PERGAMON

Available online at www.sciencedirect.com
SCIENCE DIRECT®

Acta Astronautica 54 (2004) 471 – 486

ACTA ASTRONAUTICA

www.elsevier.com/locate/actaastro

Mission to Mars using integrated propulsion concepts: considerations, opportunities, and strategies

Antonio G. Accettura[a,*,1], Claudio Bruno[b,2], Stefano Casotto[c,3], Francesco Marzari[c,4]

[a] *Space Propulsion Design Department, FiatAvio, C.so Garibaldi 22, 00034 Colleferro Rome, Italy*
[b] *University of Rome "La Sapienza", Via Eudossiana 18, 00184 Rome, Italy*
[c] *CISAS, University of Padua, Via Marzolo 8, 35131 Padua, Italy*

Received 15 April 2002; received in revised form 19 December 2002; accepted 2 June 2003

Abstract

The aim of this paper is to evaluate the feasibility of a mission to Mars using the Integrated Propulsion Systems (IPS) which means to couple Nuclear-MPD-ISPU propulsion systems. In particular both mission analysis and propulsion aspects are analyzed together with technological aspects.

Identifying possible mission scenarios will lead to the study of possible strategies for Mars Exploration and also of methods for reducing cost.

As regard to IPS, the coupling between Nuclear Propulsion (Rubbia's engine) and Superconductive MPD propulsion is considered for the Earth–Mars trajectories: major emphasis is given to the advantages of such a system. The In Situ Resource Utilization (ISRU) concerns on-Mars operations; In Situ Propellant Utilization (ISPU) is foreseen particularly for LOX-CH4 engines for Mars Ascent Vehicles and this possibility is analyzed from a technological point of view. Tether Systems are also considered during interplanetary trajectories and as space elevators on Mars orbit.

Finally strategic considerations associated to this mission are considered also.

1. Introduction

The scope of this paper is to evaluate a Manned Mars Mission by using Integrated Propulsion Systems (IPS): Nuclear-Electric-ISPU, including Tethered Systems at the mission architecture level. Two aspects shall be analyzed here: the mission and its propulsion system. In the first the following aspects shall be studied: mission scenarios, possibilities of, and strategies for, Mars Exploration/Colonization, and methods for reducing costs of Mars Exploration. In the second, different advanced propulsion systems shall be investigated at both the system and the technological levels. The following main tasks have been singled for concept validation:

- A Tethered System as an artificial gravity device during interplanetary flight.
- Coupling between Nuclear Propulsion using Rubbia's concept engine [1] and Superconductive-MPD Systems [2].

* Corresponding author. Tel.: +39-06-97285851; fax: +39-06-97285316.
E-mail address: antonio.accettura@fiatavio.it (A.G. Accettura).
[1] Technical Manager, Advanced Propulsion, AIAA member.
[2] Associate Professor, AIAA Fellow Member.
[3] Assistant Professor, AIAA Member.
[4] Researcher.

0094-5765/$ - see front matter
doi:10.1016/S0094-5765(03)00208-X

Nomenclature

Glossary of Acronyms

AF	Applied field
Ar	Argon
AU	Astronomical unit
B	Magnetic field
CH_3OH	Methanol
CH_4	Methane
CO_2	Carbon Dioxide
d	days
EP	Electric Propulsion
g	gravity acceleration at Earth surface
g/s	mass flowrate grams per seconds
GW	giga-Watts
H_2	hydrogen
HC	Hydrocarbon
HLLV	Heavy Lift Launch Vehicle
HT	Hall Thrusters
HTSC	High Temperature Super-Conducting
IPS	Integrated Propulsion Systems
ISMU	Indigenous Space Materials Utilization
Isp	Specific Impulse
ISPP	In situ propellant production
ISPU	In situ propellant utilization
ISRU	In situ resource utilization
ISS	International Space Station
JPL	(NASA) Jet Propulsion Laboratory
kg	kilograms
km	kilometer
kN	kilo-Newton
kW	kilowatt
LEO	Low Earth Orbit
LH_2	Liquid Hydrogen
LOX	Liquid Oxygen (O_2)
LTSC	Low Temperature Super-Conducting
m	meters
M2IPS	Mission to Mars using ISP
M3	Manned Mission to Mars
MAV	Mars Ascent Vehicle
MHD	Magneto-Hydro-Dynamic
mN	milli-Newton
MOE	Mars Orbit Elevator
MPD	Magneto-Plasma-Dynamic
MSO	Mars Stationary Orbit
MW	Mega-Watts
N	thrust in Newton
NASA	National Aeronautics and Space Administration
Nb	Niobium
NR	Nuclear Rockets
O/F	Oxidizer-to-Fuel ratio
s	seconds
S/WE	Sabatier/Water Electrolysis
SPE	Solid Polymer Electrolyte
T	Tesla
t	tons
Ti	Titanium
TiPS	Tether physics and survivability experiment
TSS	Tethered Satellite System
W	Watts
ΔV	delta velocity
μT	micro-Tesla

- In situ methods for LOX-CH_4 engine on Mars.
- A Tethered System as space elevator in Mars Stationary Orbit (MSO).
- Technologies and operating modes associated to different propulsion phases.

The main target of this paper is to propose a Manned Mission to Mars (M3) by using a new point of view: the IPS, extracting maximum benefit from each propulsion device considered.

2. Mission scenario

Imagine a twin-spacecraft concept (we can call it M2IPS = Mission to Mars using ISP), assembled in Low Earth Orbit, close to ISS, 200 tons each (assuming a total payload of about 400 tons, 100 for modules and 300 for the spacecraft and the propulsion system [3]). In Table 5 there is a mass breakdown for the spacecraft. The Rubbia's engine is turned on and a superconductive MPD is activated using part of the energy released by Rubbia's engine: M2IPS begin to leave its parking orbit. Both Nuclear and MPD propulsions are actives and are used to reach the Earth escape trajectory. Once the Rubbia's engine has finished its propellant, the twin-spacecraft separates into two by means of a tether, then they begin to move around their center of mass to produce artificial gravity up to 0.4 g, the same of Mars surface. The spacecraft is now ready to go toward Mars using an interplanetary trajectory, using both Nuclear and Electric propulsion by means of a continuous thrust applied during the Trans-Mars trajectory.

Once well inside Mars gravity field, the tether is retrieved and the twin-spacecraft rotates 180° opposite to the flight direction in order to decelerate. A self-capturing maneuver is performed acting on the Rubbia's engine thrust vector (an optimization of the whole interplanetary trajectory is foreseen). Then M2IPS is inserted in a Mars Stationary Orbit (MSO) 17, 000 km from Mars' surface, and the tether is deployed down to an altitude of 700 km: this tether is used as a space elevator in order to move the spacecraft from 17,000 to 700 km and vice-versa by means of a mechanical device in micro-gravity environment. The Mars descent and soft landing is performed by using an Aerobrake-Retrothruster Mars Descent Vehicle.

