assembled in a cluster, a single engine having, say, 5 MW power., for a total of 20 to 60 thrusters. Clustering reduces risks posed by engine malfunctions and enables thrust steering without gimbaling devices.

### 1.4 Final Thruster Sizing

Based on all the considerations made so far, the following set of sizing parameters is derived or assumed

- 1) Ion beam accelerating voltage:  $V_B = 70$  KV
- 2) Accel grid voltage:  $V_{ac} = -5$  KV
- 3) Mass of xenon propellant:  $m_i = 131$  AMU
- 4) Individual beam diameter:  $d = 40$  cm
- 5) overall efficiency:  $\eta_{mi} = 0.9$
- 6) Individual module power consumption:  $P = 5$  MW

Performance calculated for each GIE thruster are:

- 7) Specific impulse:  $Si = 321\ 560$  m/s = 32780 s
- 8) Thrust:  $T = 15.55$  N
- 9) Rate of flow of propellant:  $\dot{m} = 4.8 \times 10^{-5}$  Kg/s
- 10) Ion beam current:  $I_B = 35.7$  A
- 11) Absolute voltage:  $V_T = 75$  KV
- 12) Perveance:  $1.74 \times 10^{-6}$  A/V<sup>3/2</sup>
- 13) Perveance parameter :  $Pg = 4.5 \times 10^{-14}$

- 14) Effective transparency of the screen grid:  $Tg = 500$  mA/Kg<sup>1/2</sup>
- 15) Area of thruster:  $A = 1172$  cm<sup>2</sup>
- 16) Current density:  $JB = 30.46$  mA/cm<sup>2</sup>
- 17) Power density:  $Pd = 4266$  W/cm<sup>2</sup>
- 18) Thrust density:  $Td = 13.26$  mN/cm<sup>2</sup>

The performance of the entire propulsion system is therefore in Table 2

Engine number	20	30	40	60
Power consumption (MW)	100	150	200	300
Specific impulse (s)	32780	32780	32780	32780
Thrust (N)	311	467	622	933
Rate of flow of propellant (Kg/s)	$1.07 \times 10^{-3}$	$1.6 \times 10^{-3}$	$2. \times 10^{-3}$	$3.2 \times 10^{-3}$
Ion beam current (A)	713.4	1070.5	1427	2140.2
Perveance (A/V <sup>3/2</sup> )	$3.5 \times 10^{-5}$	$5.2 \times 10^{-5}$	$7 \times 10^{-5}$	$1 \times 10^{-4}$
Area (cm <sup>2</sup> )	23442	34828	46884	70326

Table 1.2 Performance of the overall propulsion system

## 2. NUCLEAR REACTOR

### 2.1 Dimensioning the core

In this M3 we have chosen a Particle Bed Reactor (PBR) because it is the type of solid-core reactor that shows the highest performance; in fact it is lighter and more compact than a traditional reactor using rods or pins. PBR represent the current best choice fulfilling requirements [7].

The core of the reactor consists of spherical particles of fuel, <sup>235</sup>UC2 or <sup>235</sup>UO<sub>2</sub>, with a diameter of about 500µm. These are contained in a matrix and supported by hexagonal moderator blocks, assembled in a cylindrical configuration. The particles structure consists of a fuel nucleus covered by graphite and by an external inert layer of zirconium hydride. The particles are packed in two coaxial porous cylinders where coolant flows. The inner cylinder is made of a carbon compound to resist high temperature, while the external is of aluminium alloy. The cylindrical moderator bulk, containing the fuel elements, is made of beryllium or lithium hydride. This fuel may reach up to 3500 K at 60 atm. The advantages of PBR are high fuel density and therefore high power density (up to 40 MW/dm<sup>3</sup>). These make PBR suitable for interplanetary missions.

In the preliminary design of the nuclear reactor many requirements and constraints must be considered [7]:

- the reactor must remain critical with a reactivity coefficient  $k_{eff} = 1.000$  (+0.005, -0.000).

- the reactor must remain subcritical in the worst accidental conditions, also when immersed in water or wet sand, and in dry sand.
- nuclear fuel  $U^{235}$  is enriched at 93.15% to reduce fuel mass.
- considering the results obtained from reactors already built, the minimum fuel burn up must be at least 4%.

The assumptions considered in dimensioning the reactor are:

- Core: cylindrical;
- Radius of shield equal to radius of core;
- Thermal efficiency of turbomachinery: 33%;
- Reactor average density:  $1600 \text{ kg/m}^3$ ; it includes coolant pipes, moderators, reflectors and all of the components except the pressure vessel [8].
- Coolant volume: about 16% of total core volume.
- Pressure drop of the propellant gas in duct: 3.5% of total pressure.
- Payload and housing module: at 30m from reactor. This value is a good trade-off between distance to reduce radiation from reactor and mass of boom.
- Shell deformation:  $< 2\%$  at operating pressure and temperature.