While or after completing the surface Exploration Program, an ISPU phase is activated in order to produce propellants for the Mars Ascent Vehicle and the return phase. The spacecraft is rocket-lifted to 700 km and then, by means of the space elevator, to the MSO at 17, 000 km, where Rubbia's engine is again started to insert the spacecraft in the Trans-Earth return trajectory.

The rendezvous with ISS ends the first phase of Mars Colonization: a number of similar missions could result in a tether constellation around Mars and enable utilization of its resources.

3. Mars exploration: possibilities and strategies

It is well known that of all bodies in the solar system other than Earth, Mars is unique in that it has the resources required to support a population of sufficient size and create locally a new branch of human civilization. It is also known that while Mars may lack any material directly exportable to Earth for cash, Mars' orbital elements and other physical parameters make for unique advantages, allowing it to act as a keystone in supporting extractive activities in the asteroid belt and probably also elsewhere in the solar system.

To do so it is mandatory to start with a Mars exploration program conducted by a human crew; robotic missions are already planned in order to improve our knowledge on Mars.

3.1. *Methods for reducing cost in Mars exploration*

We can use materials that already exist on Mars in an innovative approach to reduce the need (and expense) to bring everything from Earth. There is every indication that exploring the Solar System and expanding humanity's permanent presence on the Moon and Mars will provide significant benefits to Earth. We can process natural resources on Mars into products we need at an outpost, thus avoiding the need to bring them from Earth. Among these resources are oxygen and nitrogen (important elements for atmosphere air-based) to breathe and water to drink. As seen in Section 4.4, propellants can be manufactured. Carbon dioxide (CO_2) can be extracted from the Mars atmosphere to grow foodstuff. Bricks and panels from in situ rocks may be manufactured to build habitats. Given sufficient energy, provided by a Rubbia fission generator, metals from local rocks and soil can make beams, wires, and perhaps even solar power cells. In short, much needed for life on the new frontier can be produced from local resources [4].

Indigenous Space Materials Utilization (ISMU, also called In Situ Resource Utilization) can provide a reduction in cost and can increase our capabilities significantly as we develop and expand a Mars outpost. The goal for any ISMU program is to free these outposts from total reliance on the Earth as soon as possible, thereby rapidly shifting the nature of our space transported cargo away from bulk materials, such as propellants and building materials, to additional people and complex equipment.

3.2. *Mars resource utilization*

The Martian atmosphere, consisting mostly of carbon dioxide, can be processed to release oxygen for life support or propellant use. Carbon monoxide, which could be a moderate performance rocket fuel, is the co-product. By combining its oxygen with a small amount of hydrogen, water for a variety of uses could be produced for only a fraction of its mass if brought from Earth. One good aspect of Mars atmosphere utilization is that no mining is involved. Simple gas handling equipment can be used, providing a much more reliable system (see Section 4.3).

Life support technologies routinely deal with conversion of CO_2 to other compounds, including methane. This process (Sabatier + Electrolysis Cycles) was discovered nearly one hundred years ago and is still used in many chemical plants today. Direct application of this technology to the Martian atmosphere would allow for the production of oxygen, methane and water by ferrying to Mars only a small amount of hydrogen. Thus, large quantities of propellant could be leveraged from minimal mass import. A rocket engine using methane and oxygen could be developed for use in Martian spacecraft. Chemical procedures exist to convert carbon dioxide, which is 95% of the atmosphere, into products such as oxygen, water, and methane.

4. Integrated propulsion system scenario

For each identified propulsion system some parameters, crucial to both new and current technologies are

- Expected performance (Isp, Thrust)
- Expected mass and volume
- Expected power
- Critical technologies
- Spacecraft critical interfaces

Advanced propulsion techniques are described below in order to introduce the IPS.

4.1. Nuclear propulsion and Rubbia's engine

Conventional nuclear rockets (NR) based on passing and heating (up to about 2000 K) GH_2 in a compact nuclear reactor were developed over a period of about 20 years in the US and the former Soviet Union. In the US this program produced the [static] reactors KIWI and then the NERVA engine, before the program was stopped by a combination of political, technical and budget factors. A substantial advance in NR is very likely to occur if the so-called "Rubbia Nuclear Rocket" concept will be developed. In the Rubbia's concept the heat exchange is 'reversed', with fission fragments from the [always] subcritical fission of an isotope of Americium heating the working fluid. This means that, in counterposition to a NERVA-type concept, the solid (Americium) will always operate at temperatures much lower than those in the gas: thus the limitation posed by the melting temperature of fissionable materials may, in principle, be bypassed, leading to a substantial increase of Isp (i.e., up to 2000–4000 s). This Isp can reduce mission time to Mars and back to just one year. Critical technologies are: cooling (possibly) MHD plasma control, and Am metallurgy.

Either as a by-product of a Rubbia's NR, or using the Rubbia Americium fission process, there is the possibility of powering an ion electric thruster without the associated penalty of large solar cells array (and structural problems) typical of conventional ion thrusters. In the first case, about 50% of the heat produced by the Rubbia fission process can power an ion thruster (which could reach Isp of order 6000–10,000 s). In the second case, Americium fission would be wholly devoted to power the ion engine. A variation on this theme is a nuclear powered MHD thruster relying on the so-called HTSC (super conducting) coils to generate the Lorentz' force. Carrying LH_2 propellant enables super conducting technology in space for long duration missions.

Critical technologies are: cooling, power conversion, Am metallurgy, and shielding instrumentation from the onboard em field.

A schematic of Rubbia's engine is shown in Fig. 1 [3].

Due to the intrinsic limitation of the Rubbia's engine architecture, such an engine is unable to use all the energy produced by fragment fission heating mechanism. Starting from an energy budget of 30 MW [3], about 15 MW are used to heat liquid hydrogen for propulsion purposes, while 15 MW are transferred to the engine walls as thermal energy. Such energy can be exploited; the following is a non-exhaustive list of applications that should further investigated in order to efficiently use all the power produced by the

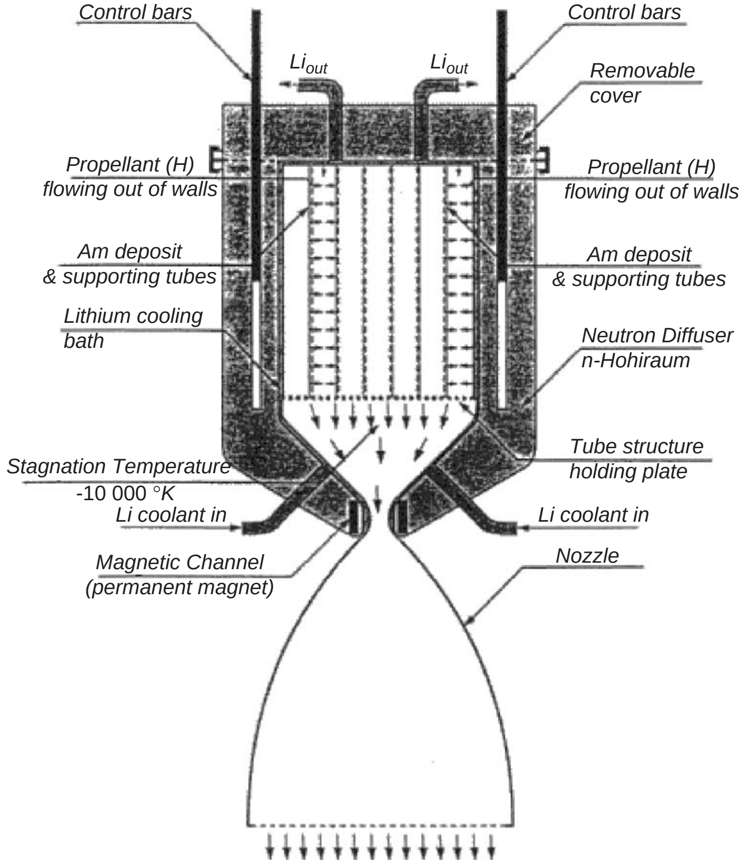

Fig. 1. Rubbia's engine.

Rubbia's engine:

- *Superconductive-MPD*: In this case energy conversion from thermal to electrical is used for MPD propulsion using superconductive coils for long-range, high performance capability.
- *Data transmission*: The goal is to use the excess thermal power to power a communication laser to transmit wide-band information to Earth. A possible scenario includes an heliocentric satellite (close the Earth) to be used as data relay system.
- *Energy accumulator*: This is one of the most pressing needs in long-duration missions, nevertheless there are a lot of technological problems related to how convert energy and how to store it for months/years.
- *Electrolysis*: Excess power could be used in an electrolysis plant to produce H_2 from stored water. Since water is denser than LH_2 by a factor 15, the spacecraft could be more compact, albeit heavier. O_2 could be used as oxidizer for hydrogen or CO/methane produced on Mars' surface, for thrust bursts, and of course for the crew.
- *On-board power*: Of course it is possible to convert thermal power into electrical power (e.g., in a Stirling cycle) in order to use it for all on-board systems, including those described above.
- Finally, the Rubbia engine should be seen as a power generator in general. It could be used for Earth orbit activities also (such as ISS, Moon missions, on-orbit Spaceport, and so on).

Another option being explored for the Rubbia's engines to reduce thermal losses by using a two-side cylindrical Americium emitter instead of a single inner surface (Fig. 2), hydrogen could flow through both

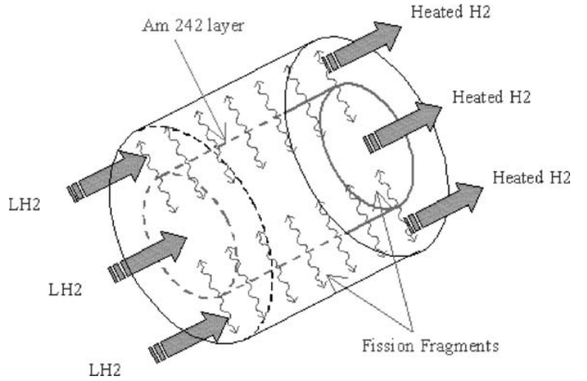

Fig. 2. Bi-cylindrical chamber Americium emitter configuration.

cylindrical surfaces (outer and inner) with increased efficiency.

All items described above should be further investigated in order to analyze their conceptual and technological feasibility.

4.2. Tether systems

Conventional launchers powered by chemical rockets have shown their limits (low specific impulse). Low Isp is a much worse bottleneck thinking of Moon and Mars colonization missions. Since future challenges will be to reach remote destinations in the solar system in the shortest time, possibly carrying more massive and multi-role payloads, it is clear that chemical propulsion alone is not viable.

Tethers have been already employed in about twenty space missions; although many consider the TSS-1 and TSS-1R missions as failures, we must remind that the problems that occurred were due to minor accidents. In fact the TSS-1R mission has demonstrated that electrodynamic tethers can convert momentum into electrical power of order of several kW. Additionally, as a consequence of the tether being cut, the satellite dedicated to electrons collection, upward deployed (and over-speeded), was able to rise 140 Km above the space shuttle (original tether length 20 Km) giving a spectacular demonstration of the momentum exchange concept.

In 1996 another mission called TiPS (tether physics and survivability experiment) wasflown. Its orbit altitude (1022 Km) lied in the so-called "death valley" because of the strong presence of orbital debris and micrometeoroids. Although the tether (length = 4 Km, diameter = 2 mm) could have been severed by a 1 mm particle coming at a relative velocity of about 14 Km/s, it remains there intact.

These brief comments about tethers missions suggest potential applications to Mars missions. There are two main propulsion concepts associated with space tethers. They are

- Electrodynamic Tethers;
- Momentum Exchange Tethers.

While the Isp of such systems is formally infinite, tethers require energy or electrical power (e.g., to drive the currents needed inside the wires). Unfortunately it is not possible to use electrodynamics systems for both interplanetary trajectory and Mars orbit, due to following reasons:

- the interplanetary magnetic field is too weak ($\sim 50\ \mu T$) to produce a useful current;
- Mars does not have a dipole magnetic field but just crustal residual magnetization, useless for electrodynamic tethers).

Thus only mechanical tethers could be useable in a whole mission, in particular as space elevators and to generate artificial gravity.

4.2.1. Mars orbit elevator

Space elevators can be conceived based on a tether in Mars orbit along which a spacecraft can move in order to reach different orbits without any propulsion system. Such tether is subjected only to aerodynamic drag (very low in a microgravity environment) so that it is possible to use a very low level of power to move spacecraft along the tether.

As a first approximation a drag force of about $5.6\ \mu N$ is assumed (altitude of 200 km in the Earth atmosphere) as the upper limit atmospheric drag for Mars (see Fig. 3). The Mars Stationary Orbit is at 17,000 km of altitude, the upper limit of Martian atmosphere is about 700 km, and the behavior of tether tension vs. spacecraft mass is shown in Fig. 4. The main advantages of such an elevator system are

- a system which permit to change orbit without propellant consumption;

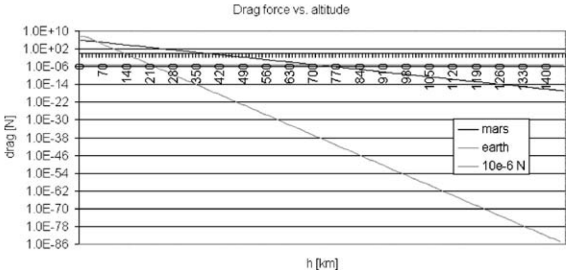

Fig. 3. Drag force vs. altitude for Earth and Mars.