Dimensioning the PBR core used results of from the Los Alamos reactor code (1986) [8]. This code evaluates criticality and required power using the Monte Carlo method. To analyze a large range of power the analysis was made with 7, 19, and 37 fuel elements. Knowing power requirements the dimensions are obtained directly. In the particular case in which two solutions are found, the configuration with the lighter mass is chosen. The expressions to calculate core radius are :

$$R_{core} = 9.0958(10)^{-10} P_{core}^4 - 1.3261 (10)^{-6} P_{core}^3 + 7.1665 (10)^{-4} P_{core}^2 - 0.1735 P_{core} + 47.625 \quad (2.1)$$

for 7 fuel elements;

$$R_{core} = -2.655 (10)^{-12} P_{core}^5 + 8.946 (10)^{-9} P_{core}^4 - 1.1703 (10)^{-5} P_{core}^3 + 7.427(10)^{-3} P_{core}^2 - 2.2955 P_{core} + 313.34 \quad (2.2)$$

for 19 fuel elements;

$$R_{core} = 4.905 (10)^{-11} P_{core}^4 - 2.881 (10)^{-7} P_{core}^3 + 6.2522(10)^{-4} P_{core}^2 - 0.5992 P_{core} + 252.28 \quad (2.3)$$

for 37 fuel elements.

Similarly, core height  $H$  is from :

$$H_{core} = 2.83 (10)^{-4} P_{core}^2 + 0.5203 P_{core} + 26.06 \quad (2.4)$$

for 7 elements;

$$H_{core} = -4.027 (10)^{-5} P_{core}^2 + 0.1427 P_{core} + 17.9883 \quad (2.5)$$

for 19 elements;

$$H_{core} = -6.502 (10)^{-6} P_{core}^2 + 0.05009 P_{core} + 18.335 \quad (2.6)$$

for 37 elements;

with  $R_{core}$ = core radius (cm)

$H_{core}$ = core height (cm)

$P_{core}$ = thermal power (MW)

The results obtained for core dimensions are:

$P_t$ (MW)	$P_e$ (MW)	$R_{core}$ 7 el. (cm)	$R_{core}$ 19 el. (cm)	$R_{core}$ 37 el. (cm)	$H_{core}$ 7 el. (cm)	$H_{core}$ 19 el. (cm)	$H_{core}$ 37 el. (cm)
333	100.018	31.5	39.5	112.0	168.1	61.1	34.3
500	150.018	31.1	35.6	76.0	215.5	79.3	41.8
667	200.018	37.2	33.8	55.0	247.2	95.2	48.8
1000	300.018	174.3	32.8	39.3	263.4	120.4	61.9

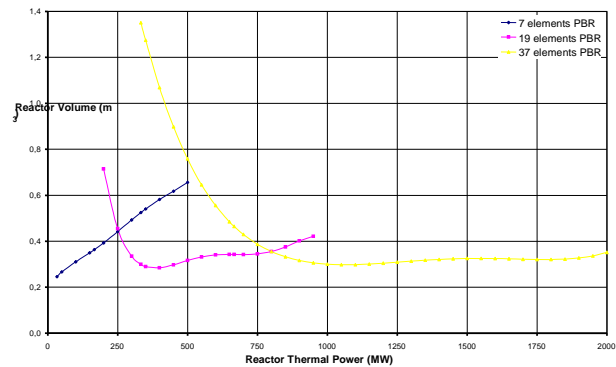
**Table 2.1 Core radius and core height for different fuel configurations**

Once core dimensions are known, its mass can be estimated. With average constant [bulk] density the mass results are 90% reliable [8]. They are:

$P_t$ (MW)	$V_{core}$ 7 el. ( $m^3$ )	$V_{core}$ 19el. ( $m^3$ )	$V_{core}$ 37el. ( $m^3$ )	$m_{core}$ 7 el. (kg)	$m_{core}$ 19 el. (kg)	$m_{core}$ 37 el. (kg)
333	0.525	0.299	1.351	840	478	2161
500	0.656	0.316	0.758	1049	506	1213
667	1.076	0.342	0.464	1722	548	743
1000	25.123	0.408	0.300	40197	653	480

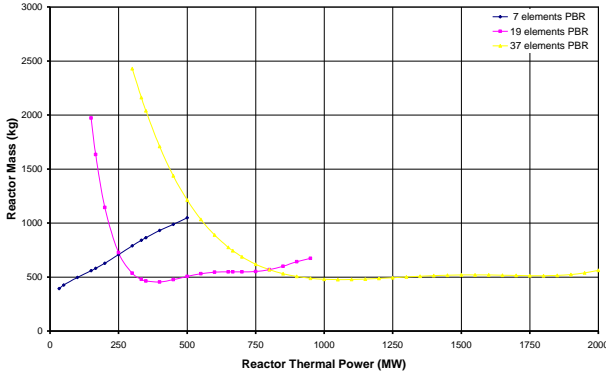
**Table 2.2 Core volume and core mass for different fuel configurations**

The following charts represents volume and mass reactor as function of thermal power.