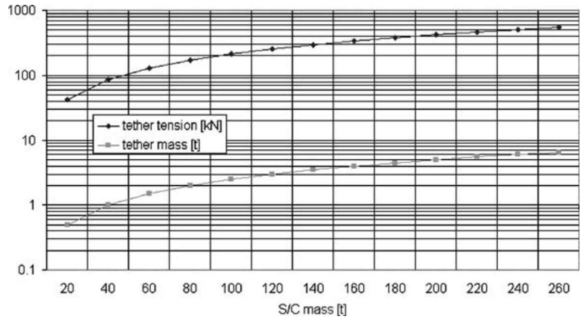

Fig. 4. Tether tension and mass vs. spacecraft mass in MSO.

- a system capable to perform communications needs, using the upper tether like a stationary satellite;
- a system to be used as escape tower (think to a crew emergency system to be deployed up to ground).

Fig. 4 shows the low tension during space elevator operation: the maximum tension [5] undergone by a tether between two spacecraft 120 tons each, is 255 kN. The weight of the tether 16,300 km long, assuming Kevlar™ as material ($T = 2$ GPa and $\rho = 1440$ kg/m^3), has been estimated in about 3 tons. Using Spectra™ ($T = 4$ GPa and $\rho = 970$ kg/m^3) as material [6], for the tether we obtain a mass of about one ton.

4.2.2. *Artificial gravity generator*

A rotating tether could induce artificial gravity during interplanetary flights, a very important issue in the frame of a M3 scenario, being a manned mission subject to very restrictive environmental constraints. Fig. 5 shows the relationship between spin and tether length for different induced gravity (1, 0.8, and 0.4 g, the last is about the same as that on Mars surface).

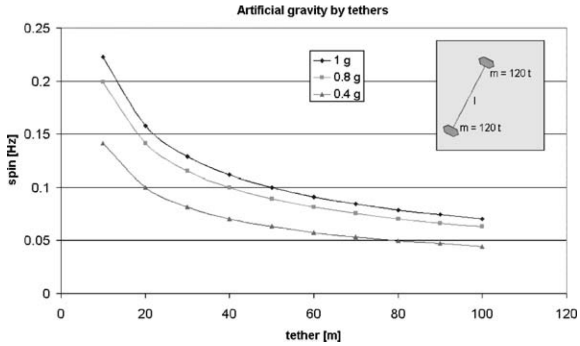

Fig. 5. Tether spin during interplanetary flight for gravity generation.

There are clearly many aspects to be studied in this area, both technological and operational that shall be further studied in the future because of their importance at architecture level. At the moment the primary question is: are rotating tethers compatible with continuous thrust during an interplanetary trajectory, without producing instabilities on the whole system? This question is still open.

4.3. *Superconductivity*

In chemical propulsion the exhaust velocity is limited by thermodynamics to a few thousands m/s. This limits performance according to the Tsiolkovsky relations:

$$\Delta v = v_e \ln(m_0/m_f);$$

where $m_f/m_0 = e^{-\Delta v/v_e}$

indicating a need for high exhaust velocities (v_e); for missions requiring large Δv (in which we still want initial mass about the same of final mass, $m_f \cong m_0$), the exhaust velocity must be high. Exhaust velocities above 10,000 m/s could be achieved by an Electric Propulsion (EP) system.

EP systems based on electromagnetic thrusters are capable of much higher exhaust velocity when compared to chemical propulsion. However, one must keep in mind the limit imposed on v_e by the "power supply (PS) penalty". It can be seen that an increment in v_e will lead to a reduction in propellant mass, but, at the same time, will increase the PS mass. Therefore, for a given mission Δv there is an optimum exhaust velocity minimizing the propellant plus power supply total mass. Achieving higher exhaust velocities without an excessive PS mass penalty could be done by reducing the specific power plant mass; this means using, wherever possible, lighter materials; a second possibility specifically for electromagnetic thrusters, is using superconductive (SC) materials for the coils. Thus, among all electric propulsion systems we concentrate on applying steady-state electromagnetic (e.g., Lorentz force) propulsion; this kind of propulsion also yields the highest exhaust velocities and largest thrust densities compared to other EP systems. Applying SC technology can lead to serious benefits for EP in terms of volume and weight savings, improvements in EP performance, and slow cathode erosion rates. The basic feature of SC materials is the property of having, at temperature less than a critical value, resistivity and magnetic permeability values close to zero. It is known now that there are two types of SC: those whose resistivity goes to zero at T close to a few K (LTSC, thus when immersed in LHe, or LH_2), and the recent family of SC which can be used as such at much higher T, in practice requiring LN_2 temperatures (HTSC). This enables high current densities flowing through the SC coils, which are only limited by the magnetic field (induced or externally applied) in which the SC is immersed; in fact, if the magnetic field is greater than a critical value, the SC state is destroyed and the conductivity rises to "normal" values. These high currents do not need a voltage source to be maintained: at zero resistivity a starter circuit creates them after which they keep circulating indefinitely. Thus a magnetic field can be created without Ohmic losses. The high current densities imply much larger B fields can be generated than possible with conventional conductors. Thus Lorentz force electric thrusters could be in principle designed in which fields, and therefore thrust, could be dramatically higher with a simultaneous reduction in coil power requirements and weight [7].

SC technology in EP can only be utilized to applied field MPD thrusters (AF-MPDT) and Hall thrusters (HT). In these propulsion systems, the electrical coils, needed to generate the magnetic field, can be made of a SC material, instead of using conventional materials such as copper. Noting that the number of turns is inversely proportional to current density, it is evident that the use of SC materials lead to a volume saving.

Comparing the current density flowing in a SC coil made of NbTi, with that of copper, we see that, for a B field in the range of 0–1 T, there is a difference of three orders of magnitude; the same reduction goes for the number of turns, and, thus for the volume needed by the coil. The same estimate holds for weight savings.

As an example, consider a coil having the following features:

Length: $L = 0.1$ m;
Diameter: $D = 0.14$ m;
Wire diameter: $d_w = 1$ mm.

Tables 1–4 report the results of calculations performed for copper and a HTSC material (NbTi), varying the B field in the range of 0.1–0.6 T (common for satellite EP)

Even for current density well below the critical value, use SC material is still advantageous; this can be seen in Table 3. Conversely, SC materials can be utilized to produce higher B fields for a fixed (low) weight and/or volume, as shown in Table 4.

High B fields, possible with SC materials, lead to an increment of Lorentz force ($F_1 = J \times B$), which is exploited as thrust. Fig. 6 shows thrust relative to the B field in the 1–7 T range. To perform the calculations the following assumption have been made:

1. Arc current $I = 400$ A [8].
2. Anode–cathode distance $I = 0.02$ m [9].
3. Gas (Ar) flow rate mdot = 100 mg/s [10].

So, the following conclusions can be drawn:

- For a magnetic field $B = 7$ T a Lorentz force $F = 56$ N can be achieved, with an arc current $I = 400$ A and a SC coil weighing about 4 Kg. Utilizing a copper coil weighing the same, only $B = 0.05$ T would be obtained, with a corresponding Lorentz force $F = 0.4$ N.
- I_{sp} up to 56,000 s can be achieved.
- This performance can be obtained with a relatively low arc current ($I = 400$ A), that is, with a reduction in cathode erosion rate.