**Figure 2.1 Volume of reactor as function of thermal power for 7, 19 and 37 fuel elements [8]**





**Figure 2.2 Mass of reactor as function of thermal power for 7, 19 and 37 fuel elements [8]**

The charts above show that for thermal power < 250 MW<sub>th</sub> the lightest configuration is with 7 fuel elements, for power between 250 e 750 MW<sub>th</sub> is 19 fuel elements, while for thermal power above 750 MW<sub>th</sub> the best configuration is with 37 fuel elements. Here the final choice must consider also shield mass and other subcomponents, as illustrated below.

### 2.2 Dimensioning the radiation shield

To dimension the shield different materials and configurations were analyzed using the Monte Carlo Neutronics Program [7]. Beyond the shield, neutrons fluence falls with the inverse of square distance from reactor, while the gamma dose with the inverse of distance to the 2.3 power. Moreover, it was found that at the edge of the shadow projected by the shield the dose and the fluence change strongly with the radial position: the rate of change is approximately exponential with radial distance. From this analysis a simplified model was developed to estimate gamma dose and neutrons fluence beyond the shield. The model uses the best configuration using classic materials, LiH, W and Be: Be is a material that reflect neutrons; W is used for its high attenuation of gamma rays and its high cross section in absorbing neutrons. Lithium hydride completes the attenuations of the neutron flux: in particular, hydrogen slows down them, while lithium has a large cross section. In Table 2.3 below the attenuation coefficients of materials for gamma rays and neutrons are reported:

Material	Neutrons attenuation	Gamma ray attenuation	Density
Be	5.8cm	35cm	1850 kg/m <sup>3</sup>
W	5.2cm	2.0cm	19300 kg/m <sup>3</sup>
LiH	5.6cm	33cm	757 kg/m <sup>3</sup>

**Table 2.3 Attenuation lengths for neutron fluence and gamma dose [7]**

The simplified expressions to calculate neutrons fluence and gamma dose are [7]:

$$F_n = A_n \exp\left(\frac{-L_{Be}}{\lambda_{nBe}} + \frac{-L_{LiH}}{\lambda_{nLiH}} + \frac{-L_W}{\lambda_{nW}}\right) \frac{P_r t_i}{L_p^2} \quad (2.7)$$

$$D_\gamma = A_\gamma \exp\left(\frac{-L_{Be}}{\lambda_{\gamma Be}} + \frac{-L_{LiH}}{\lambda_{\gamma LiH}} + \frac{-L_W}{\lambda_{\gamma W}}\right) \frac{P_r t_i}{L_p^2} \quad (2.8)$$

with  $A_n$ = fluence coefficient (8.1\*10<sup>8</sup>n/J)  
 $A_\gamma$ = dose coefficient (0.30 rad cm<sup>2</sup>/J)  
 $P_r$ = reactor power (W)  
 $t_i$ = reactor operation time (s)  
 $L_p$ = distance from reactor to payload (cm)

Setting the values of beryllium length, fluence and dose, the above expressions can be inverted to obtain tungsten and lithium hydride length:

$$L_{LiH} = \lambda_{nLiH} \lambda_{\gamma LiH} \frac{\left(-\lambda_{nW} \ln\left(\frac{F_n}{C_n}\right) + \lambda_{\gamma W} \ln\left(\frac{D_\gamma}{C_\gamma}\right)\right)}{\lambda_{nW} \lambda_{\gamma LiH} - \lambda_{\gamma W} \lambda_{nLiH}} \quad (2.9)$$

$$L_W = \lambda_{nW} \lambda_{\gamma W} \frac{\left(\lambda_{nLiH} \ln\left(\frac{F_n}{C_n}\right) - \lambda_{\gamma LiH} \ln\left(\frac{D_\gamma}{C_\gamma}\right)\right)}{\lambda_{nW} \lambda_{\gamma LiH} - \lambda_{\gamma W} \lambda_{nLiH}} \quad (2.10)$$

$$\text{with } C_n = A_n \exp\left(\frac{-L_{Be}}{\lambda_{nBe}}\right) \frac{P_r t_i}{L_p^2} \quad (2.11)$$

$$C_\gamma = A_\gamma \exp\left(\frac{-L_{Be}}{\lambda_{\gamma Be}}\right) \frac{P_r t_i}{L_p^2} \quad (2.12)$$