Moreover the high B fields could perhaps be exploited to replace the physical nozzle of the thruster with a "magnetic nozzle"; if this indeed possible it is conceivable to obtain a reduction in gasdynamic heat losses through the walls. Scaling laws for wire

Table 1
Coil characteristics for copper winding

B field	T	0.05	0.1	0.2	0.3	0.4	0.5	0.6
Current density	A/mm^2	10	10	10	10	10	10	10
Total wire length	m	637	1273	2546	3820	5093	6366	7639
Total wire resistance	Ω	13.78	27.56	55.12	82.68	110.24	137.8	165.36
Required power	W	850	1700	3400	5100	6800	8500	10200
Total wire weight	Kg	4.465	8.930	17.86	26.79	35.72	44.65	53.580
Winding layers		15	29	58	87	116	145	174

Table 2
Coil characteristics for HTSC NbTi

B field	T	0.05	0.1	0.2	0.3	0.4	0.5	0.6
Current density	A/mm^2	7500	7125	6750	6375	6000	5625	5250
Total wire length	m	1	2	4	6	8	11	15
Total wire resistance	Ω	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Required power	W	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Total wire weight	Kg	0.004	0.009	0.018	0.029	0.041	0.054	0.070
Winding layers		1	1	1	1	1	1	1

Table 3
Coil characteristics for SC material (NbTi) for “low” current densities

B field	T	0.05	0.1	0.2	0.3	0.4	0.5	0.6
Current density	A/mm^2	300	300	300	300	300	300	300
Total wire length	m	21	42	85	127	170	212	255
Total wire resistance	Ω	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Required power	W	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Total wire weight	Kg	0.102	0.204	0.409	0.613	0.817	1.022	1.226
Winding layers		1	1	2	3	4	5	6

Table 4
Coil characteristics for SC material (NbTi) for high B field

B field	T	1	2	3	4	5	6	7
Current density	A/mm^2	4000	3525	3075	2625	2175	1725	1050
Total wire length	m	32	72	124	194	293	443	849
Total wire resistance	Ω	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Required power	W	0.00	0.00	0.00	0.00	0.00	0.00	0.00
Total wire weight	Kg	0.153	0.348	0.598	0.934	1.409	2.132	4.087
Winding layers		1	2	3	5	7	11	20

weight indicate that, everything being equal, weight is inversely proportional to wire diameter, with an additional nonlinearity being due to the nonlinear dependence of the maximum J_c on the induced B field itself [2].

To conclude: current SC technology can be put to good use for future electric thrusters, with the goal of reducing power needs and weight of the coil volume and complexity. It is strongly suggested that combining emerging miniature cryostat technology

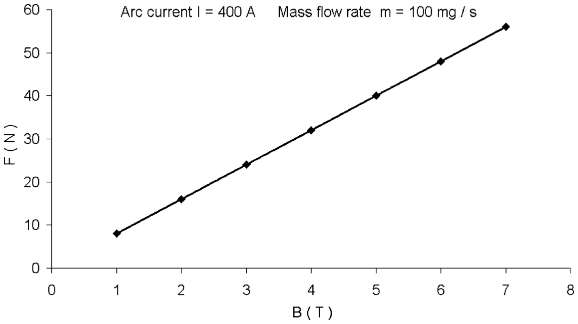

Fig. 6. Thrust vs. magnetic field using superconductivity.

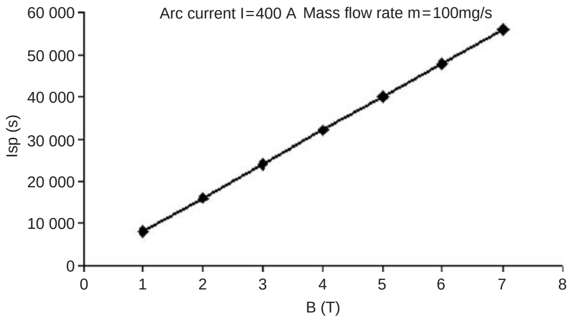

Fig. 7. Specific impulse vs. magnetic field using superconductivity.

with HTSC and LTSC technology could result in electric thrusters substantially smaller than currently available. Fig. 7 shows thrust and specific impulse vs. *B*: it is easy to understand what are the advantages one can obtain using superconductivity in high performance missions like M3.

4.4. In situ resources utilization concepts

Widely known concepts consists in A. using Carbon Dioxide (CO_2) in the Martian atmosphere for conversion into other chemical components that can be burned in a rocket engine. Alternatively, B., using the Martian atmosphere in a CO_2 breathing engine, burning the CO_2 directly with on-board fuels (as in a turbojet). These two concepts can be summarized as follows:

A. Extract carbon dioxide from the Martian atmosphere (concentration of $CO_2 = 95\%$) and convert it into other chemical species that can be burned into a rocket engine, e.g.:

- Oxygen (O_2) as oxidizer and
- Methane (CH_4) or Methanol (CH_3OH) as fuels. Oxygen (O_2) can be generated directly from carbon dioxide through an oxide-electrolyzer (e.g. with a solid ceramic zirconia electrolyte, see below). The generation process of Methane or Methanol from CO_2 on the other hand needs hydrogen (H_2), which has to be transported from Earth. Recent information on Mars surface indicates, however, the presence of substantial water. Hydrogen could be extracted from Martian water.

B. Using the Martian atmosphere in "air breathing" engines, which means burning the carbon dioxide (CO_2) directly with on-board metallic fuels (Mg, Al). A possible propellant combination would be:

- Carbon dioxide as the oxidizer and
- Magnesium (Mg) powder as the fuel.

The CO_2 of the Mars atmosphere offers the possibility of in situ-propellant utilization by burning metallic fuels to be carried from earth with CO_2. Studies published by researchers from the Russian Academy of Science, Chermogolovka (Moscow) have shown the feasibility of Mars sample return missions. Calculations have shown that by burning Mg or Al with CO_2 as oxidizer, Isp between 200 and 230 s are possible under ideal conditions. Based on these values different scenarios with a direct return from Mars were studied proving the theoretical feasibility of this propulsion concept for Martian spacecraft applications. Problems areas to be studied are the oxidizing process of Mg or Al particles and the combustion stability.

4.4.1. Mars atmosphere processing

Historically, the terms In-Situ Propellant Production (ISPP) or In-Situ Resource Utilization (ISRU) have been applied to processes designed to extract O_2 from the CO_2 in the Martian atmosphere. Two approaches have been considered for this process [10].

4.4.1.1. Zirconia cell process. In the zirconia process (Fig. 8), the critical step involves the extraction of oxygen (O_2) from the CO_2, O_2, and carbon monoxide (CO) gas mixture formed by thermal dissociation of CO_2.

$$CO_2 \rightarrow CO + 1/2 O_2.$$

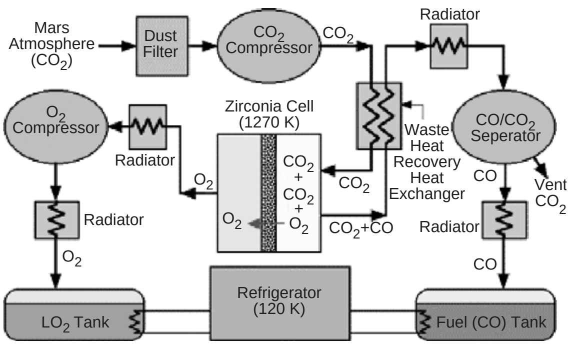

Fig. 8. Zirconia process.

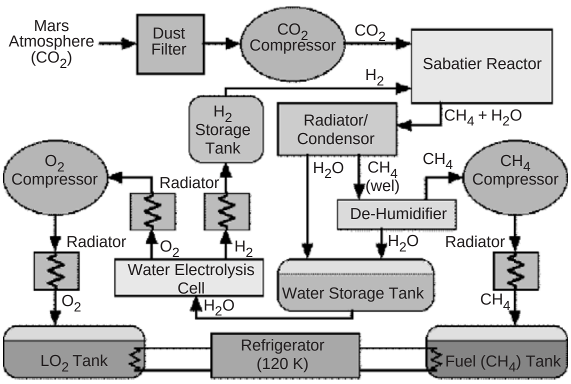

Fig. 9. Sabatier/Water Electrolysis process.

Extraction is accomplished by means of a zirconia membrane which has the property of transporting oxygen ions through its lattice at high temperatures when the appropriate voltage is applied across the membrane. The O_2 can then be used for propellant for the return rocket in a robotic sample return mission with a fuel (e.g., propane, C_3H_8) brought from Earth. In a piloted mission, the O_2 can be used for life support as well as propulsion. The CO produced in the process can also be used for fuel.

4.4.1.2. Sabatier/Water Electrolysis. The CO_2 can also be reacted with hydrogen in the Sabatier/Water Electrolysis (S/WE) process to produce methane (CH_4) and oxygen if a source of hydrogen (H_2) can be provided (either brought from Earth or indigenous). The process involve two reactions (Fig. 9). The first involves the catalytic reduction of CO_2 by H_2 in a Sabatier reactor to produce methane and water. This is followed by electrolysis of the water, with the H_2 recycled to the Sabatier reactor:

$$CO_2 + 4H_2 \rightarrow CH_4 + 2H_2O$$

$$2H_2O + \text{electricity} \rightarrow 2H_2 + O_2.$$

The overall S/WE process may then be written:

$$CO_2 + 2H_2 \rightarrow CH_4 + O_2.$$

The O_2 and CH_4 are then stored for use as rocket propellant. In all these schemes, the water would be electrolyzed in an electrolysis cell: for the extraterrestrial sources, the water would be freed by digging and "baking" the wet soil or ice to produce water vapor, condensing the steam vapor to produce liquid water, and finally filtering or purifying the water to prepare it for electrolysis.

Much of the required water electrolysis technology has been developed for terrestrial applications, such as hydrogen generation or for submarine life-support (O_2 generation). Water electrolysis systems for space uses are being developed for Space Station life-support systems. In these, carbon dioxide (CO_2) produced by crew respiration is reduced with H_2 to give water and carbon (Bosch process) or methane (Sabatier process). The water is then electrolyzed to produce oxygen for the crew and to regenerate the hydrogen needed for CO_2 reduction. Losses in the system (O_2 or H_2) are made-up by electrolyzing excess water from the crew ("wet" food, metabolism, or washing) or from excess water used in the laboratory modules.

A number of water electrolysis technologies are available for space applications; one system developed by Hamilton Standard uses a solid polymer electrolyte (SPE) for the electrolysis cell. This system has the advantage that liquid or gas phase water can be electrolyzed, and the O_2 and H_2 gasses can be output from the cell at high pressures. Also, no strong acid or alkaline solutions are required as in conventional systems (with platinum, etc. electrodes). The technology of water electrolysis is well established; the major technology unknowns are those associated with extracting the water from an extraterrestrial source. These issues include those of digging or drilling in a low or milli-gee environment (e.g., on an asteroid), separating the water from the soil (e.g., bulk thermal baking versus microwave heating), and liquefaction (e.g. cryo-refrigerators).

4.4.2. Chemical propulsion on Mars

For O_2/H_2 thrusters the primary requirement is to operate at higher oxidizer-to-fuel (O/F) ratios than those commonly in use (e.g., Space Shuttle main engine O/F = 6, Centaur RL-10 O/F = 5). For example, stoichiometric water electrolysis produces O_2/H_2 at an O/F of 8; even higher O/F values might be useful in order to save hydrogen for non-propulsive uses (e.g., as chemical reducing agents for lunar regolith processing). Also, the needs for H_2 is due to chemical reaction in the Sabatier cycle for methane production.

Thruster technologies for robotic Mars ascent/return vehicles using propellant produced on the surface are likely to involve small O_2/propane thrusters with propane brought from Earth. Current NASA O_2/hydrocarbon thruster test-bed activities are aimed at developing large engines (100 times higher thrust than is needed for robotic Mars missions) for future launchers and are not directly transferable to robotic Mars exploration needs.

One issue associated with the use of the Sabatier/Water Electrolysis process for robotic missions is the need to develop O_2/CH_4 engines: methane fuel may cause coking in small engines, although this should not be an issue for the larger engines required for manned missions. Small O_2/CO thrusters are under development at NASA LeRC.

4.4.3. In situ associated technologies

Starting from what said above, the main task is to perform an assessment of the most promising technologies and critical areas related to ISPU. This means to be able to identify what can be made at both technological and Mars knowledge levels in order to realize a first low cost mission (Italian-based?) on Mars by using in situ concepts. To do so, a feasibility study for mid term ISPU is necessary. The scope of this section is:

A. to evaluate the effective capability/possibility of in situ concepts for Manned/Unmanned Mission to Mars.
B. to assess related technological development/testing phases.

As currently envisioned, a large part of the cost of a Martian outpost will be that of ferrying supplies and propellant from Earth. The major component of this propellant is liquid oxygen (LOX). NASA's "Report on the 90-Day Study on the Human Exploration of the Moon and Mars" estimated that the amount of mass launched to low Earth orbit (LEO) could be reduced by 300 tons per year if LOX were produced on the Moon. This is equivalent to 10 Shuttle launches at a cost of a several billion dollars per year. Even if we develop a new heavy lift launch vehicle (HLLV), the price tag to launch this propellant will be huge. By producing the oxygen on the planet where it is actually needed there is no shipping penalty.

In 1987 the NASA "Ride Report" stated that, "Exploring and prospecting the Moon, learning to use lunar resources and work within lunar constraints, would provide the experience and expertise necessary for further human exploration of the solar system." It went on to assert that, "There is no doubt that exploring, prospecting, and settling Mars should be the ultimate objectives of human exploration.