The maximum values of fluence and dose were assumed, respectively, 4.455 x 10<sup>9</sup> n/cm<sup>2</sup> and 15 rad. This choice is based on the threshold values that can be absorbed by humans [9]. Biological effects due to absorption of radiation do not depend only from the dose, but also from the type of radiation and from its density. In fact gamma rays, X rays and beta particle are less dangerous than the heavier alpha particles and fission fragments. Radiations interact with human cells, causing death in acute cases, and DNA damage for long exposure. The results of shield dimensioning for a 333 MW<sub>th</sub> reactor and for a 500 MW<sub>th</sub> reactor, obtained after analyzing different PBR configurations are shown in table 2.4:

	333 MW <sub>th</sub> reactor		500 MW <sub>th</sub>	
	7 elements	19 elements	7 elements	19 elements
<b>R<sub>shield</sub>(m)</b>	0.315	0.395	0.311	0.356
<b>Shield Area (m<sup>2</sup>)</b>	0.312	0.489	0.304	0.399
<b>L<sub>Be</sub> (cm)</b>	18	18	18	18
<b>L<sub>LiH</sub> (cm)</b>	58.71	58.71	59.33	59.33
<b>L<sub>W</sub> (cm)</b>	29.53	29.53	29.83	29.83
<b>m<sub>shield</sub> (kg)</b>	2023	3169	1990	2607

<b>m<sub>core</sub> (kg)</b>	840	478	1049	506
<b>M<sub>core+shield</sub> (kg)</b>	2863	3647	3039	3113

**Table 2.4 Shield geometry and mass for 333 MWth and 500 MWth reactors**

The results of shield dimensioning for a 667MWt reactor and for a 1000MWt reactor are:

	<b>667 MWth reactor</b>		<b>1000 MWth reactor</b>	
	19 elements	37 elements	19 elements	37 elements
<b>R<sub>shield</sub>(m)</b>	0.338	0.550	0.328	0.393
<b>Shield Area (m<sup>2</sup>)</b>	0.359	0.951	0.339	0.484
<b>L<sub>Be</sub> (cm)</b>	18	18	18	18
<b>L<sub>LiH</sub> (cm)</b>	60.08	60.08	61.57	61.57
<b>L<sub>w</sub> (cm)</b>	30.19	30.19	30.91	30.91
<b>m<sub>shield</sub> (kg)</b>	2377	6288	2292	3274
<b>m<sub>core</sub> (kg)</b>	548	743	653	480
<b>M<sub>core+shield</sub> (kg)</b>	2924	7031	2945	3754

**Table 2.5 Shield geometry and mass for 667 MWth and 1000 MWth reactors**

### 2.3 Dimensioning of pressure vessel and electric power generation system

The reactor is inside a pressure vessel, and its mass was also calculated based on standard empirical engineering formulae [8]:

$$m_{pv} = \frac{p_i V_{tot}}{g_0 \phi_v} \quad (2.13)$$

with  $p_i$ = vessel peak pressure (Pa)  
 $V_{tot}$ =vessel total volume (m<sup>3</sup>)  
 $\Phi_v$ = vessel mass factor(m)

The internal pressure is assumed 6 MPa, and peak pressure twice as much. The mass factor depends on material: for completely metal vessel is 2500 m. The vessel is assumed cylindrical with two hemispherical end caps.

Heat from core is absorbed by a coolant gas, here a mixture of 10% helium and 90% xenon, with a total mass flow rate = 8.9 kg/s. This hot flow produces mechanical work via turbomachinery driving an alternator to produce electric power. The power conversion system is a simple Brayton cycle with 33% efficiency. The mass of the turbomachinery is also calculated with empirical formulae [7]:

$$m_m = \sqrt{(200 * (power\ level\ in\ kWe) / 100)} \quad (2.14)$$

At these power levels the alternator is slightly heavier than turbomachinery: its mass is conservatively estimated

20% heavier. Based on experimental data, the mass of coolant ducts is assumed to be 15% that of turbomachinery. The mass of space radiators was estimated considering the results in [10] that employ C/C radiators. Their areal mass is about 1.5kg/m<sup>2</sup>. We have considered a radiator inlet temperature = 900K and a C/C emissivity = 0.8. The results are in Table 2.6:

Thermal power reactor (MW)	Radiator Mass (ton)
333	11.7
500	17.5
667	23.4
1000	35.1

**Table 2.6 Radiators Mass for Different Reactor P<sub>th</sub>**

Finally we have assumed that the support structure is 10% of total inert mass (core shield and pressure vessel).

In summary, the results obtained are in the Table below:

	<b>333 MWth reactor</b>		<b>500 MWth reactor</b>	
	7 elements	19 elements	7 elements	19 elements
core + shield mass (kg)	2863	3647	3039	3113
pressure vessel mass (kg)	321.2	272.2	382.6	247.3
Turbo machinery mass (kg)	447.2		547.7	
Alternator mass (kg)	536.7		657.3	
Ducts mass (kg)	67.1		82.2	
Support structure mass (kg)	325.2	398.7	350.4	344.2
<b>Total Mass (kg)</b>	<b>4560.6</b>	<b>5369.0</b>	<b>5059.6</b>	<b>4991.1</b>

**Table 2.7 Reactor and power generation masses for 333 MWth and 500 MWth reactors**

	<b>667 MWth reactor</b>		<b>1000 MWth reactor</b>	
	19 elements	37 elements	19 elements	37 elements
core + shield mass (kg)	2924	7031	2945	3750
pressure vessel mass (kg)	246.7	568.4	272.2	270.6
Turbo machinery mass (kg)	632.5		774.6	
Alternator mass (kg)	758.9		929.5	

Ducts mass (kg)	94.9		116.2	
Support structure mass (kg)	326.6	769.4	333.3	414.0
<b>Total Mass (kg)</b>	<b>4984.0</b>	<b>9855.0</b>	<b>5370.6</b>	<b>6258.4</b>

**Table 2.8 Reactor and power generation masses for 667 and 1000 MWth reactors**

At this stage the objective is to choose the reactor configuration minimizing mass. A core with 7 fuel elements is chosen for the 333MWth reactor, and with 19 fuel elements for the 500MWth, 667 MWth and 1000 MWth reactors.

### 3. ORBIT TRANSFER

#### 3.1 Key requirements

The main problem for this M3 is the interplanetary flight time, influenced by the GCR (health risks) and driving fuel mass and cost. The low thrust of NEP is another problem, because present-day all orbital changes are based on impulsive manoeuvres, standard in CP, but unfeasible with the low thrust of EP. In fact, low and constant thrust causes different types of losses:

- 1) Gravitational losses, due to the angle between the local horizon of the spacecraft and the tangent to the trajectory (called "flight angle") that cannot be kept zero during the entire manoeuvre
- 2) Steering losses, due to the attitude control system
- 3) Misalignment losses, due to misalignment between velocity and thrust vectors

A MATLAB™ program to calculate and chart the Earth-Mars legs and back, was used with the following input:

- spacecraft mass, specific impulse and thrust
- altitude and parameters of starting and final orbits
- day of departure from LEO.
- days of acceleration
- days of breaking

Starting from this input the program reads a subroutine with Earth, Moon and Mars positions at the starting location and time, and then integrates the three velocity components of all bodies under the effect of Newton's Law and Gravitation among all bodies (a n-body problem). Earth and Mars orbits are assumed unperturbed and elliptic, inclined with respect to the Sun equator. The program includes interaction among all bodies. For simplicity, thrust and velocity vectors are assumed always aligned

The spacecraft starts from a 400 km LEO and arrives in a low Mars orbit (200 km LMO). The program calculates total  $\Delta V$  and fuel mass consumed by dividing the trans-Mars leg into two segments: in the first the spacecraft accelerates to the maximum speed possible, in the second the spacecraft slows down to capture the LMO. Thrust

for the four thruster electric powers assumed is given below:

- 1) Power = 100 MWe => thrust = 311 N
- 2) Power = 150 MWe => thrust = 467 N
- 3) Power = 200 MWe => thrust = 622 N
- 4) Power = 300 MWe => thrust = 933 N

All the configurations assume same Isp (32,780 s) and we assumed the starting mass of 150 ton, but after many iteration we had the following results:

Power	100	MWe
<i>Outward journey:</i>		
Initial mass	120.5	ton
Days	180	
Fuel consumption	15	ton
Mass consumption	3.3	ton
$\Delta V$	42.7	km/s
Arrival mass	102.2	ton
<i>Stay on Mars:</i>		
Days	365	
Mass consumption	6.8	ton
<i>Return journey:</i>		
Initial mass	102.2	ton
Days	155	
Fuel consumption	12.9	ton
Mass consumption	2.9	ton
$\Delta V$	43.45	km/s
Arrival mass	86.4	ton
Total days	700	
Total fuel consumption	30.7	ton
Total mass consumption	6.2	ton
Structures and propulsion system mass	76.94	ton
Exceeded mass	6.7	ton