When we return to the Moon and proceed to Mars, we are not going there solely to utilize the resources. We are going

- to satisfy our need to explore, to strive, to seek, to find,
- to increase the pool of scientific knowledge,
- to enhance our understanding of life in the universe and to find out if life once existed on Mars, and
- to improve Earth's political and economic conditions.

However, using Mars resources can have a major effect on the way we proceed, the cost of the program, its timetable, milestones along the way, and ultimately on whether the program is successful or not.

The real test of whether a program to put people permanently into space is successful is whether they stay. And people will only stay on the Moon and Mars if they learn how to use local resources to make their settlements permanent. Otherwise, the continual cost of supplying everything from Earth will become too great a burden, and these settlements will be abandoned.

The timing, or phasing-in, of ISMU will be a natural evolution of productivity as power levels and capabilities increase. As in any market, the needs of the outpost will dictate what products are produced. Many technologies already exist and need only be modified for use in space. Robotic units sent ahead of crews will perform critical experiments. These units will be followed by engineering prototypes for the demonstration and verification of technologies and products. Each step forward will provide minimal risk and will have graceful upgrades and safe fallbacks. As pressurized living space is expanded, the material necessary for radiation protection shall, of necessity, be produced. This will also provide material for other uses at the outpost. The technology and vehicles used for mining this material will later be used to mine feed-stock for a lunar LOX plant and perhaps burrow under the surface for the creation of inhabitable tunnels. During this time the technology necessary for metal and light gas production will be developed, providing the flexibility for whatever avenues we choose to follow in our exploration of space, with little support required from Earth.

The timing of the use of Mars resources must fit into the overall mission of a lunar and Mars program. In some mission “architectures,” only simple products will be possible because of the limited power, mass, and logistics available to produce them. Other schemes can be envisioned where the first flights bring large numbers of robotic processing plants so we can build up the outpost from local materials. A scaled-back, man-tended base would provide the core for this bootstrapping activity. Only after we can refuel landing vehicles and build structures would we attempt to bring down large payloads or increase stay times. A number of similar concepts of varying complexity exist.

5. The technological challenge

One of the most interesting concept relative to Propulsion System for Mars missions (especially for M3) is the possibility to integrate different technologies in order to achieve a higher benefit. In this paper one of major task is the evaluation of such a concept. This is a job not only at technological level but also at system level, in fact using IPS means to be able to obtain from two systems much more than one can obtain from a single technology. For example, about 50% of energy is loss from a Rubbia’s engine, but using MPD with superconductors could be possible to produce additional thrust for both main thruster and artificial gravity systems. Thus this paper identifies two key areas that research should address:

1. Possibilities of coupling Nuclear Propulsion and superconductive MPD Thrusters.
2. Systems for energy storage.

Both could be important items in view of a M3 because of their repercussions on the whole mission architecture. This could be considered as a novel idea in the field of space missions, especially as regard manned Mars missions.

6. Mission to Mars: a possible strategy

One of the most important tasks of our paper is to show that IPS provides an interesting alternative way to go to Mars. To do so we need to demonstrate that IPS allows a short travel time and, at the same time, an efficient way to use MPD and Nuclear Propulsion together.

The importance of a short trip time is clear to everybody: it reduces crew exposure to radiation, it reduces unusual lifestyle in restricted space, it increases useful time on Mars for scientific purposes, and reduces the total mission duration to an extremely short interval (for the first mission at the least).

Starting from the mass breakdown shown in Table 5, we can consider an initial mass of about 380 tons and an arriving mass on Mars of about 240 tons. Our simulations are based on the following hypotheses:

- mission starts at about 300 km above Earth surface
- spheres of influence are considered with Tisserand equation
- perturbations from all planets in the Solar System are considered
- 4 superconductive MPD thrusters are used, the thrust vector being constantly aligned with the anti-velocity direction
- the behavior of the rigid body tether system during the interplanetary trajectory is not considered
- for the input data refer to Table 6.

Table 5
Mass breakdown for the Nuclear-SC/MPD-Tether architecture (approximate)

Subsystem	Mass [tons]
Crew module (interplanetary)	31
Descent stage (+propellant)	16
Aeroshell and parachute	4
Crew module (surface and ascent)	4
Surface payload	3
Ascent stage (dry mass)	4
Ascent propellant	14
Propulsion system	35
Tanks	18
Outbond LH_2	132
Return LH_2	92
Tether system	13
Superconductive MPD	12
Total	378

Table 6
Input data for the simulation

Assumed data	Note	Dimension	Value
Mass	Initial	[t]	390
Isp	Rubbia's engine	[s]	3500
Mdot	Rubbia's engine	[g/s]	30
Isp	SC-MPD	[s]	56,000
Mdot	SC-MPD	[g/s]	0.1

Note that for the Rubbia's engine a lower specific impulse should be considered (2500 s instead of 3500 s), this because of energy losses due to radiation.

As regards the mass increment due to the superconductive MPD, from Tables 1–4 it is possible to see the low amount of mass for these systems. Also, it is important to note that for conventional MPD systems the specific power is about 100 W/mN [7], but including technological growth and superconductivity, we can assume that using 15 MW we can still perform the complete mission!

By integrating the equations of motion by Runge–Kutta method, taking continuous tangential thrust into account, it is possible to obtain a trajectory which shows the capability of Rubbia's engine coupled with SC-MPD to perform the mission. In Fig. 10 it is possible to see a simulation for the escape trajectory, while in Figs. 11 and 12 it is possible to see the simulations

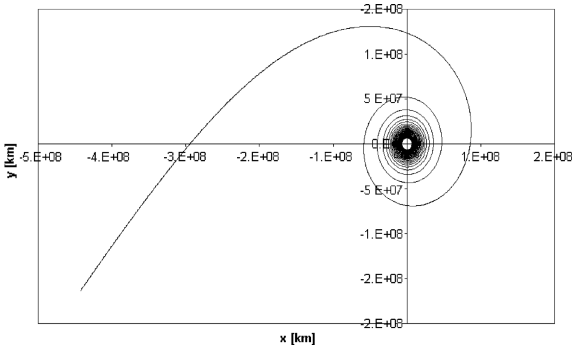

Fig. 10. Earth escape trajectory using Rubbia's engine together with four SC-MPD.