**Table 3. 1 Leg duration, mass and fuel consumption**

Power	150	MWe
<i>Outward journey:</i>		
Initial mass	130	ton
Days	157	
Fuel consumption	19.6	ton
Mass consumption	2.9	ton
$\Delta V$	52.6	km/s
Arrival mass	107.5	ton
<i>Stay on Mars:</i>		
Days	333	
Mass consumption	6.2	ton
<i>Return journey:</i>		
Initial mass	107.5	ton
Days	125	
Fuel consumption	15.6	ton
Mass consumption	2.3	ton
$\Delta V$	50.53	km/s
Arrival mass	89.5	ton

Total days	615
Total fuel consumption	38.8 ton
Total mass consumption	5.2 ton
Structures and propulsive system mass	83.53 ton
Exceeded mass	2.5 ton

**Table 3. 2 Leg duration, mass and fuel consumption**

Power	200	MWe
<i>Outward journey:</i>		
Initial mass	140	ton
Days	125	
Fuel consumption	20.8	ton
Mass consumption	2.3	ton
$\Delta V$	51.8	km/s
Arrival mass	116.8	ton
<i>Stay on Mars:</i>		
Days	285	
Mass consumption	5.3	ton
<i>Return journey:</i>		
Initial mass	116.8	ton
Days	110	
Fuel consumption	18.4	ton
Mass consumption	2.0	ton
$\Delta V$	54.9	km/s
Arrival mass	96.4	ton
Total days	520	
Total fuel consumption	43.1	ton
Total mass consumption	4.4	ton
Structures and propulsive system mass	89.8	ton
Exceeded mass	2.7	ton

**Table 3. 3 Leg duration, mass and fuel consumption**

Power	300	Mwe
<i>Outward journey:</i>		
Initial mass	160	Ton
Days	100	
Fuel consumption	25	Ton
Mass consumption	1.9	Ton
$\Delta V$	54.7	km/s
Arrival mass	133.01	Ton
<i>Stay on Mars:</i>		
Days	205	
Mass consumption	3.8	Ton
<i>Return journey:</i>		
Initial mass	133.01	Ton
Days	80	
Fuel consumption	20.0	Ton
Mass consumption	1.5	Ton
$\Delta V$	60.1	km/s

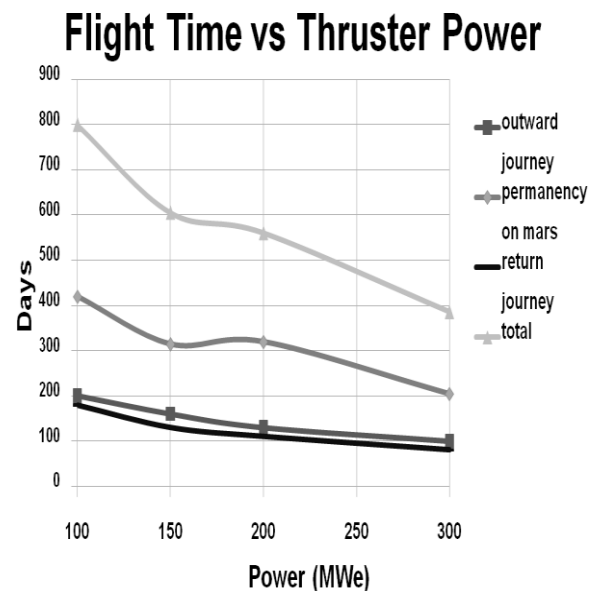
Arrival mass	111.6	Ton
Total days	385	
Total fuel consumption	49.5	Ton
Total mass consumption	3.3	Ton
Structures and propulsive system mass	102.6	Ton
Exceeded mass	4.5	Ton

**Table 3. 4 Leg duration, mass and fuel consumption**

The *mass consumption* includes all consumables for a Mars-direct trajectory, a crew of four and includes also 10% safety margins.

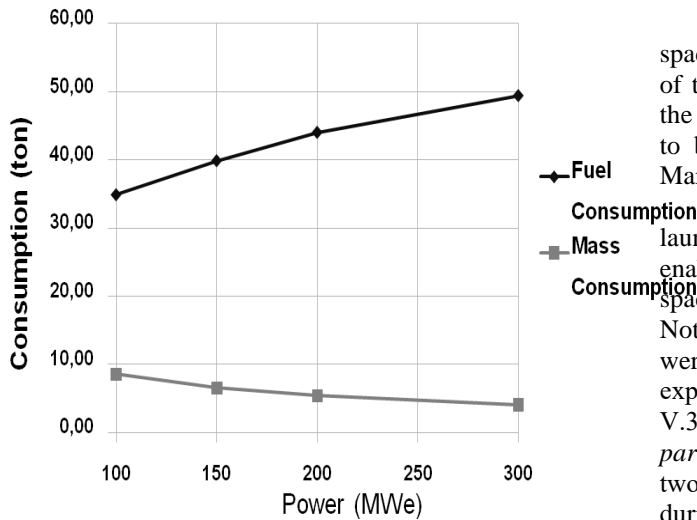
The *total mass consumption* does not include the mass consumed during the crew stay on Mars, assumed sent in advance via a cargo mission.

The *total fuel* consumption includes a 10% safety margin.



**Figure 3.1: Flight time as function of power**

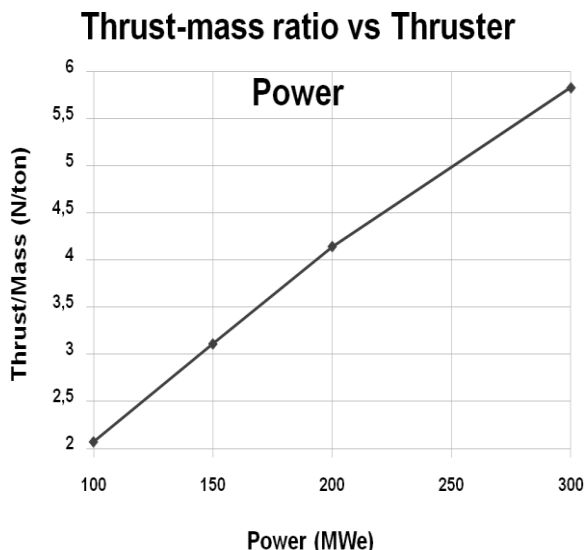
## Fuel and Mass Consumption



**Figure 3.2: Fuel and mass consumption as function of power**

Note that raising the reactor power (thus thrust and  $\Delta V$ ), the fuel consumption increases only moderately. Fuel is only 32.2% of the spacecraft mass with power = 100 MWe, decreasing to 29.85%, 30.78% and 30.94% for reactor's power of 150, 200 300 MWe. Flight time, at the largest power, decreases more, by 46,27%, and the entire mission time by 45%.

Note that there is an exceeded mass in the final mass of the spacecraft, clearly, iterating should be kept until convergence is actually reached. Alternatively, excess mass should be assumed to increase safety margins on fuel and consumables.



**Figure 3.3: Thrust-Mass ratio as function of power**

## 4. MASS BUDGET

The mass budget consists in the total mass of the spacecraft with all its systems, subsystems and modules of the mission. This calculation is fundamental, because the cost of a mission depends strongly on the total mass to be launched to orbit and then to be accelerated to Mars.

Currently, the whole spacecraft cannot be launched in a single launch. (Presumably, that could be enabled by a future version of ARES 5.) In any case, the spacecraft would probably have to be assembled in LEO. Note also that with CP propulsion two cargo missions were estimated to be necessary in advance of the human expedition in the NASA's reference mission (V.2 and V.3) of the '90s. Here the goal was to design the *human part* of the mission. Accordingly, its mass is divided into two parts: a fixed mass that doesn't change significantly during the trip, and the propellant mass.

In the following sections these masses are reported after having been calculated in detail; for brevity these details are not shown, but will be published later on in a dedicated paper.

### 4.1 Fixed mass

This section lists the components constituting the fixed mass, including all electrical and structural systems and subsystems, the propulsion system and the Martian modules. These mass estimates are based on assuming 100 MW, 150 MW, 200 MW and 300 MW of electric power available, influencing reactor and propulsion system mass.

#### 4.1.1 Structure, modules and avionics

##### Structure

The spacecraft structure supports all other subsystems and ensures the necessary strength during its life. It provides protection to the astronauts from GRC and from meteorites.

##### Modules [11]

- **MAS:** it is the module where the crew will live during the journey to Mars and the stay on the planet. After capturing LMO (Low Mars Orbit), will reach Mars' surface with a parachute and the propulsion system. The mass budget includes therefore also the thermal shield needed during the entry in the Martian atmosphere. The following table shows all the elements composing the module.

Component	Mass (ton)
Structure of MAS	8.5
Mechanisms	4.05
Landing propellant	9
Propeller and Tank	2.2
Module Parachute	0.6
<b>TOTAL</b>	<b>24.35</b>

**Table 4.1**

- **CAP:** it is the capsule for the re-entry on Earth.
- **MTT:** it's the module for the Mars-Earth transfer. It is formed from CAP and MAS.

#### Avionics

Includes all the electrical systems and subsystems necessary to pilot the spacecraft; it has a mass of 600 kg.