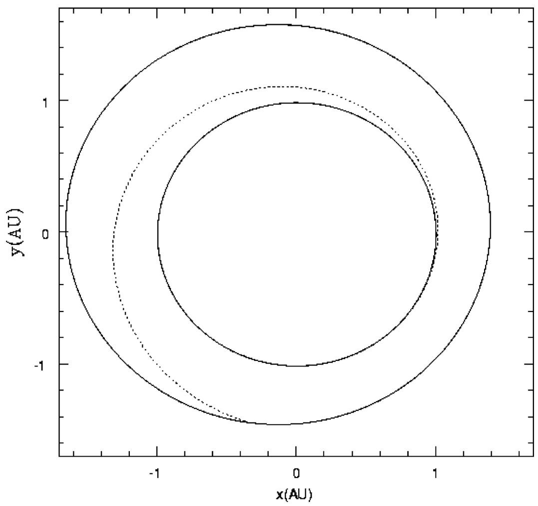

Fig. 11. Earth–Mars trajectory using single SC-MPD.

performed using one and four SC-MPD respectively. In the first case the trip time is about 6 months (arriving with an eccentricity $e = 0.19$), in the second about 2 months ($e = 0.3$). Further studies will be needed to optimize the trajectory (particularly using a optimum Mars approach, because in the simulations carried out so far the insertion velocity is unacceptably large). Table 7 shows the mission summary.

7. Conclusions and recommendations

First result is that using Rubbia's engine coupled with SC-MPD, it is possible to reduce trip time from

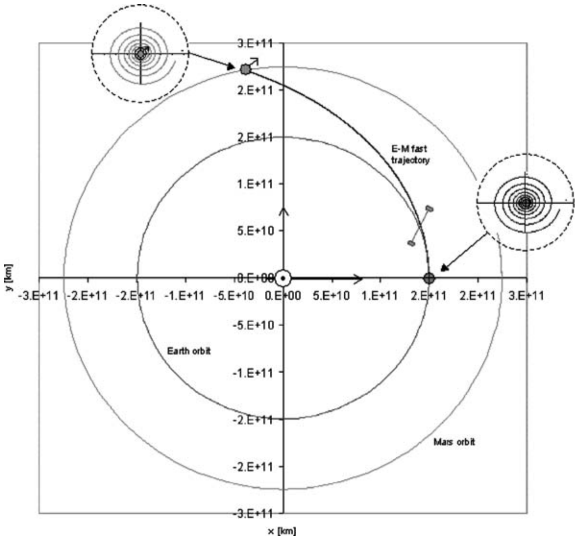

Fig. 12. Earth–Mars trajectory using four SC-MPD.

Table 7
Mission summary

Mission event	Value
Starting altitude on Earth orbit	400 km
Total thrust	1974 N
Escape time	17.34 d
Earth distance	693, 276 km
Spacecraft mass (close to LEO)	400 t
Hyperbolic velocity	2.781 km/s
Heliocentric initial velocity	32.6 km/s
Total thrust	224 N
Spacecraft mass (close to Earth)	325 t
Trip time	119 d
Mars approach velocity	30.15 km/s
Spacecraft mass (close to Mars)	240 t

150 days [3] to 119 days, although in our simulation an optimization is foreseen. An other result is that SC-MPD is capable to provide thrust using the power in excess from Rubbia's engine.

In conclusion, there are some consideration to be made about benefits of such a mission:

- Mars colonization with reduced energy problems;
- Tethers as new space frontiers (not just electrodynamics);
- Rubbia's engine as a way to produce energy;
- Management/control of large space structure;
- Use of electric propulsion using SC;
- Space elevator in MSO;
- ISPU could enable a new approach toward the use of local resources;
- Contribution to Mars knowledge (i.e.: has Mars the secret of life?);
- Potential spin-off of the involved technologies in "non-space" areas.

Future activities shall be directed toward two topics: first, the coupling between Rubbia's Engine and SC-MPD propulsion systems from a system and technological point of view; second, to perform a trade-off among different M3 scenarios in order to determine the best (taking into account the special constraints associated with human missions).

Further studies shall be directed towards the analysis of both conceptual and technological feasibility, and further studies are recommended especially for three important areas:

1. Rubbia's engine: coupling between Nuclear and superconductive-MPD and/or application of power in excess in other subsystems.
2. Application of Tether systems other than electrodynamic ones.
3. Feasibility of ISRU methods for Mars missions applications.

Needless to say, this would pose significant engineering challenges from technological standpoint, and not only.

For further reading

[11] R.W. Bussard, R.D. De Lauer, Nuclear Rocket Propulsion, McGraw-Hill Book Company, Inc., New York, 1958.
[12] E.E. Rice, Advanced System Concept for Total ISRU-Based Propulsion and Power Systems for Unmanned and Manned Exploration—Final Report, NIAC-PHASE I CONTRACT, Research Grant 07600-020, January 2000.
[13] NASA Human Exploration of Mars: The Reference Mission of the Mars Exploration Study Team, 1997, including Addendum Version 3.0, NASA SP-607, June 1998.
[14] A.G. Accettura, F. Malgarini, IAF-00-Q.2.06, Integrated Solar System Exploration Strategy (ISSES), 51st International Astronautical Congress, 2–6 October 2000, Rio de Janeiro, Brasil.

[15] Personal conversation with Mme Cristina Piva.

[16] M. Reichert, W. Seboldt, D. Koenies, M. Leipold, A. Bichi, J. Gonzalez, Ch. Savage, M. Novara, IAA-99-IAA.13.3.03, Comparison of Promising Concepts for the First Manned Mars Mission, 50th IAF Congress, 4–8 October 1999, Amsterdam, The Netherlands.

[17] C. Bruno, P.A. Czysz, An Electro-Magnetic-Chemical Hypersonic Propulsion System, AIAA Paper 98-1582.

References

[1] M. Augelli, G. Bignami, C. Bruno, C. Rubbia, et al. (1999), Report of the Working Group on a Preliminary Assessment of a New Fission Fragment Heated Propulsion Concept and its Applicability to Manned Missions to the Planet Mars (Project 242), ASI Internal Report, Roma, March 15, 1999 (Proprietary).

[2] C. Bruno, S. Giucci, Cryogenic Technology to Improve Electric Thrusters, Paper IAF-99-S.4.04, Presented at the 50th IAF Congress, Amsterdam, October 4–8, 1999.

[3] M. Augelli, A. Pellizzoni, B. Procacci, Project 242: Interplanetary Missions and Heavy Launch Vehicles, 12th European Aerospace Conference, Paris, November 29–30, December 1, 1999.

[4] R. Zubrin, The Economic Viability of Mars Colonization, report issued by Lockheed Martin Astronautics.

[5] F.P.J. Rimrott, Introductory Orbit Dynamics, Vieweg, Wiesbaden, 1989.

[6] A.G. Accettura, A. Ferretti, NTEPRP10000, PROPULSION 2000—Phase I Final Report, Rome, November 2000.

[7] G.P. Sutton, Rocket Propulsion Elements, Wiley, New York, 1976 (Chapter 19).

[8] S. Giucci, Propulsione satellitare: linee di tendenza attuali e applicazioni future della superconduttività, DMA Thesis, University of Rome “La Sapienza”, Rome, Italy (in Italian).

[9] J.S. Sovey, M.A. Mantenieks, Performance and lifetime assessment of magnetoplasmadynamic arc thruster technology, J. Propulsion Power, 7 (1991) 71–83.

[10] http://sec353.jpl.nasa.gov/apc