#### 4.1.2 ECLSS, EVA subsystem, Crew

##### ECLSS [11]

The Environmental Control and Life Support System is the system that reproduces Earth-like atmosphere in the spacecraft. It includes water, nitrogen, oxygen and all the subsystems necessary to create a livable spacecraft environment. In particular it generates the internal pressure and controls the local temperature and the air cycle.

##### Water and food

	Daily need per person (kg)	Percent Recycled %	100MW <sub>e</sub> reactor*	150MW <sub>e</sub> reactor*	200MW <sub>e</sub> reactor*	300MW <sub>e</sub> reactor*
Journey time (days)			335	282	235	180
Oxygen (kg)	1	80	294.8	248.2	206.8	158.4
Dry food (kg)	0.5	0	737.0	620.4	517.0	396
Whole food (kg)	1	0	1474.0	1240.8	1034.0	792
Drinking water (kg)	4	80	179.2	992.6	827.2	633.6
Washing water (kg)	26	90	3832.4	3226.1	2688.4	2059.2
Total (kg)	32,5	87	6227.6	5242.4	4368.6	3346.2

**Table 4.2 (\* Mass includes 10% safety margins)**

#### 4.1.3 Propulsion system

The propulsion system includes the nuclear reactor and its shielding, the turbo machinery and the generator to power the electrical thrusters. These electrical are assumed 30 m away from the reactor and its shield.

Water is utilized in many systems and subsystems and to wash and drink. The calculations also include food for the astronauts, see Table 4.2 [12]. The crew consumable mass per day is 32.5 kg.

##### EVA subsystem [11]

The Extravehicular Activity (EVA) subsystem spacesuit operates when the astronauts leave the spacecraft. The spacesuit keeps pressure constant; controls temperature and rejects heat; provides the air-cycle; allows drinking and urinating; monitors vital parameters; and grants telecommunication(s) with the modules.

##### Crew

The mission foresees a four-person crew. Each has been assumed to weigh 110 kg. Training equipment is included in the mass budget. This mass is reported in table 4.3.

#### 4.2 Total mass

<b>MAS (ton)</b>	24.35
<b>CAP (ton)</b>	7
<b>Structure (ton)</b>	14.2

<b>Avionics (ton)</b>	0.6
<b>Crew and Crew support (ton)</b>	9
<b>ECLSS (fixed) (ton)</b>	4.8
<b>Total (ton)</b>	59.95

**Table 4.3**

The mass of the propulsion system, radiator and propellant changes with power (100 MW, 150 MW, 200 MW, 300 MW). In the first case there are twenty 5-MW ion thrusters, thirty in the second, forty in the third and sixty in the last case. Each engine weighs ~50 kg. The weights of reactor, radiators and propellant are shown in the following table with the other results.

The following table lists all the main masses.

<b>Power (MWe)</b>	100	150	200	300
<b>Fixed mass (ton)</b>	59.95	59.95	59.95	59.95
<b>ECLSS total (ton)</b>	6.23	5.24	4.37	3.35
<b>Reactor (ton)</b>	4.56	4.99	4.98	5.37
<b>Electric Thrusters (ton)</b>	0.73	1.09	1.45	2.18
<b>Radiators (ton)</b>	11.70	17.50	23.40	35.10
<b>Propellant (ton)</b>	30.66	38.77	43.13	49.55
<b>Total (ton)</b>	113.82	127.54	137.28	155.49
<b>Initial mass (ton)</b>	120.5	130.0	140.0	160.0
<b>Excess mass (ton)</b>	6.68	2.46	2.71	4.51

**Table 4.4**

## CONCLUSIONS

The combination of Isp and power of the GIE NEP system for the M3 considered here results in predictions for both mass and mission times that are significantly better than with other CP and NTR propulsion systems. Especially with respect to CP, the mass saving is noteworthy, as well as the fact that using NEP increases power, but does not increase mass proportionately, while mission time is substantially reduced. These results were obtained with detailed calculations for the mass budget, after reactor and thruster sizing based on current technology and know-how. The Isp assumed for the GIE system is, in fact, rather conservative, as there is no conceptual or engineering obstacle to raise it by a factor 2 or more in the near term [6].

This said, there are many engineering issues connected with NEP for a M3 that were not discussed here because of time and space limitations; among them reactor integrity (the effect of radiation on materials), thruster life, how to refuel a compact reactor while it is operating or 'hot', and many others. They will be examined in the future.

Based on the results presented, a NEP-powered M3 appears not only feasible, but also more convenient than CP- and likely also NTR-powered missions in terms of cost, besides being the only way to drastically reduce travel time and thus GCR dose for the crew. This last is gradually being recognized the bottleneck of interplanetary human missions.

To enable a future NEP M3, investing in this propulsion technology is necessary. That is an unlikely perspective in the current financial climate, but would spare much time and effort to our future generations.

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